AERONAUTICS

SIXTH ANNUAL REPORT

OF THE

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

1920

INCLUDING TECHNICAL REPORTS
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LETTER OF TRANSMITTAL.

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS,
Washington, D. C., November 20, 1920.

The PRESIDENT:

In compliance with the provisions of the act of Congress approved March 3, 1915 (naval appropriation act, Public, No. 273, 63d Cong.), I have the honor to transmit herewith the Sixth Annual Report of the National Advisory Committee for Aeronautics, including a statement of its expenditures for the fiscal year ending June 30, 1920.

In addition to the exercise of its prescribed functions in the field of scientific research in aeronautics, the National Advisory Committee for Aeronautics has, during the past year, given special consideration to the question of organization of governmental activities in aeronautics, and has effected a coordination of views on this subject between the military and naval air services and other governmental agencies concerned. The agreements reached have been given definite expression in a draft of legislation providing for the establishment of a Bureau of Aeronautics in the Department of Commerce for the regulation and encouragement of commercial air navigation. In this connection attention is invited to that section of the report entitled "Organization of Governmental Activities in Aeronautics."

The attention of the President and of the Congress is especially invited to the closing section of the report, entitled "A National Aviation Policy," and the specific recommendations of the National Advisory Committee for Aeronautics therein set forth as to the legislative steps which in its judgment are necessary to carry such a national aviation policy into effect.

Respectfully submitted.

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS,
CHARLES D. WALCOTT, *Chairman.*

LETTER OF SUBMITTAL.

To the Senate and House of Representatives:

In compliance with the provisions of the act of March 3, 1915, making appropriations for the naval service for the fiscal year ending June 30, 1916, I transmit herewith the Sixth Annual Report of the National Advisory Committee for Aeronautics for the fiscal year ended June 30, 1920.

The attention of the Congress is invited to the recommendation of the National Advisory Committee for Aeronautics for the establishment of a Bureau of Aeronautics in the Department of Commerce for the regulation and encouragement of commercial aviation. The national aviation policy as formulated by the National Advisory Committee for Aeronautics and the constructive recommendations therein set forth for the consideration of the Congress have the hearty approval of the departments concerned as well as myself.

WOODROW WILSON.

THE WHITE HOUSE,
7 December, 1920.

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS.

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Charles D. Walcott, Sc. D., *Chairman,*
Secretary Smithsonian Institution, Washington, D. C.
S. W. Stratton, Sc. D., *Secretary,*
Director Bureau of Standards, Washington, D. C.
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Chief Engineering Division, Air Service, Dayton, Ohio.
T. T. Craven, Captain, United States Navy,
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Chief United States Weather Bureau, Washington, D. C.
Charles T. Menoher, Major General, United States Army,
Chief of Air Service (War Department), Washington, D. C.
Michael I. Pupin, Ph. D.,
Professor of Electro-Mechanics, Columbia University, New York City.
D. W. Taylor, Rear Admiral, United States Navy,
Chief Constructor, Navy Department, Washington, D. C.
Orville Wright, B. S.,
Dayton, Ohio.

EXECUTIVE COMMITTEE.

Joseph S. Ames, *Chairman.*
S. W. Stratton, *Secretary.*

Thurman H. Bane.
T. T. Craven.
John F. Hayford.
Charles F. Marvin.
Charles T. Menoher.
D. W. Taylor.
Charles D. Walcott.

SIXTH ANNUAL REPORT

OF THE

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS.

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS,
Washington, D. C., November 20, 1920.

To the Congress:

In accordance with the provision of the act of Congress, approved March 3, 1915, establishing the National Advisory Committee for Aeronautics, the committee submits herewith its Sixth Annual Report. In this report the committe has described its activities during the past year, the technical progress in the study of scientific problems relating to aeronautics, the assistance rendered by the committee in the formulation of a policy regarding the organization of governmental activities in aeronautics, the coordination of research work in general, the examination of aeronautical inventions, and the collection, analysis, and distribution of scientific and technical data. This report also contains a statement of expenditures, estimates for the fiscal year 1922, and a discussion of a national aviation policy with certain specific recommendations for the consideration of Congress.

FUNCTIONS OF THE COMMITTEE.

The National Advisory Committee for Aeronautics was established by act of Congress, approved March 3, 1915. The organic act charges the committee with the supervision and direction of the scientific study of the problems of flight with a view to their practical solution, the determination of problems which should be experimentally attacked, their investigation and application to practical questions of aeronautics. The act also authorizes the committee to direct and conduct research and experimentation in aeronautics in such laboratory or laboratories in whole or in part as may be placed under its direction.

Supplementing the prescribed duties of the committee, its broad general functions may be stated as follows:

First. Under the law the committee holds itself at the service of any department or agency of the Government interested in aeronautics, for the furnishing of information or assistance in regard to scientific or technical matters relating to aeronautics, and in particular for the investigation and study of problems in this field with a view to their practical solution.

Second. The committee may also exercise its functions for any individual, firm, association, or corporation within the United States, provided that such individual, firm, association, or corporation defray the actual cost involved.

Third. The committee institutes research, investigation, and study of problems which, in the judgment of its members or of the members of its various subcommittees, are needful and timely for the advance of the science and art of aeronautics in its various branches.

Fourth. The committee keeps itself advised of the progress made in research and experimental work in aeronautics in all parts of the world, particularly in England, France, and Italy, and will extend its efforts in the securing of information from Germany, Austria, Canada, and other countries.

Fifth. The information thus gathered is brought to the attention of the various subcommittees for consideration in connection with the preparation of programs for research and experimental work in this country. This information is also made available promptly to the

military and naval air services and other branches of the Government, and such as is not confidential is immediately released to university laboratories and aircraft manufacturers interested in the study of specific problems, and also to the public.

Sixth. The committee holds itself at the service of the President, the Congress, and the executive departments of the Government for the consideration of special problems which may be referred to it, such as organization of governmental activities in aeronautics, recommendations as to proper action under the Convention for the Regulation of International Air Navigation, questions of policy regarding the regulation and development of civil aviation, advanced education in aeronautical engineering, etc.

ORGANIZATION OF THE COMMITTEE.

The committee has 12 members, appointed by the President. The law provides that the personnel of the committee shall consist of two members from the War Department, from the office in charge of military aeronautics; two members from the Navy Department, from the office in charge of naval aeronautics; a representative each of the Smithsonian Institution, of the United States Weather Bureau, and of the United States Bureau of Standards; and not more than five additional persons acquainted with the needs of aeronautical science, either civil or military, or skilled in aeronautical engineering or its allied sciences. All members as such serve without compensation.

During the past year Mr. Orville Wright was appointed by the President to membership on the committee to succeed Dr. John R. Freeman, resigned.

The full committee meets twice a year, the annual meeting being held in October and the semiannual meeting in April. The present report includes the activities of the committee between the annual meeting held on October 9, 1919, and that held on October 7, 1920.

The present organization of the committee is as follows:

Charles D. Walcott, Sc. D., chairman.
S. W. Stratton, Sc. D., secretary.
Joseph S. Ames, Ph. D.
Maj. Thurman H. Bane, United States Army.
Capt. T. T. Craven, United States Navy.
William F. Durand, Ph. D.
John F. Hayford, C. E.
Charles F. Marvin, M. E.
Maj. Gen. Charles T. Menoher, United States Army.
Michael I. Pupin, Ph. D.
Rear Admiral D. W. Taylor, United States Navy.
Orville Wright, B. S.

THE EXECUTIVE COMMITTEE.

For carrying out the work of the Advisory Committee the regulations provide for the election annually of an executive committee, to consist of seven members, and to include further any member of the Advisory Committee not otherwise a member of the executive committee, but resident in or near Washington and giving his time wholly or chiefly to the special work of the committee. The executive committee, as elected and organized on October 7, 1920, is as follows:

Joseph S. Ames, Ph. D., chairman.
S. W. Stratton, Sc. D., secretary.
Maj. Thurman H. Bane, United States Army.
Capt. T. T. Craven, United States Navy.
John F. Hayford, C. E.
Charles F. Marvin, M. E.
Maj. Gen. Charles T. Menoher, United States Army.
Rear Admiral D. W. Taylor, United States Navy.
Charles D. Walcott, Sc. D.

The executive committee, in accordance with the general instructions of the Advisory Committee, exercises the functions prescribed by law for the whole committee, administers the affairs of the committee, and exercises general supervision over all its activities. The executive committee held regular monthly meetings throughout the year, and in addition held three special meetings, on the following dates: October 9, 1919; March 1 and June 28, 1920.

The executive committee has organized the necessary clerical and technical staffs for handling the work of the committee proper. General responsibility for the execution of the programs and policies approved by the executive committee is vested in the executive officer, Mr. George W. Lewis. In the subdivision of general duties, he has immediate charge of the scientific and technical work of the committee, being directly responsible to the chairman of the executive committee, Dr. Joseph S. Ames. The assistant secretary, Mr. John F. Victory, has charge of administration and personnel matters, property, and disbursements, under the direct control of the secretary of the committee, Dr. S. W. Stratton.

SUBCOMMITTEES.

The executive committee has organized six standing subcommittees, divided into two classes, administrative and technical, as follows:

ADMINISTRATIVE.	TECHNICAL.
Personnel, buildings, and equipment.	Aerodynamics.
Publications and intelligence.	Power plants for aircraft.
Governmental relations.	Materials for aircraft.

The organization and work of the technical subcommittees are covered in the reports of those committees appearing in another part of this report. A statement of the organization and functions of the administrative subcommittees follows:

COMMITTEE ON PERSONNEL, BUILDINGS, AND EQUIPMENT.

FUNCTIONS.

1. To handle all matters relating to personnel, including the employment, promotion, discharge, and duties of all employees.
2. To consider questions referred to it and make recommendations regarding the initiation of projects concerning the erection or alteration of laboratories and the equipment of laboratories and offices.
3. To meet from time to time on the call of the chairman, and report its actions and recommendations to the executive committee.
4. To supervise such construction and equipment work as may be authorized by the executive committee.

ORGANIZATION.

Dr. Joseph S. Ames, chairman.
Dr. S. D. Stratton, vice-chairman.
Prof. Charles F. Marvin.
J. F. Victory, secretary.

COMMITTEE ON PUBLICATIONS AND INTELLIGENCE.

FUNCTIONS.

1. The collection, classification, and diffusion of technical knowledge on the subject of aeronautics, including the results of research and experimental work done in all parts of the world.
2. The encouragement of the study of the subject of aeronautics in institutions of learning.
3. Supervision of the office of aeronautical intelligence.
4. Supervision of the foreign office in Paris.
5. The collection and preparation for publication of the technical reports, technical notes, and annual report of the committee.

ORGANIZATION.

Dr. Joseph S. Ames, chairman.
Prof. Charles F. Marvin, vice-chairman.
Miss M. M. Muller, secretary.

COMMITTEE ON GOVERNMENTAL RELATIONS.

FUNCTIONS.

1. Relations of the committee with executive departments and other branches of the Government.
2. Governmental relations with civil agencies.

ORGANIZATION.

Dr. Charles D. Walcott, chairman.
Dr. S. W. Stratton.
J. F. Victory, secretary.

QUARTERS FOR COMMITTEE.

On January 12, 1920, pursuant to authorization by the Public Buildings Commission, the administrative offices of the National Advisory Committee for Aeronautics were moved from the Air Service Building, Fourth Street and Missouri Avenue NW., Washington, D. C., to the Navy Building, Seventeenth and B Streets NW., Washington, D. C. The technical work of the committee, conducted by or under the supervision of the various subcommittees, has been carried on in various governmental laboratories and shops, including the Bureau of Standards and the committee's own field station at Langley Field, Va., known as the Langley Memorial Aeronautical Laboratory, and also in various laboratories connected with institutions of learning whose cooperation in the conduct of scientific research in aeronautics has been secured.

THE LANGLEY MEMORIAL AERONAUTICAL LABORATORY.

In previous annual reports, the committee described the progress made in the development of its field station at Langley Field, Va., for the prosecution of scientific research in aeronautics. The station now comprises three principal units, namely, an aerodynamical laboratory or wind tunnel, an engine dynamometer laboratory, and a research laboratory building, the latter including administrative and drafting offices, machine and wood-working shops, and photographic and instrument laboratories. The research laboratory and the wind tunnel building are of permanent brick construction; the engine dynamometer laboratory is housed in a temporary four-section steel airplane hangar.

With the completion of the wind tunnel proper, in April, 1920, the committee sought the approval of the President to name its field station in honor of the late Dr. Samuel Pierpont Langley. With the approval of the President and the Attorney General, the field station was accordingly given the name "Langley Memorial Aeronautical Laboratory," and was formally opened as such, with appropriate exercises, on June 11, 1920. Special invitations to men prominent in the development of aviation in the United States were issued jointly by the committee and the Director of Air Service of the Army. The executive committee, the aerodynamics committee, and the power plants committee held meetings at the field in the morning, and in the afternoon the members, the invited guests, of whom a number had flown to the field from Washington and more distant points, and the officers of the field assembled in the wind tunnel building, where formal dedicatory remarks were made by Dr. Joseph S. Ames, as Chairman of the Executive Committee, by Maj. Gen. Charles T. Menoher, as Director of Air Service of the Army and member of the committee, and by Rear Admiral D. W. Taylor, as Chief Constructor of the Navy and member of the committee.

The Langley Memorial Aeronautical Laboratory occupies a plot of ground known as Plot 16, Langley Field, Va., the plot having been set aside for the committee's use by the Chief Signal Officer of the Army at the time the site was selected as a proposed joint experimental station and proving ground for the Army and Navy air services and the Advisory Committee. The use of that plot of ground was officially approved by the Acting Secretary of War on April 24, 1919. The three buildings at present constituting the Langley Memorial Aeronautical Laboratory have been erected by the committee pursuant to authority granted by Congress.

OFFICE OF AERONAUTICAL INTELLIGENCE.

The Office of Aeronautical Intelligence was established in the early part of 1918 as an integral branch of the committee's activities. Its functions are the collection, classification, and diffusion of technical knowledge on the subject of aeronautics to the military and naval air services and civil agencies interested, including especially the results of research and experimental work conducted in all parts of the world. It is the officially designated Government depository for scientific and technical reports and data on aeronautics. The principal sources of such technical information are the following: The technical subcommittees and their engineering staffs, the Engineering Division and the Information Group of the Army Air Service, the Naval Air Service, the Bureau of Standards, the Forest Service, educational institutions, individual professors and experimenters, foreign governmental and private laboratories, and university professors.

Promptly upon receipt, all reports are analyzed and classified, and brought to the special attention of the subcommittees having cognizance, and to the attention of other interested parties, through the medium of public and confidential bulletins. Reports are duplicated where practicable, and distributed upon request. Confidential bulletins and reports are not circulated outside of governmental channels.

To efficiently handle the work of securing and exchanging reports in foreign countries, the committee maintains a technical assistant in Europe, with headquarters in Paris. It is his duty to personally visit the Government and private laboratories, centers of aeronautical information, and private individuals in England, France, Italy, Germany, and Austria, and endeavor to secure for America not only printed matter which would in the ordinary course of events become available in this country, but more especially to secure advance information as to work in progress, and any technical data not prepared in printed form, and which would otherwise not reach this country.

The service rendered by the Office of Aeronautical Intelligence during the past year has increased by approximately 60 per cent, including increases in services rendered to the Naval Air Service of 45 per cent; other governmental agencies, 60 per cent; aircraft manufacturers, 100 per cent; educational institutions, 180 per cent.

The technical assistant in Europe, in addition to rendering periodical reports as to developments in aeronautics, based on close personal observation of conditions in European countries, has secured for the committee many valuable reports and documents, including a number of complete sets of reports of the scientific research and experimental work conducted in Germany during the war.

THE AERONAUTICAL BOARD OF THE ARMY AND NAVY.

The Aeronautical Board was appointed by the Secretary of War and the Secretary of the Navy, and is composed exclusively of Army and Navy officers. It has no official connection with the National Advisory Committee for Aeronautics, its functions being the consideration of military questions regarding the use of aeronautics in both services. The committee feels that there is a positive need for such a joint board; in fact, the present Aeronautical Board is a development of the Joint Army and Navy Technical Aircraft Board which was established

during the war on the recommendation of this committee. There is no friction or duplication of functions whatsoever between the Aeronautical Board and this committee. On the contrary, a cordial contact has always existed where the work of the two organizations brought them together.

ORGANIZATION OF GOVERNMENTAL ACTIVITIES IN AERONAUTICS.

During the past year the committee has on numerous occasions given consideration to the subject of organization of governmental activities in aeronautics. A number of bills had been introduced in Congress providing widely differing solutions of the question, and each of these bills was discussed by the committee. After the adjournment of Congress and throughout the summer and fall of 1920 the committee endeavored to coordinate the views of the various governmental agencies interested, and to develop a tentative draft of legislation giving definite expression to the agreements reached. In its consideration of each of the measures introduced the committee was guided by an intimate knowledge of the problems peculiar to the military and naval air services, by the necessity of providing for, and reckoning with, a healthy development of civil aviation, and by broad general considerations of sound governmental policy in regard to matters of organization and administration. Of all the bills analyzed by the committee, two were selected for more earnest consideration, and in each case this has led to agreement upon amendments which will, in the committee's judgment, render either measure, if enacted into law, operative with a minimum of friction, confusion, or waste, at the same time utilizing existing agencies to the best interests of good administration. An analysis of the two bills referred to, as modified by the committee, follows:

House bill 14061 was introduced into the House of Representatives by Mr. Kahn, May 13, 1920. With the modifications recommended by the National Advisory Committee for Aeronautics, it provides for the establishment of a Bureau of Aeronautics in the Department of Commerce, in charge of a Commissioner of Air Navigation whose duties will comprise the licensing of aircraft, pilots, and airdromes, the designation of flying routes, cooperation with the States and municipalities in the laying out of landing fields, and, in general, the promotion of all matters looking to the advancement of commercial aviation. The bill provides also that all rules and regulations governing air navigation, licenses, etc., shall be formulated by the Commissioner of Air Navigation, who shall submit the same to the National Advisory Committee for Aeronautics for consideration, criticism, and recommendation to the Secretary of Commerce, who, if the same meet with his approval, shall formally promulgate the same. The bill provides further that the Commissioner of Air Navigation shall be appointed a member of the National Advisory Committee for Aeronautics, and shall seek the approval of the committee in certain matters, such as the laying out of flying routes, etc., which may hold a vital interest for other departments of the Government. The committee believes that all such extensive plans should be carefully considered with a view to serving the national interests as far as possible; that the Commissioner of Air Navigation should have the benefit of the counsel and advice of the other governmental agencies concerned; and that the method proposed in this bill would be practicable and effective.

House bill 14137 was introduced in the House of Representatives by Mr. Hicks, May 19, 1920. With the modifications recommended by the National Advisory Committee for Aeronautics, it makes substantially the same provisions for the regulation and development of air navigation as the modified bill, H. R. 14061, described above, with several additions, viz.: That the various departments of the Government shall prepare programs for experimental research or development work in aeronautics, and for the purchase or construction of aircraft, engines, accessories, and hangars, and the acquisition of land for purposes in connection with aviation, and shall submit such programs to the National Advisory Committee for Aeronautics for consideration and recommendation before contracts are made or orders are placed for same; that for the purpose of preliminary correlation of estimates the various departments shall submit their estimates for all aviation purposes to the National Advisory Committee

for Aeronautics for consideration and recommendation by the committee before the estimates are submitted to Congress, the comments and recommendations of the Advisory Committee to be transmitted to Congress along with the estimates.

The committee has been actuated in its suggested revision of these bills by a desire to produce practicable workable plans for improving the existing situation. The committee believes that the Hicks bill as modified is responsive to that sentiment in Congress which has sought to prevent duplication of expenditures and effort in the military and naval air services. The committee is not wholly convinced that the necessity for such legislation exists at the present time, nor that the method proposed would have the desired result. On the other hand, the committee is unanimous in supporting the Kahn bill as modified. The most urgent need at this time is the development of commercial aviation under Federal regulation. There has been some objection to placing the regulation of air navigation under the Department of Commerce, but the National Advisory Committee for Aeronautics believes it unnecessary and unwise to create another independent Government establishment for the exercise of such functions, and that by making the Commissioner of Air Navigation a member of the National Advisory Committee for Aeronautics, and requiring him to submit his plans to the committee, he can not fail to be guided in his actions by considerations of paramount national interests. The text of the two bills referred to, as modified by the National Advisory Committee for Aeronautics, follows:

H. R. 14061.

INTRODUCED IN THE HOUSE OF REPRESENTATIVES BY MR. KAHN, MAY 13, 1920.

A BILL To regulate air navigation within the United States and its dependencies, and between the United States or any of its dependencies and any foreign country or its dependencies.

Be it enacted by the Senate and House of Representatives of the United States of America in Congress assembled, That to provide for the regulation of air navigation and to render effective the provisions of any treaty or convention relating to air navigation that may hereafter be entered into by the United States, there is hereby established in the Department of Commerce a bureau to be known as the Bureau of Aeronautics, and a Commissioner of Air Navigation, who shall be the head thereof, who shall be appointed by the President, by and with the advice and consent of the Senate, and who shall receive a salary of $6,000 per annum. The Commissioner of Air Navigation shall be appointed by the President an additional member of the National Advisory Committee for Aeronautics.

SEC. 2. That there shall be in said bureau, an assistant commissioner of recognized technical ability, who shall be appointed by the President, by and with the advice and consent of the Senate and who shall receive a salary of $5,000 per annum. The assistant commissioner shall perform such duties as may be prescribed by the commissioner, or as may be required by law. There shall also be in said bureau a chief clerk, and such other clerical assistants, inspectors, experts, and special agents as may be required from time to time and authorized by Congress.

SEC. 3. That all rules and regulations hereinafter provided for, except as otherwise provided for in section 9 hereof, shall be formulated by the Commissioner of Air Navigation, who shall submit the same to the National Advisory Committee for Aeronautics for consideration, criticism, and recommendation to the Secretary of Commerce, who, if the same meet with his approval, shall formally promulgate the same. When approved and duly promulgated by the Secretary of Commerce such rules and regulations shall be legally binding and enforceable from the date of such promulgation, unless otherwise provided therein: *Provided*, That hereafter the National Advisory Committee for Aeronautics, in addition to the exercise of its present functions, is authorized to act in an advisory capacity in connection with the formulation and promulgation of such rules and regulations, and for the consideration of questions of policy affecting the development of civil or commercial aviation, including recommendations from time to time for amendments to this act or subsequent acts.

SEC. 4. That the Commissioner of Air Navigation shall, in accordance with section 3 hereof, formulate all necessary and proper rules and regulations respecting air navigation and air traffic, issuance of licenses for aircraft, aviators and aeronauts, rules of the air, the giving and heeding of signals, periodical and before-flight inspection of aircraft, the carrying of lights and signals, the landing at and departure from airdromes, using prescribed routes and avoiding prohibited areas, the carrying and lightening of ballast, the carryng and use of wireless telegraph and telephone instruments and other radio equipment, the carrying, keeping, and exhibiting of log books and other records, the landing for customs or immigration inspectors, and other matters for the safety and convenience of air navigation. Such rules and regulations shall prescribe air routes and prohibited areas over which aircraft shall not fly for military reasons or in the interest of public safety. Such rules and regulations shall include descriptions of, and, if necessary,

maps showing such air routes and prohibited areas, and shall include all areas over which the Secretary of War or the Secretary of the Navy may request in writing the prohibition of the movement of aircraft. All aircraft engaged in air navigation within the jurisdiction of the United States or its dependencies, or coming into such jurisdiction from a foreign country or its dependencies, or upon the high seas as to such aircraft as are flying under a United States license, and as to aircraft over which the United States has jurisdiction on other grounds, are hereby required to conform to the rules and regulations duly promulgated in accordance with this act.

SEC. 5. That all airdromes within the jurisdiction of the United States and its dependencies are hereby required to conform to such rules and regulations regarding the placing and use of lights and signals, the size and marking of landing places, and other matters for the safety of air navigation as may be prescribed by the rules and regulations duly promulgated in accordance with this act.

SEC. 6. That all rules and regulations, as herein provided, shall be so formulated as to carry out the provisions of this act and subsequent acts and of any treaty or convention which may hereafter be entered into by the United States. The Secretary of Commerce may alter, modify, amend, or revoke such rules and regulations in the same manner as provided in section 3 for the promulgation thereof, subject to the provisions of this act and any subsequent act and the provisions of any treaty which has been or may hereafter be entered into by the United States.

SEC. 7. That it shall be the province and duty of the Bureau of Aeronautics, except as may be otherwise provided, to foster, develop, and promote all matters pertaining to civil or commercial aeronautics, including the collection and dissemination of information relating thereto, the administration of all rules and regulations provided for in this act, the regulation and arrangement of landing fields and airdromes, and the allotment of such funds as may be provided by law to aid the various States in the establishment of landing fields and airdromes.

SEC. 8. That the Commissioner of Air Navigation is authorized and directed to plan aerial routes throughout the United States and its possessions, and to this end shall cooperate with the various States, cities, and municipalities for the purpose of setting aside and establishing airdromes and landing fields to be used in common by Federal, State, municipal, commercial, and private aircraft under the rules and regulations to be duly promulgated in accordance with section 3 of this act: *Provided*, That such plans for aerial routes and the establishment of airdromes and landing fields shall be submitted to the National Advisory Committee for Aeronautics, and upon approval by said advisory committee shall be carried into effect by the Commissioner of Air Navigation to the extent of the appropriations available for such purpose.

SEC. 9. That for the purpose of encouraging the development of commercial aeronautics in the United States, full cooperation shall be given by the Bureau of Aeronautics to the owners or operators of private or commercial aircraft, and that the Secretary of War, the Secretary of the Navy, the Postmaster General, and the Secretary of Commerce shall furnish to any owner or operator of private or commercial aircraft landing on an airdrome or landing field under their respective jurisdictions, aviation fuel, oil, supplies, and necessary mechanical assistance of an emergency character, under such regulations as they may approve and promulgate for their respective services. The proceeds from such sales and assistance shall be deposited in the Treasury of the United States to the credit of the appropriations involved.

SEC. 10. That no aircraft shall be used or operated in air navigation within the United States or its dependencies, or between the United States and any of its dependencies, or between the United States or any of its dependencies and any foreign country or its dependencies, or on the high seas as to aircraft over which the United States has jurisdiction, except under and in accordance with a license granted by the Commissioner of Air Navigation to the owner of the aircraft. Aircraft so licensed shall not be used or operated in air navigation except in accordance with the rules and regulations duly promulgated in accordance with this act: *Provided*, That aircraft and operators of the same duly registered and licensed in other countries and only transitorily or periodically in the United States and its dependencies may be exempted by treaty or convention from the requirements as to securing a license provided for in this section and section 11 hereof, but shall be subject to all other sections of this act and the rules and regulations duly promulgated in accordance with this act so long as they are within the boundaries of the United States and its dependencies.

Such license for aircraft shall not be granted unless the owner is a citizen of the United States or of its dependencies, or if such owner be a company or corporation, then a company or corporation organized under the laws of the United States or some one of the States thereof, the president or chairman and the majority of the members of which company, or the president or chairman and a majority of the board of directors and the holders of a majority of the stock of which corporation, are citizens of the United States or of its dependencies, nor unless such aircraft shall be constructed in a manner suitable for the service in which it is to be employed and is in a condition to warrant the belief that it may be used for such service in air navigation with reasonable safety, such construction, including standards of both workmanship and material, and condition to be determined in accordance with the rules and regulations duly promulgated in accordance with this act.

The Commissioner of Air Navigation shall keep a record in which licensed aircraft shall be registered, and such record shall contain, in a statement made under oath, the name of the owner of the aircraft, a state-

ment as to his citizenship, or, in the case of a company or corporation, the facts showing that it comes within the provisions of this section, the purpose for which the aircraft is to be used, and an accurate description of such aircraft.

Such license shall expire one year from the date of its issuance or upon a change of ownership of the aircraft, whichever may first occur, and shall not be renewed or extended, but upon expiration thereof in either of the manners mentioned the owner of the aircraft may apply for a new license upon complying with the laws and the rules and regulations duly promulgated in accordance with this act.

Any such license may be revoked at any time by the Commissioner of Air Navigation upon its being shown to his satisfaction that any of the facts and qualifications upon which the issuance of such license was based have ceased to exist or upon failure to comply with the existing laws and rules and regulations.

For the purpose of ascertaining the facts upon which to determine whether licenses shall be granted or revoked the Commissioner of Air Navigation shall have the right to conduct hearings, to summon witnesses, to administer oaths, and to inspect books and records, including the stock books of companies and corporations.

Sec. 11. That no person shall operate an aircraft engaged in air navigation as provided in section 10 of this act, except under and in accordance with a license granted by the Commissioner of Air Navigation, and the Commissioner of Air Navigation is authorized to grant such license in accordance with rules and regulations duly promulgated in accordance with this act.

Such license shall expire one year from date of its issuance unless sooner revoked by the Commissioner of Air Navigation upon its being shown to his satisfaction that any of the facts and qualifications upon which the issuance of such license was based have ceased to exist or upon failure to comply with the existing laws and rules and regulations, but upon the expiration of such license the holder thereof may apply for a new license by complying with the laws and rules and regulations duly promulgated in accordance with this act.

Sec. 12. No airdrome shall be operated except under and in accordance with a license granted by the Commissioner of Air Navigation to the owner of the airdrome in accordance with the rules and regulations duly promulgated in accordance with this act: *Provided*, That such owner shall be a citizen of the United States or of its dependencies, or, if such owner be a company or corporation, then a company or corporation organized under the laws of the United States or some one of the States thereof, the president or chairman and the majority of the members of which company, or the president or chairman of the board of directors and a majority of the board of directors and the holders of a majority of the stock of which corporation, are citizens of the United States or its dependencies, and upon its being shown that the airdrome is prepared to operate in accordance with the said rules and regulations. Such license shall expire upon a change of ownership of the airdrome and shall not be renewed or extended, but a new license may be issued upon compliance with the laws and rules and regulations duly promulgated in accordance with this act. Any such license may be revoked at any time by the Commissioner of Air Navigation upon its being shown to his satisfaction that any of the facts upon which the issuance of such license was based have ceased to exist, or upon failure to comply with existing laws and rules and regulations duly promulgated in accordance with this act.

Sec. 13. That the Commissioner of Air Navigation is authorized, subject to the approval of the Secretary of Commerce, to fix the fees and charges for the licenses which are authorized by this Act, which fees and charges shall be collected by the Commissioner of Air Navigation and covered into the Treasury of the United States to the credit of miscellaneous receipts.

Sec. 14. That any person, partnership, joint-stock company, association, or corporation operating aircraft or an airdrome over which the United States may have jurisdiction on any grounds, who shall violate any of the provisions of this act or of the rules and regulations duly promulgated in accordance with this act, or who shall aid or abet in such violation, or who shall obstruct or impede compliance with or the enforcement of the provisions of this act or of any such rules and regulations shall, upon conviction thereof, be fined not more than $1,000 or be imprisoned for not more than one year, or both, in the discretion of the court. In the event that such a violation shall be by a partnership, joint-stock company, association, or corporation, any officer, agent, or member thereof who is personally responsible for the violation shall be subject to the punishment herein prescribed. The Commissioner of Air Navigation may also, in case of a conviction, in his discretion, revoke or suspend for such length of time as he may deem proper any license issued by him to the owner or operator, or both, of the aircraft or airdrome involved in any such violation.

The jurisdiction of the Federal courts of offenses against the provisions of this act, or the rules and regulations made pursuant thereto, and the venue for the trial of the same shall be as prescribed by existing law for offenses triable before the Federal courts.

Sec. 15. That the provisions of this act authorizing the regulation of air navigation and airdromes, and the rules and regulations made pursuant thereto, and the provisions of this act relating to licensing of aircraft and airdromes and the operators of aircraft, and the rules and regulations made pursuant thereto, shall not apply to aircraft nor airdromes owned by the Government of the United States nor to the operators employed by any department or other governmental agency to operate or assist in the operation of aircraft owned by the Government of the United States, nor to aircraft built for the purpose of experiment and flown for the purpose of experiment or test within three miles of the airdrome or aircraft factory, nor to operators of aircraft within the precincts of an airdrome as defined in the said rules and regulations when such persons

are under the instruction of a person duly licensed in accordance with the provisions of section 11 of this act, except that all aircraft and operators, Government or otherwise, are required to observe the rules and regulations for lights, signals, and rules of the air.

SEC. 16. That such portions of the air as are navigable by aircraft and all aircraft navigating the air are hereby declared to be within the admiralty jurisdiction of the Federal courts; and the district courts of the United States shall have jurisdiction of all cases involving air navigation and aircraft, with the right of appeal as in other cases, in accordance with existing laws or such laws as may be hereafter enacted, saving to suitors in all cases the right of a common-law remedy where the common law is competent to give it. The maritime law and all existing acts and acts hereafter enacted relating to water craft and water navigation shall be held to govern aircraft and air navigation in so far as applicable thereto and except as modified by this act and subsequent acts, and by the rules and regulations made pursuant thereto, and by the treaties or conventions that may hereafter be entered into by the United States, and the rules and regulations made pursuant thereto.

SEC. 17. That the Commissioner of Air Navigation shall submit estimates for appropriations through the Secretary of Commerce. He shall be charged with the duty of examining all money accounts covering disbursements of funds appropriated for the Bureau of Aeronautics, with the examination of all property accounts covering all aeronautical property in the custody of the Bureau of Aeronautics, and shall have power to prescribe, institute, and enforce such system of money and property accountability as in his judgment will best safeguard the interests of the United States.

SEC. 18. That for the purposes of this act, not chargeable to existing appropriations, including personal services in the field and in the District of Columbia, there is hereby appropriated, out of any money in the Treasury not otherwise appropriated, to be available immediately, the sum of $100,000.

SEC. 19. That this act shall take effect from and after the date of its passage, and all acts or parts of acts contrary to the provisions of this act or inconsistent therewith be, and the same are hereby, repealed.

SEC. 20. That the Commissioner of Air Navigation shall annually, at the close of each fiscal year, make a report to the Secretary of Commerce, giving an account of all moneys received and disbursed by him and describing the work done by the bureau, and the Secretary of Commerce shall transmit the report to Congress with the annual report of the Department of Commerce.

SEC. 21. That if any section or provision of this act shall be held to be invalid, it is hereby provided that all other sections and provisions of this act which are not expressly held to be invalid shall continue in full force and effect.

H. R. 14137.

INTRODUCED IN HOUSE OF REPRESENTATIVES BY MR. HICKS, MAY 19, 1920.

A BILL To create a Bureau of Aeronautics in the Department of Commerce, and providing for the organization and administration thereof.

Be it enacted by the Senate and House of Representatives of the United States of America in Congress assembled, That to provide for the regulation of air navigation and to render effective the provisions of any treaty or convention relating to air navigation that may hereafter be entered into by the United States there is hereby established in the Department of Commerce a bureau to be known as the Bureau of Aeronautics, and a Commissioner of Air Navigation, who shall be the head thereof, who shall be appointed by the President, by and with the advice and consent of the Senate, and who shall receive a salary of $6,000 per annum. The Commissioner of Air Navigation shall be appointed by the President an additional member of the National Advisory Committee for Aeronautics.

SEC. 2. That there shall be in said bureau an assistant commissioner of recognized technical ability, who shall be appointed by the President, by and with the advice and consent of the Senate, and who shall receive a salary of $5,000 per annum. The assistant commissioner shall perform such duties as may be prescribed by the commissioner or as may be required by law. There shall also be in said bureau a chief clerk and such other clerical assistants, inspectors, experts, and special agents as may be required from time to time and authorized by Congress.

SEC. 3. That all rules and regulations herein provided for, except as otherwise provided for in section 12 hereof, shall be formulated by the Commissioner of Air Navigation, who shall submit the same to the National Advisory Committee for Aeronautics for consideration, criticism, and recommendation to the Secretary of Commerce, who, if the same meet with his approval, shall formally promulgate the same; when approved and duly promulgated by the Secretary of Commerce, such rules and regulations shall be legally binding and enforceable from the date of such promulgation unless otherwise provided therein: *Provided,* That hereafter the National Advisory Committee for Aeronautics, in addition to the exercise of its present functions, is authorized to act in an advisory capacity in connection with the formulation and promulgation of such rules and regulations, for the consideration of questions of policy affecting the development of civil or commercial aviation, including recommendations from time to time for amendments to this act or subsequent acts, and for the coordination of the aeronautical activities of the various departments of the Government.

The said National Advisory Committee for Aeronautics shall have authority to consider and recommend to the heads of departments concerned, on questions of policy regarding the development of civil aviation, with particular reference to education, preliminary training, commercial production of aircraft, establishment, elimination, and consolidation of all flying fields and air stations, and all other matters in connection therewith.

SEC. 4. That hereafter the War, Navy, and other departments of the Government shall prepare programs for experimental research and development work in aeronautics, and for the purchase or construction of air craft, engines, accessories, and hangars, and the acquisition of land for purposes in connection with aviation, and shall submit same to the said advisory committee for consideration and recommendation before contracts are made or orders are placed for the purchase, manufacture, or construction of the same.

SEC. 5. That the National Advisory Committee for Aeronautics shall have authority to recommend to the heads of the departments concerned the tarnsfer of aircraft and aircraft equipment and accessories from one department to another for the civil uses of the Government. The heads of the various departments concerned are authorized to make such transfers of aircraft, equipment, and accessories when recommended by the said advisory committee.

SEC. 6. That the said advisory committee shall consider and report upon any question dealing with aviation referred to it by the President or by any of the departments, and shall initiate, report, and recommend to departmental heads desirable undertakings or developments in the field of aviation, and each department shall furnish the said advisory committee such information as to its aviation activities as may be requested.

SEC. 7. That the Commissioner of Air Navigation shall, in accordance with section 3 hereof, formulate all necessary and proper rules and regulations respecting air navigation and air traffic, issuance of licenses for aircraft, aviators, and aeronauts, rules of the air, the giving and heeding of signals, periodical and before-flight inspection of aircraft, the carrying of lights and signals, the landing at and departure from airdromes, using prescribed routes and avoiding prohibited areas, the carrying and lightening of ballast, the carrying and use of wireless telegraph and telephone instruments and other radio equipment, the carrying, keeping, and exhibiting of log books and other records, the landing for customs or immigration inspections, and other matters for the safety and convenience of air navigation.

Such rules and regulations shall prescribe air routes and prohibited areas over which aircraft shall not fly, for military reasons or in the interest of public safety. Such rules and regulations shall include descriptions of, and, if necessary, maps showing such air routes and prohibited areas, and shall include all areas over which the Secretary of War or the Secretary of the Navy may request in writing the prohibition of the movement of aircraft. All aircraft engaged in air navigation within the jurisdiction of the United States or its dependencies or coming into such jurisdiction from a foreign country or its dependencies, or upon the high seas as to such aircraft as are flying under a United States license, and as to aircraft over which the United States has jurisdiction on other grounds, are hereby required to conform to the rules and regulations duly promulgated in accordance with this act.

SEC. 8. That all airdromes within the jurisdiction of the United States and its dependencies are hereby required to conform to such rules and regulations regarding the placing and use of lights and signals, the size and marking of landing places, and other matters for the safety of air navigation as may be prescribed by the rules and regulations duly promulgated in accordance with this act.

SEC. 9. That all rules and regulations as herein provided shall be so formulated as to carry out the provisions of this act and subsequent acts and of any treaty or convention which may hereafter be entered into by the United States. The Secretary of Commerce may alter, modify, amend, or revoke such rules and regulations in the same manner as provided in section 3 for the promulgation thereof, subject to the provisions of this act and any subsequent act and the provisions of any treaty which has been or may hereafter be entered into by the United States.

SEC. 10. That it shall be the province and duty of the Bureau of Aeronautics, except as may be otherwise provided, to foster, develop, and promote all matters pertaining to civil or commercial aeronautics, including the collection and dissemination of information relating thereto; the administration of all rules and regulations provided for in this act; the regulation and arrangement of landing fields and airdromes; and the allotment of such funds as may be provided by law to aid the various States in the establishment of landing fields and airdromes.

SEC. 11. That the Commissioner of Air Navigation is authorized and directed to plan aerial routes throughout the United States and its possessions and to this end shall cooperate with the various States, cities, and municipalities for the purpose of setting aside and establishing airdromes and landing fields to be used in common by Federal, State, municipal, commercial, and private aircraft under the rules and regulations to be duly promulgated in accordance with section 3 of this act: *Provided*, That such plans for aerial routes and the establishment of airdromes and landing fields shall be submitted to the National Advisory Committee for Aeronautics and, upon approval by said advisory committee, shall be carried into effect by the Commissioner of Air Navigation to the extent of the appropriations available for such purpose.

SEC. 12. That for the purpose of encouraging the development of commercial aeronautics in the United States full cooperation shall be given by the Bureau of Aeronautics to the owners or operators of private or commercial aircraft, and that the Secretary of War, the Secretary of the Navy, the Postmaster General, and the Secretary of Commerce shall furnish to any owner or operator of private or commercial aircraft landing on an airdrome or landing field under their respective jurisdictions aviation fuel, oil, supplies, and necessary mechanical assistance of an emergency character under such regulations as they may approve and promulgate for their respective services; the proceeds from such sales and assistance shall be deposited in the Treasury of the United States to the credit of the appropriations involved.

SEC. 13. That no aircraft shall be used or operated in air navigation within the United States or its dependencies, or between the United States and any of its dependencies, or between the United States or any of its dependencies and any foreign country or its dependencies, or on the high seas as to aircraft over which the United States has jurisdiction, except under and in accordance with a license granted by the Commissioner of Air Navigation to the owner of the aircraft. Aircraft so licensed shall not be used or operated in air navigation except in accordance with the rules and regulations duly promulgated in accordance with this act: *Provided*, That aircraft and operators of the same duly registered and licensed in other countries and only transitorily or periodically in the United States and its dependencies, may be exempted by treaty or convention from the requirements as to securing a license provided for in this section and section 14 hereof, but shall be subject to all other sections of this act and the rules and regulations duly promulgated in accordance with this act so long as they are within the boundaries of the United States and its dependencies. Such license for aircraft shall not be granted unless the owner is a citizen of the United States or of its dependencies, or if such owner be a company or corporation, then a company or corporation organized under the laws of the United States or some one of the States thereof, the president or chairman and the majority of the members of which company, or the president or chairman and a majority of the board of directors and the holders of a majority of the stock of which corporation, are citizens of the United States or of its dependencies, nor unless such aircraft shall be constructed in a manner suitable for the service in which it is to be employed and is in a condition to warrant the belief that it may be used for such service in air navigation with reasonable safety, such construction, including standards of both workmanship and material, and condition to be determined in accordance with the rules and regulations duly promulgated in accordance with this act.

The Commissioner of Air Navigation shall keep a record in which licensed aircraft shall be registered, and such record shall contain, in a statement made under oath, the name of the owner of the aircraft, a statement as to his citizenship, or, in the case of a company or corporation, the facts showing that it comes within the provisions of this section, the purpose for which the aircraft is to be used, and an accurate description of such aircraft.

Such license shall expire one year from the date of its issuance or upon a change of ownership of the aircraft, whichever may first occur, and shall not be renewed or extended, but upon expiration thereof in either of the manners mentioned the owner of the aircraft may apply for a new license upon complying with the laws and the rules and regulations duly promulgated in accordance with this act.

Any such license may be revoked at any time by the Commissioner of Air Navigation upon its being shown to his satisfaction that any of the facts and qualifications upon which the issuance of such license was based have ceased to exist or upon failure to comply with the existing laws and rules and regulations.

For the purpose of ascertaining the facts upon which to determine whether licenses shall be granted or revoked, the Commissioner of Air Navigation shall have the right to conduct hearings, to summon witnesses, to administer oaths, and to inspect books and records, including the stock books of companies and corporations.

SEC. 14. That no person shall operate an aircraft engaged in air navigation as provided in section 13 of this act except under and in accordance with a license granted by the Commissioner of Air Navigation, and the Commissioner of Air Navigation is authorized to grant such license in accordance with the rules and regulations duly promulgated in accordance with this act.

Such license shall expire one year from date of its issuance unless sooner revoked by the Commissioner of Air Navigation upon its being shown to his satisfaction that any of the facts and qualifications upon which the issuance of such license was based have ceased to exist, or upon failure to comply with the existing laws and rules and regulations, but upon the expiration of such license the holder thereof may apply for a new license by complying with the laws and rules and regulations duly promulgated in accordance with this act.

SEC. 15. That no airdrome shall be operated except under and in accordance with a license granted by the Commissioner of Air Navigation to the owner of the airdrome in accordance with the rules and regulations duly promulgated in accordance with this act: *Provided*, That such owner shall be a citizen of the United States or of its dependencies; or if such owner be a company or corporation, then a company or corporation organized under the laws of the United States, or some one of the States thereof; the president or chairman and the majority of the members of which company, or the president or chairman of the board of directors and a majority of the board of directors and the holders of a majority of the stock of which corporation, are citizens of the United States or its dependencies; and upon its being shown that the airdrome is prepared to operate in accordance with the said rules and regulations. Such license shall expire upon a change of ownership of the airdrome and shall not be renewed or extended, but a new license may be issued upon

compliance with the laws and rules and regulations duly promulgated in accordance with this act. Any such license may be revoked at any time by the Commissioner of Air Navigation upon its being shown to his satisfaction that any of the facts upon which the issuance of such license was based have ceased to exist or upon failure to comply with existing laws and rules and regulations duly promulgated in accordance with this act.

SEC. 16. That the Commissioner of Air Navigation is authorized, subject to the approval of the Secretary of Commerce, to fix the fees and charges for the licenses which are authorized by this act, which fees and charges shall be collected by the Commissioner of Air Navigation and covered into the Treasury of the United States to the credit of miscellaneous receipts.

SEC. 17. That any person, partnership, joint-stock company, association, or corporation operating aircraft or an airdrome over which the United States may have jurisdiction on any grounds, who shall violate any of the provisions of this act or of the rules and regulations duly promulgated in accordance with this act, or who shall aid or abet in such violation, or who shall obstruct or impede compliance with or the enforcement of the provisions of this act, or of any such rules and regulations, shall, upon conviction thereof, be fined not more than $1,000 or be imprisoned for not more than one year, or both, in the discretion of the court. In the event that such a violation shall be by a partnership, joint-stock company, association, or corporation, any officer, agent, or member thereof who is personally responsible for the violation shall be subject to the punishment herein prescribed. The Commissioner of Air Navigation may also, in case of a conviction, in his discretion, revoke or suspend for such length of time as he may deem proper any license issued by him to the owner or operator, or both, of the aircraft or airdrome involved in any such violation.

The jurisdiction of the Federal courts of offenses against the provisions of this act or the rules and regulations made pursuant thereto and the venue for the trial of the same shall be as prescribed by existing law for offenses triable before the Federal courts.

SEC. 18. That the provisions of this act authorizing the regulation of air navigation and airdromes, and the rules and regulations made pursuant thereto, and the provisions of this act relating to licensing of aircraft and airdromes and the operators of aircraft, and the rules and regulations made pursuant thereto, shall not apply to aircraft nor airdromes owned by the Government of the United States nor to the operators employed by any department or other governmental agency to operate or assist in the operation of aircraft owned by the Government of the United States nor to aircraft built for the purpose of experiment and flown for the purpose of experiment or test within three miles of the airdrome or aircraft factory nor to operators of aircraft within the precincts of an airdrome as defined in the said rules and regulations when such persons are under the instruction of a person duly licensed in accordance with the provisions of section 14 of this act, except that all aircraft and operators, Government or otherwise, are required to observe the rules and regulations for lights, signals, and rules of the air.

SEC. 19. That such portions of the air as are navigable by aircraft and all aircraft navigating the air are hereby declared to be within the admiralty jurisdiction of the Federal courts; and the district courts of the United States shall have jurisdiction of all cases involving air navigation and aircraft, with the right of appeal as in other cases, in accordance with existing laws or such laws as may be hereafter enacted, saving to suitors in all cases the right of a common-law remedy where the common law is competent to give it. The maritime law and all existing acts and acts hereafter enacted relating to water craft and water navigation shall be held to govern aircraft and air navigation in so far as applicable thereto, and except as modified by this act and subsequent acts, and by the rules and regulations made pursuant thereto, and by the treaties or conventions that may hereafter be entered into by the United States, and the rules and regulations made pursuant thereto.

SEC. 20. That all estimates of funds necessary to meet the requirements of all services and departments of the Government for the development, production, operation, and maintenance of aircraft, aircraft material and accessories, including fields, shops, airdromes, and all other facilities connected therewith, shall be submitted to the National Advisory Committee for Aeronautics for consideration and recommendation before they are submitted to Congress, and any recommendations or suggestions made by the said advisory committee shall be transmitted to the Congress with the estimates by not later than the 15th day of October of each year.

SEC. 21. That the Commissioner of Air Navigation shall submit estimates for appropriations through the Secretary of Commerce after compliance with the provisions of section 20 of this act. He shall be charged with the duty of examining all money accounts covering disbursements of funds appropriated for the Bureau of Aeronautics, with the examination of all property accounts covering all aeronautical property in the custody of the Bureau of Aeronautics, and shall have power to prescribe, institute, and enforce such system of money and property accountability as in his judgment will best safeguard the interests of the United States.

SEC. 22. That for the purposes of this act, not chargeable to existing appropriations, including personal service in the field and in the District of Columbia, there is hereby appropriated, out of any money in the Treasury not otherwise appropriated, to be available immediately, the sum of $100,000.

SEC. 23. That this act shall take effect from and after the date of its passage, and all acts or parts of acts contrary to the provisions of this act or inconsistent therewith be, and the same are hereby, repealed.

SEC. 24. That the Commissioner of Air Navigation shall annually, at the close of each fiscal year, make a report to the Secretary of Commerce, giving an account of all moneys received and disbursed by him and describing the work done by the bureau, and the Secretary of Commerce shall transmit the report to Congress with the annual report of the Department of Commerce.

SEC. 25. That if any section or provision of this act shall be held to be invalid, it is hereby provided that all other sections and provisions of this act which are not expressly held to be invalid shall continue in full force and effect.

CANADA'S COURTESY REGARDING AMERICAN AIR PILOTS.

In June, 1920, the committee received through the State Department information to the effect that the Canadian Air Board had promulgated regulations permitting United States qualified aircraft and pilots to fly in Canada until November 1, 1920, on the same basis as if the United States had established air regulations as contemplated under the Convention for the Regulation of International Air Navigation. The committee, by resolution adopted at the July meeting, recommended that the State Department express the appreciation of the Government of the United States for the courtesy of the Canadian Government in this matter, and, in view of the fact that the Congress of the United States was not then in session and would not meet until December, 1920, further recommended that the State Department inquire if the Canadian Government would be willing to extend by six months from November 1, 1920, the period during which United States pilots and aircraft would be permitted to fly in Canada under the existing conditions. The State Department acted upon these recommendations, and as a result the Canadian Government has extended its courtesies to American pilots and aircraft until June 1, 1921. The entire incident, however, serves to emphasize the need for Federal legislation for the regulation of air navigation, as recommended in another part of this report.

INTERNATIONAL CONVENTION ON AIR NAVIGATION.

During the past year the committee has given consideration to a number of questions dealing with the subject of international air navigation referred to it by the State Department, and in each case has submitted its recommendations to the State Department. The questions considered by the committee have been mainly those arising under the pending International Convention on Air Navigation and a few miscellaneous questions in regard to the general subject of international air navigation.

In regard to the Convention on International Air Navigation, the committee formulated and recommended to the State Department reservations which were accepted by the department and communicated as instructions to the American ambassador to France for his official notation at the time of signing the convention preliminary to ratification by the Government of the United States. The committee has also considered reservations formulated by Canada and submitted recommendations to the State Department as to concurrence and nonconcurrence therewith on the part of the United States.

CIVIL USE OF GOVERNMENT LANDING FIELDS.

In June, 1920, the Aeronautical Board sought the advice of the committee as to the governmental policy to be followed in regard to the question of permitting the use of Government landing fields and facilities by private or commercial aircraft. The meeting at which this subject was discussed was held at Langley Field, Va., in connection with the formal opening of the Langley Memorial Aeronautical Laboratory, at which many persons prominent in aeronautical development were present, a number of whom had flown to the field from Washington and more distant places.

The committee at that meeting adopted a resolution stating that, "It is the sense of the National Advisory Committee for Aeronautics that each governmental agency having airdromes or landing fields under its jurisdiction should be authorized by law to furnish to owners or operators of private or commercial aircraft, landing on or near such airdromes or landing fields, aviation fuel, oil, supplies, and necessary mechanical assistance at cost plus 10 per cent,

under such regulations and restrictions as may be approved from time to time by the heads of the departments concerned;" and that "Private or commercial aircraft should not be allowed to use Government airdromes or landing fields as home stations, and that only mechanical assistance or repairs of an emergency nature should be furnished at Government airdromes or landing fields, such as are necessary to permit an aircraft to resume its journey."

These principles have been embodied in a draft of proposed legislation for the regulation of air navigation, referred to in another part of this report.

PROTECTION OF AIRCRAFT INDUSTRY FROM UNFAIR FOREIGN COMPETITION.

A bill was introduced in the House of Representatives during the last session by Representative Tilson, known as the "anti-dumping bill" (H. R. 14287), the principal object of which was the prevention of unfair foreign competition in the sale of airplanes imported into the United States. At the time there was some question as to the facts in the matter, the merits of the bill, and the need for such legislation. After the adjournment of Congress, the bill not having been passed, the committee recommended to the Secretary of Commerce that the Bureau of Foreign and Domestic Commerce be authorized to make a thorough investigation with a view to determining all the facts bearing upon the advisability of the proposed legislation, in order that the Secretary of Commerce might be able to present to Congress at its next session the facts as determined by the investigation, together with his recommendations. This the Secretary of Commerce, under date of June 21, 1920, agreed to do.

DEVELOPMENT OF RIGID AIRSHIPS.

The National Advisory Committee for Aeronautics at the semiannual meeting of the full committee had under consideration the question of the development of rigid airships, which the committee considers essential for our national defense. The Army and Navy had agreed that until standard types were developed in this country, the work of development should rest with the Navy. The proper development of this type of aircraft for military purposes will unquestionably lead to the development of commercial types, but it is felt that the Government must take the lead by first developing rigid airships for military purposes.

The committee at that time submitted a special report to the President recommending that adequate provision be made in the then pending naval appropriation bill for the construction of rigid airships and suitable hangars, and that a continuing building program for this type of aircraft be authorized, extending over a period of years. The committee at this time reiterates this recommendation, and expresses its belief that this experimental development is of vital importance to the effectiveness of the Army and Navy in time of war, and particularly to the military and naval air services as combatant arms.

PRODUCTION OF HELIUM.

At the semiannual meeting of the full committee, held in April, 1920, consideration was given to the question of the production of helium. Helium has such advantages over any other known gas as to make its use imperative for military and naval airships in time of war, provided it can be made available in sufficient quantity. In letters to the Secretary of War, the Secretary of the Navy, and the Secretary of the Interior, the committee stated that it is necessary to encourage the economical production of helium in order that an increased demand may bring about a greater increase in the supply, a simplification of the processes of extraction, and a lessening of the cost of production. The committee especially invited their attention to the necessity for thoroughly investigating all sources from which helium may be extracted or secured, and recommended that every practicable effort be made both to increase production and to decrease cost, having due regard for conservation of the sources of supply for military purposes.

EDUCATION IN ADVANCED AERONAUTICAL ENGINEERING.

At the semiannual meeting of the full committee in April, 1920, consideration was given to the question of education in advanced aeronautical engineering. This meeting was attended by all the members of the committee connected with universities: Drs. Ames, Durand, Hayford, and Pupin, and it is deemed worthy of special notice that each of these members individually expressed his approval of the resolution which was adopted at that meeting in the following terms:

Whereas it is deemed essential to the development of aviation in America for military and naval purposes that advanced instruction in aeronautical engineering be given to military and naval officers at a competent educational institution; and

Whereas the public demand for such instruction will in all probability not be sufficient to justify or permit the offering of such advanced courses in more than one institution at the present time; and

Whereas such an advanced course is now being given at the Massachusetts Institute of Technology; and

Whereas it is deemed further essential that actual experience with aerodynamic research should form a part of such advanced instruction: Therefore be it

Resolved, That the National Advisory Committee for Aeronautics hereby recommends to the Secretary of War and to the Secretary of the Navy the adoption of a continuing policy for the instruction of officers in advanced aeronautical engineering, and that for the next three years classes of 15 Army officers and 15 Navy officers be detailed annually to take such instruction in advanced aeronautical engineering at the Massachusetts Institute of Technology at the expense of the War and Navy Departments, respectively.

Resolved further, That, in connection with the course in advanced aeronautical engineering, the National Advisory Committee for Aeronautics cooperate in every way with the Massachusetts Institute of Technology by offering to its faculty and students the facilities for investigations in aerodynamics and experimental work on actual airplanes at the committee's research laboratory, Langley Field, Va.

Resolved further, That the National Advisory Committee for Aeronautics offer to give at various engineering universities courses of lectures in advanced aeronautical engineering by members of its engineering staff.

Resolved further, That the National Advisory Committee for Aeronautics recommend that educational institutions generally not consider the establishment of courses in aeronautical engineering at the present time, as it is the opinion of the committee that the demand for such instruction outside of the Government service is not sufficient, and competent instructors for such courses are not available.

This resolution was transmitted to the Secretary of War and to the Secretary of the Navy. The War Department, acting on the committee's recommendation, secured the necessary authority from Congress to detail 25 officers for special instruction at the Massachusetts Institute of Technology. It is understood that the Navy has not secured similar authority. The committee therefore strongly recommends to Congress that similar authority be given for the detail of naval officers for such special training. At the present time both services are weak in respect to the number of officers sufficiently educated in aeronautical engineering. The committee considers that the diligent prosecution of a continuing program of education will be of great value within a few years in the development of military and naval aviation.

NOMENCLATURE FOR AERONAUTICS.

The National Advisory Committee for Aeronautics, to secure uniformity with reference to aeronautical terms in official documents of the Government, and, so far as possible, in technical and other commercial publications, has prepared a report on nomenclature for aeronautics, in classified and dictionary forms, and including a list of symbols used. This report was issued during the past year under the title "Report No. 91, Nomenclature for Aeronautics." It supersedes Report No. 25, on the same subject, which appeared in the Fourth Annual Report of the committee.

The subcommittee on aerodynamics had charge of the preparation of the nomenclature for aeronautics, and was materially assisted by the Interdepartmental Conference on Aeronautical Nomenclature and Symbols, which was especially organized by the executive committee, with the approval of the War and Navy Departments, for the purpose of giving proper representations to all technical divisions of the Army Air Service and the bureaus of the Navy Department.

The first meeting of the interdepartmental conference was held on October 23, 1919; the second meeting, on January 15, 1920, at which meeting the nomenclature was unanimously approved and recommended to the subcommittee on aerodynamics, with the reservation that stability terms and power plant terms be given further and special consideration.

The stability terms were accordingly referred for special consideration to Messrs. E. B. Wilson, J. C. Hunsaker, A. F. Zahm, E. P. Warner, and H. Bateman, and the power plant terms were referred to the subcommittee on power plants for aircraft. The complete report was adopted by the subcommittee on aerodynamics on March 8, 1920, and recommended to the executive committee for approval and publication.

Upon recommendation of the subcommittee on aerodynamics, the executive committee of the National Advisory Committee approved the nomenclature for publication as a technical report on April 1, 1920.

BIBLIOGRAPHY OF AERONAUTICS.

During the past year, the committee has continued the bibliography of aeronautics. The first work on this subject was prepared by Mr. Paul Brockett, of the Smithsonian Institution, and included the period up to 1910. The committee has prepared a bibliography of aeronautics from 1910 to 1916 in one volume, and a bibliography for the years 1917, 1918, and 1919 has been prepared in one volume. The bibliography for each year in the future will be prepared annually.

DISTRIBUTION OF METEOROLOGICAL INFORMATION BY WIRELESS.

The State Department, under date of June 5, 1920, referred to the committee for consideration and recommendation a copy of a note from the British ambassador, together with a copy of a report from an international commission which had been considering methods for the distribution of meteorological information by wireless telegraphy under the provisions of article 35 and Annexe G of the proposed Convention on International Air Navigation.

After consideration of this question at two meetings the committee reported to the Secretary of State that the proposal as submitted by the British ambassador appeared to be satisfactory in general, but that it would be impracticable to carry out the entire program in detail, the principal difficulty centering around the proposal to take observations at hours corresponding to 1, 7, 13, and 19 Greenwich mean time. At the present time observations are taken in the United States only at hours corresponding to 1 and 13 G. M. T. After taking up the matter with the Navy Department the committee, in its special report, also recommended that Annapolis be designated as the station for the dissemination of such reports for North America.

AEROLOGICAL WORK OF THE WEATHER BUREAU.

At the regular meeting of the executive committee of the National Advisory Committee for Aeronautics held on December 18, 1919, consideration was given to the increasing needs of aviation for improvements and extensions in the making of meteorological observations in the free air, and the issuance of forecasts and warnings for the promotion of safety of aerial navigation over the land and the oceans. Work of this character was at that time being conducted by the Weather Bureau under an appropriation of $100,000, which was originally granted by Congress in 1917 upon the recommendation of the National Advisory Committee for Aeronautics.

The organic act defining the duties and functions of the Weather Bureau clearly required it to perform this service. The making of local meteorological observations by the Army at certain military posts and by the Navy at base stations and aboard ships was necessary for local needs and obviated the maintenance by the Weather Bureau of stations at those points which would otherwise have been necessary.

The executive committee accordingly authorized the submission of a special report to the Committee on Agriculture of the House of Representatives in which the committee stated that there was no duplication of work or expenditure in these activities, the work of the Army and Navy in this connection being wholly supplementary and complementary to that of the Weather Bureau, their observations being telegraphed to the Weather Bureau daily for its use in conjunction with reports from over 200 stations of its own. In this special report the executive committee strongly recommended the appropriation of additional funds by Congress to extend this feature of the Weather Bureau's activities in order to meet the requirements of aviation and to safeguard lives and property engaged in aerial navigation.

REPORT OF THE COMMITTEE ON AERODYNAMICS.

Following is a statement of the organization and functions of the committee on aerodynamics:

ORGANIZATION.

Dr. John F. Hayford, Northwestern University, chairman.
Dr. Joseph S. Ames, Johns Hopkins University, vice chairman.
Maj. T. H. Bane, United States Army.
Dr. L. J. Briggs, Bureau of Standards.
Maj. V. E. Clark, United States Army.
Commander J. C. Hunsaker, United States Navy.
Franklin L. Hunt, Bureau of Standards.
Prof. Charles F. Marvin, Chief Weather Bureau.
Edward P. Warner, Massachusetts Institute of Technology, secretary.
Dr. A. F. Zahm, United States Navy.

FUNCTIONS.

1. To aid in determining the problems relating to the theoretical and experimental study of aerodynamics to be experimentally attacked by governmental and private agencies.
2. To endeavor to coordinate, by counsel and suggestion, the research and experimental work involved in the investigation of such problems.
3. To act as a medium for the interchange of information regarding aerodynamic investigations in progress or proposed.
4. The committee may direct and conduct research and experiment in aerodynamics in such laboratory or laboratories as may be placed (either in whole or in part) under its direction.
5. The committee shall meet from time to time on call of the chairman, and report its actions and recommendations to the executive committee.

The committee on aerodynamics, by reason of the representation of the Bureau of Standards, the Army, the Navy, technical institutions, and the industry, is in close contact with aerodynamical research and development work being carried on in the United States. Its representation enables it, by counsel and suggestion, to coordinate the experimental research work involved in the investigation of aerodynamical problems, and to influence the direction of the proper expenditure of energy toward those problems which seem of greatest importance.

The committee has direct control of aerodynamical research conducted at the Langley Memorial Aeronautical Laboratory and also directs propeller research conducted at Leland Stanford Junior University under the supervision of Dr. W. F. Durand, and through its membership it keeps in close touch with the work being carried on at the Bureau of Standards, at McCook Field by the engineering division of the Army Air Service, and at the Washington Navy Yard by the Bureau of Construction and Repair, United States Navy.

Two new wind tunnels have been completed and put in operation in the United States within the past year. A new 5-foot wind tunnel at the Langley Memorial Aeronautical Laboratory has gone into service and has already run at speeds slightly in excess of 110 miles per hour. It is anticipated that speeds of 140 miles per hour will be attained with a new

propeller which will be better suited to the characteristics of the electric motor employed. The other new wind tunnel of the year is that constructed by the Curtiss Engineering Corporation at Garden City and is of the true Eiffel type.

The committee on aerodynamics, in directing the research work at the Langley Memorial Aeronautical Laboratory, has adopted a definite policy with reference to research work to be conducted at this laboratory. The policy adopted confines the work to three general problems, and, in order to obtain results which will be of general use, experiments are to be conducted in such a manner that general conclusions and, if possible, general theories may result from them. The following three general problems covering the work of the aerodynamical laboratory for the coming year have been adopted:

(*a*) Comparison between the stability of airplanes, as determined from full-flight test and as determined from calculations based on wind tunnel measurements.

The committee will endeavor to determine the characteristics and peculiarities of certain existing airplanes, and attempt to account for these by calculations based on wind tunnel work. The matter of control will also fall under this heading. The first work conducted will probably be confined to the explanation of the theory of small oscillations and its verification with full-scale work. Later, a study of maneuverability and controllability will follow, as it is felt that in the present state of the art there is not available to airplane designers a rational method of predicting the maneuverability of airplanes from the drawings of the airplanes or from wind tunnel experiments with models.

(*b*) Similar comparison between the performance of airplanes full-scale and the calculations based on wind tunnel experiments.

A great deal of attention has been given by the British to the prediction of performance based on aerodynamic data, but there is still a gap between model and full-scale results which can not be bridged until we have more information. The performance is intimately connected with the propeller, and it is the intention of the committee to have all propeller research conducted at the Aerodynamical Laboratory of Leland Stanford Junior University under the direction of Dr. Durand. An effort will be made to tie in the results obtained at Leland Stanford with the performance work being done at Langley Field. Experiments will also be conducted on models of well-known airplanes to better understand the landing and starting characteristics of airplanes and to determine exactly what it is that makes certain airplanes require a long run.

(*c*) General aerofoil problem, including control surfaces, with particular reference to thick sections and combinations and modifications of such sections.

The committee is to undertake a systematic investigation of thick wing sections, after a thorough analysis of what has been done in this matter, and to duplicate some of the experiments already performed. After the determination of what properties of thick wing sections are of interest, work will then be carried along with a view to systematic variation of the variables which determine the aerodynamic properties of a series. Determination will also be made of the relation between aerodynamic properties of such standard aerofoils and aerofoils of similar profile but of different aspect ratio and taper. It is also desirable to know biplane and other interference effects when the aerofoils are used in combination. A careful study will also be made of recent work, by which it appears possible to predict from a knowledge of the lift coefficient the properties of aerofoils in combination and of different aspect ratio, as well as the influence of a boundary.

Such problems arising in connection with the Army and Navy programs of development as fit in logically with the above program will be referred to the committee on aerodynamics, and the research work covering the problems will be conducted at the Langley Memorial Aeronautical Laboratory.

At the Langley Memorial Aeronautical Laboratory a large number of experiments have been carried on with model wind tunnels in the past year to determine the best form for steadiness of flow and efficiency of operation. The effect of various shapes of cones, experimental chambers, and types of propellers, honeycombs, and diffusers were thoroughly studied. A special recording air-speed meter and recording yaw meter were designed in order to study the steadiness of flow, and it was found that the tunnel with a continuous throat was superior to the open or Eiffel type of tunnel both in efficiency and steadiness of flow. It was also demonstrated that a honeycomb placed in the entrance cone is of the greatest value in straightening the air flow, but a diffuser placed in the return circuit was apparently of little value.

The National Advisory Committee's 5-foot wind tunnel was completed in the spring of 1920 and has been in continuous operation since. This tunnel is designed from the data obtained in the model experiments and is very satisfactory both in efficiency and steadiness of flow. The 10-foot four-bladed propeller is driven by a 200-horsepower variable-speed electric motor. The power for this motor is obtained from gasoline-driven generating sets, and the control system is very convenient, the motor being started and stopped simply by pushing a button in the experimental chamber, and the speed being controlled by a rheostat from the same place.

The balance used in this tunnel is of the modified N. P. L. type, and was constructed in the shop of the National Advisory Committee at Langley Field. Unlike the usual balance, the weight is supported on a ball bearing socket, rather than a conical pivot, as this device considerably reduces the friction and will carry a much larger load. It is also possible with this balance to simultaneously read the lift, drag, and pitching moment. As the N. P. L. type of balance is not suited to holding tapered wings, and as a large amount of work of this kind is planned for the future, a simple wire type of balance is being constructed at the present time, similar to that used in the wind tunnel at Göttingen.

It has been the practice in the past when setting up a model to align the chord of the wing with the wind by placing a thin wooden batten on the wing and comparing this batten with a straight line on the floor of the tunnel. But as this method is rather laborious and inaccurate, a new type of aligning apparatus has been designed for this tunnel, consisting essentially of a mechanism for reflecting a beam of light from a plain mirror which is attached parallel to the chord of the wing, so that by rotating the wing the reflected beam of light is brought to a cross line on a small target on the side of the experimental chamber. In this way a wing can be lined up with an accuracy of 0.01° in a very few seconds. As the air speeds used in this wind tunnel are considerably higher than those usually encountered, a special type of manometer was constructed to obviate the necessity of having an extremely long inclined tube. This gauge changes the head of liquid and at the same time the inclination of the tube, so that the fluctuations of the liquid are approximately equal at any speed. A multiple manometer has also been constructed for pressure distribution work on models, containing 20 glass tubes, the inclination of which can be adjusted to any desired angle.

A thorough investigation has been made of the problem of spindle interference and the best manner of protecting the spindle by a fairwater. Different types and lengths of fairwater were tested in order to determine which condition would give the least total interference. An accurate determination of the effective resistance of the spindle was made for various lengths of spindle and for various air speeds so that a complete set of data is available for use on any model tests for the future. In order to provide data for stability calculations a wing was tested through an angle of 360°, and a model of an airplane was tested in the same way. In order to determine the scale corrections for model airplanes a model of the JN4H was constructed with great accuracy, and all details of the airplane were reproduced in the model, including the radiator and motor, but the wires were omitted as it was thought that their resistance could be determined better from tests of the full-sized wires. This model was tested at speeds of 30, 60, and 90 miles per hour in order to determine the corrections that must be applied to it in order to give the full-flight performance which was carefully determined on the full-sized machine.

FREE FLIGHT.

The machines available for the committee's use at the Langley Memorial Aeronautical Laboratory consist of two JN4H training machines and one DH4. During the summer the machines have been in the air about 60 hours. Numerous small changes have been made on these machines during the different tests, including changing the stagger, changing the angle of the tail plane, and changing the position of the center of gravity by adding weight at the front or rear of the fuselage. A large number of special instruments have been designed and constructed at Langley Field for research in full flight. An accelerometer has been developed for obtaining the loads on an airplane during stunts and landings, and satisfactory results have been obtained with it, which are of considerably greater accuracy than those obtained by other types of instruments. Instruments were also developed for recording the position of and the force on all three controls of the airplane, and valuable results have been obtained with these instruments. For obtaining the pressure distribution on the tail of the full-sized machine a special multiple manometer was constructed having 110 glass manometer tubes, all of which could be photographed at one time by an automatic film camera placed in the fuselage. As this instrument will only determine accurately the pressure distribution in steady flight, another manometer is now being constructed consisting of a large number of small diaphragm gauges which will record continuously on a moving film so that the rise and fall of the pressure at various points on the tail surfaces can be recorded during any stunt maneuver.

An air-speed meter and yaw meter have been constructed, working on the optical recording principle, having the actual period of the instrument high and its friction small, so that air-speed records can be obtained of any small or high period fluctuations in the wind velocity. To determine the angular rotation of the airplane during flight, in order to study its stability properties, a kymograph was constructed consisting of a narrow slit which focused the image of the sun on moving bromide paper, and another instrument of the same type has been constructed working on the gyroscopic principle. For obtaining the full-flight lift and drag coefficient a special longitudinal inclinometer was constructed which would give a large scale deflection and would be convenient and accurate to read.

The investigations undertaken consist of the determination of the lift and drag coefficients of the JN4H in free flight, and it is found possible by careful piloting to flay the machine at or slightly beyond the burble point. A thorough experimental investigation has been made of the static longitudinal stability of the airplane and a great many factors have been altered on the full-sized machine, such as changing the angle of the tail plane, changing the center of gravity of the machine, changing the section of the tail plane, and inclining the angle of the propeller axis. A study was made of the angle of attack and the air speed at the wing tips during spins and loops. This was accomplished by placing vanes and air-speed meters at the wing tips and photographing them during the maneuver by means of a camera gun and then plotting the curve of angle and speed against time from the photographs so obtained.

A very extensive investigation of the pressure distribution over the tail of an airplane in free flight has been undertaken. The pressure at 110 points on the left and right hand sides of the tail have been taken independently and the total pressure determined from these two curves. By means of photographic recording methods the time taken for making this investigation in the air is brief, but the computation and plotting of the results are laborious and require a long time for their completion. Runs were made with three positions of the center of gravity and two angles of setting of the stabilizer, as well as one run with celluloid over the crack between the stabilizer and the elevator. In all cases the pressure found over the tail was extremely low and in steady flight the load on the tail would be found very small compared with the load resulting from accelerated flight. A large number of records have been taken with the recording accelerometer designed by the N. A. C. A., these records being taken in the JN4H and several other machines during various stunts and landings. It was found that the maximum acceleration experienced in any stunt was during a roll, where the acceleration reached a maximum of 4.2 g. In order to determine the characteristics of an airplane during circling flight a record of the forces on all three controls was made doing banks of various angles up to 60° and side slips up to 20° of yaw.

The wind tunnel at Leland Stanford Junior University has again been occupied entirely with propeller tests. The results of the research work conducted this year are contained in technical report No. 109. Preparations are being made for tests on propellers at large angles of yaw, which will give data for the analysis of helicopters traveling horizontally.

Dr. George de Bothezat, aerodynamical expert of the National Advisory Committee for Aeronautics, has carried on at McCook Field, with the cooperation of the Engineering Division of the Army Air Service, a special investigation for the measurement of aerodynamic performance. The report on this investigation has been completed and approved as technical report No. 97, entitled "General Theory of the Steady Motion of an Airplane." This investigation involved the design and construction of a new type of barograph. Also in connection with his investigation of airplane performance, Dr. de Bothezat has designed a torque meter and a rate-of-climb meter, which are under construction. The torque meter is a very simple design, and present indications are that it will be a most serviceable and efficient instrument. The rate-of-climb meter is not based on a new principle; it is simply a new construction and design embodying the experience obtained in the use of other instruments.

The research work conducted by the Bureau of Construction and Repair of the Navy Department is carried on at the aerodynamical laboratory of the Washington Navy Yard and at the naval aircraft factory, Philadelphia Navy Yard. At the Washington Navy Yard two wind tunnels are in operation, and during the year a large number of airplanes and seaplane models have been given routine tests, and tests on many new aerofoil sections have also been made. Special attention has been given to testing streamline forms and struts. Yawing tests were conducted on the EP and the IE envelopes, which are formed from mathematical curves and have very low resistance. The tests indicate that the yawing moment about the center of gravity of a bare streamlined form varies but little from one shape to another. In connection with the tests on struts, it was shown that the Navy I strut has approximately 15 per cent less resistance than that given for the "Best" strut by the National Physical Laboratory. Wind tunnel tests were also conducted on two airship cars, one of faired contour and the other with facets of the same general contour, the results of which show the great value of fairing. The resistance of the faired car was 15 per cent less than that of the unfaired.

In connection with the wind tunnel at the Washington Navy Yard, a new aerodynamic balance of great interest has been developed. The balance is so designed that all adjustments of weights to bring the balance into equilibrium are automatic, and the time required for testing and the number of skilled operators are thus much decreased.

The Bureau of Construction and Repair has also undertaken the development and construction of the following instruments:

A precision recording barograph intended for use in airplane trials, and especially for measuring the landing angle of airplanes, for which no wind tunnel test is available. This instrument will have a range of from 0 to 5,000 feet, and will incorporate the desirable features of the present Bureau of Standards precision altimeter.

Two thermometer altimeters and density indicators. These instruments will combine a thermometric element with a pressure element in such a manner as to show at all times the altitude corrected for temperature.

Two instruments intended to measure quantitatively the permeability of gas cells of envelopes without the removal of samples. The construction of these instruments has been suggested by the technical staff of the Bureau of Standards. This instrument is to take the form of a cup of suitable area which is pressed against the envelope at the point where the permeability is to be determined. A current of air is either sucked or driven through the cup, sweeping it out at a known rate. The mixture of gas and air from the cup is then passed through a thermal conductivity cell, and the proportion of hydrogen contained in the mixture is determined from the thermal conductivity of the mixture.

In the high-speed wind tunnel at McCook Field, which is operated under the direction of the technical staff of the Army Air Service, work has been continued along the same general lines as those indicated in technical report No. 83. During the year it is contemplated that tests will be conducted to determine the flow around a sphere and around biplane combinations. It is hoped thus to determine how nearly the action of the visible vapor particles indicate the true air flow about a body, and to visualize the flow around combinations of more than one supporting surface so as to determine the nature of the interference between the upper and lower surfaces should be of the greatest interest. It is also hoped to photograph the vapor action about a sphere over as large an air-speed range as possible. The sphere is to be supported in a manner to produce a minimum disturbance due to the support, and the photographs obtained are to be compared with existing photographs of flow about spheres and with the theoretical streamlines.

It is also hoped that tests will be conducted to determine the effect of rake and tapered wing tips on air flow, as this information may make it possible to further improve the airplane form and nature of taper in wings.

Performance tests are also conducted at McCook Field, and the committee on aerodynamics has requested that special tests be made on longitudinal stability to obtain an index of the dynamic longitudinal stability of the various airplanes used by the Army. The work already done by the staff of the National Advisory Committee for Aeronautics at Dayton with the cooperation of the Engineering Division of the Air Service on five airplanes is but a beginning of longitudinal stability investigation. It is desirable to obtain readings of stick forces and elevator angles on every type of machine in the Army's possession, and to have curves plotted in the same way as in National Advisory Committee's report No. 96.

The investigations carried on at the two wind tunnels of the Bureau of Standards under the direction of Dr. L. J. Briggs have consisted largely in instrument calibration and testing. The principal research has been in connection with the resistance of spheres and projectiles.

The work of the Aeronautic Instruments Section of the Bureau of Standards comprises the investigation, experimental development, and testing of aircraft instruments; also the development of methods of testing, fundamental researches on the physical principles involved in such instruments, and the study of their behavior in actual service.

The more important investigations which have been undertaken by the section during the past year are as follows:

An investigation has been completed and prepared for publication through the National Advisory Committee for Aeronautics on the effect on the performance of Venturi tube airspeed indicators of changes in atmospheric pressure. The results show that in certain instruments commonly used a correction should be applied for the viscosity of the air, a factor which has not hitherto been taken into account. This is of special interest in dirigible work where the air speeds may be low, and also in aircraft performance tests where exceptional precision is required.

An altimeter of exceptional accuracy designed and made at the Bureau of Standards has been completed and submitted to the Army. Another model with additional improvements has recently been designed and is under construction.

At the request of the National Advisory Committee for Aeronautics a fundamental investigation of the factors determining the behavior of flexible diaphragms as used in aeronautic instruments has been undertaken. The irreversible effects which cause the lag in diaphragm instruments has been formulated mathematically. The relation between force and deflection for diaphragms of different sizes, thickness, and materials has been studied graphically, practical methods for spinning diaphragms and building up diaphragm boxes have been investigated, and the possibilities of mechanical seasoning by repeated stress considered.

An improved rate of climb indicator, which indicates directly the rate of climb of aircraft in hundreds of feet per minute, has been completed and tested, and specifications have been prepared for the Army to use in the manufacture of a number of these instruments.

Information regarding instruments available for aerial navigation in cloudy weather or at night or for long-distance flights has been compiled at the request of the National Advisory Committee for Aeronautics and the Air Mail Service by the Aeronautic Instruments Section. This work will be continued and the development of new instruments undertaken.

Other investigations have been the development of a motion-picture apparatus for recording instrument readings during the flight of an airplane; a study of the errors in instruments used for determining the direction of aircraft, such as gyroscopic and liquid inclinometers and banking indicators, gyroscopic and magnetic compasses and turn indicators, a systematic investigation of commercial sphygmomanometers; a paper on the results of investigations on German instruments; a statistical study of the causes of failure in aeronautic instruments.

Assistance has been given the Air Service, the Aero Club of America, and others interested during the past year in the world's altitude competition for airplanes. Instruments have been calibrated and the best procedure for determining the altitude attained formulated.

REPORT OF COMMITTEE ON POWER PLANTS FOR AIRCRAFT.

Following is a statement of the organization and functions of the committee on power plants for aircraft:

ORGANIZATION.

Dr. S. W. Stratton, chairman.
Commander A. K. Atkins, United States Navy, vice chairman.
Henry M. Crane, Wright Aeronautical Corporation.
Harvey N. Davis, Harvard University.
Dr. H. C. Dickinson, Bureau of Standards, acting secretary.
L. M. Griffith, Langley Memorial Aeronautical Laboratory.
Capt. G. E. A. Hallett, United States Army.
G. W. Lewis, National Advisory Committee for Aeronautics.
J. G. Vincent, Packard Motor Car Co.

FUNCTIONS.

1. To aid in determining the problems relating to power plants for aircraft to be experimentally attacked by governmental and private agencies.
2. To endeavor to coordinate, by counsel and suggestion, the research and experimental work involved in the investigation of such problems.
3. To act as a medium for the intercharge of information regarding aeronautic power-plant investigations in progress or proposed.
4. The committee may direct and conduct research and experiment on aeronautic power-plant problems in such laboratory or laboratories, either in whole or in part, as may be placed under its direction.
5. The committee shall meet from time to time on call of the chairman and report its actions and recommendations to the executive committee.

By reason of the representation of the Army, the Navy, and the industry upon this subcommittee, it has been possible to maintain close contact with the research and development being carried on in this country and to exert an influence toward the expenditure of energy on those problems whose solution appears of the greatest importance, as well as to avoid waste due to unnecessary repetition of research. The activities of this committee can be advantageously considered under the following main classes of problems relating to aircraft power plants:

New engine types.
Performance characteristics of aircraft engines.
Supercharging compressors.
Improvement of engine details.
Cooling problems.
Radiating systems.
Carburetion systems.
Ignition systems.
Fuels and combustion.
Lubricants and lubrication.
Cylinder pressure indicators.
Miscellaneous problems.
Extension of laboratory facilities.

NEW ENGINE TYPES.

Owing to the disadvantages of high fire risk, high fuel cost, carburetion and ignition difficulties, low service reliability, and specific power limitation of the four-cycle engine, constantly increasing interest has been shown in the fuel injection engine of both automatic and electric ignition types. The two-cycle fuel injection automatic ignition engine appears especially promising. The problems incident thereto are being energetically studied abroad and some work is being done in this country. In particular, the Bureau of Engineering of the Navy Department has recently approved a fuel injection research program to be carried out at the Langley Memorial Aeronautical Laboratory, as the development of a successful engine of the fuel injection type is of especial interest to the Navy in connection with the power plants of large airships.

The program for the immediate future covers the study of the phenomena of fuel injection by means of a special glass-walled pressure chamber, in which many of the engine operating conditions may be simulated, equipped with apparatus for taking very high-speed photographs of the events occurring in the pressure chamber. The results are to be applied to an experimental engine and a study made of the possibilities of the double-piston two-cycle engine in this connection. The problem of altering standard carbureted four-cycle engines will receive attention as well.

The direct air-cooled engine offers important possible advantages which have been studied by foreign laboratories and to a small extent by those in this country, largely in connection with the general problem of radiation. In connection with direct fuel injection, the air-cooled engine is especially interesting as looking toward the increase of thermal efficiency and the reduction of engine weight.

The program provides for the continuation of the research into the problem of direct transfer of heat from cylinder walls to air, and, if possible, the extension of the results to the development of efficient cylinder forms.

The development of a radial engine of the air-cooled type has received the serious attention of research laboratories of both Great Britain and France. In this country very little has so far been done along this line, but the Air Service of the Army at McCook Field is now undertaking the problem, and at the present time is developing at two outside laboratories radial air-cooled engines.

The Army Air Service has nearing completion an 18-cylinder engine of 600 to 700 horsepower.

The development of an engine particularly suitable for a power unit for the operation of lighter-than-air craft is now being carried on by the Navy Department. To further this work and obtain an engine of general characteristics, and still allow leeway for individual design of detail, the Bureau of Engineering has let contracts to three separate engine manufacturers. The general specifications call for an engine with six cylinders in line, of approximately 300 horsepower, the main characteristics of which will be low fuel and oil consumption, together with a high degree of reliability. One engine of this class being constructed is of the Ricardo type, as it is hoped that by the use of the Ricardo principle of construction the life and reliability of the engine will be greatly increased.

PERFORMANCE CHARACTERISTICS OF AIRCRAFT ENGINES.

An investigation of the performance characteristics of aircraft engines, with special reference to conditions met with in flight, is being carried on in the altitude chamber of the Bureau of Standards. The purpose of the investigation is to secure information to be used in the selection of engines for specific purposes, and in improving the design of power plants. In conducting this research work, observations of the performance characteristics by means of the altitude laboratory equipment were made to supply information on a variety of subjects; such observations were made on a number of typical engines covering the following subjects:

(*a*) Conditions attendant upon supercharging (with special reference to Liberty engine).

(*b*) Study of indicated horsepower under altitude conditions.

(c) Study of relations between air to fuel ratio and maximum power at full and part throttle.
(d) Effect of intake air temperature, jacket-water temperature, etc., on performance.
(e) Study of mechanical losses.
(f) Comparison of performance of different types of spark plugs in operation.

SUPERCHARGING COMPRESSORS.

The General Electric exhaust turbine-driven centrifugal compressor as developed by the Air Service and Dr. S. A. Moss has been continuously perfected until it may be considered as a proved device. It is, in fact, being ordered in appreciable numbers for the equipment of the Liberty–12 engine. The gain in power output at high altitudes is sufficient to much more than offset the added weight.

The Sturtevant centrifugal-blower type, driven by gears and belt from the engine crankshaft, has been developed so far as to demonstrate its value, and its application to the Liberty–12 is now in progress. The gear-driven centrifugal compressor is being investigated and the Air Service has partly developed a hydraulic clutch for the same to eliminate the destructive effect of the very high inertia of the rotor.

The positive gear-driven centrifugal-fan type eliminates the high inertia forces by reason of the very low inertia of the rotor. This type was partially investigated by the engineering division of the American Expeditionary Forces in Paris during the last months of the war, and its further investigation has been undertaken by this committee at Langley Field.

The postive blower of the Root type is also being studied by the committee at the Langley Field laboratory, an experimental model having been built to supercharge the Liberty–12 up to 20,000 feet altitude. This device, as also the fan type above mentioned, are being subjected to performance tests on the dynamometer and will later be tested in flight if advisable.

The program of the committee for the immediate future involves the closest contact with all progress in this field, with a view to the early solution of the problem of the most desirable form of compressor and drive. The investigation in our own laboratory will be pushed forward as fast as possible consistent with accurate results.

IMPROVEMENT OF ENGINE DETAILS.

Little research has been carried out during the past year under this head. Perhaps the most important item is the study of the use of Monel metal for exhaust valves, the results of which indicate conclusively that this metal has quite favorable thermal properties and is on the whole well adapted for this purpose. Some work has also been done upon the effect of varying width of exhaust-valve seat upon the valve-head temperature. The "mercury-cooled" exhaust valve was developed by private laboratories and their conclusions have to a certain extent been checked by other laboratories, indicating that this valve does maintain a lower head temperature than the ordinary valve, provided the thermal capacity of the stem and contacting guide is sufficient to transfer the heat which the mercury carries from the exhaust-valve head to the stem. Some attention has also been paid to the question of reducing the temperature of the piston head, although sufficient data has not as yet been secured to justify any general conclusion.

The program for the future emphasizes the necessity for the energetic study of the piston head temperature, inasmuch as this is likely to prove one of the limitations to the continuous increase of brake mean effective pressure, and is of especial importance in connection with the two-cycle engine. Provision is also made for the continuation of the study of exhaust-valve temperatures and the thermal resistance of such constructional expedients as threaded joints in the combustion-chamber walls.

COOLING PROBLEMS.

The problem including water cooling and direct-air cooling of aircraft engines has been undertaken to complete the unfinished portion of the research on cooling radiators and to

secure data desired for the design of direct air-cooled engines by means of fundamental laboratory experiments, mathematical analysis of these researches, and observations made in flight to check the foregoing. This work is to be undertaken by—

(*a*) Tests of new designs of radiator cells as developed.

(*b*) Study of heat and temperature distribution in model air-cooled cylinders by laboratory method and in cylinders of engines under operating conditions.

(*c*) Pressure and temperature gradients in air tubes of radiators.

(*d*) Mathematical analysis of results of the foregoing.

(*e*) Flight tests of radiators of standard and special construction to verify laboratory and mathematical study.

RADIATION SYSTEMS.

During the past year the testing of sample radiator cores has continued, together with the application of the results to full-size radiators. At the same time the results of previous work in this field have been collated and placed in such form as to be of direct value to the designing engineer. The results of research have to a large extent been checked by tests carried out by the Air Service at McCook Field.

The program provides for the continuation of the testing of interesting radiator core samples, as well as for the extension of the laboratory work and the verification of the fundamental relations. The scientific development of the direct air-cooled engine in reality comes under this same class, since the problems are quite similar to those of radiation involved in the water-cooled engine radiator, except that the temperature gradients are greatly increased. The fundamental data secured in the study of water radiators is of the greatest value in connection with the study of direct air-cooled engine cylinder.

CARBURETION SYSTEMS.

During the past year the problem of automatic carburetor compensation for altitude has received additional study and some work has been done in connection with its development in free flight tests. However, the growing importance of the supercharger has to a certain extent lessened the importance of this problem. A mathematical study has been made of the laws of flow of air and fuel in carburetors as affected by changes in altitude, and this is available as a foundation for the further development of automatic or inherent compensation means. The problems of the atomization and mixing characteristics of aircraft carburetors have received some additional experimental study, but the subject is so involved that it has not as yet been possible to develop any satisfactory foundation for the scientific comparison of the experimental results from different carburetors. These problems are interrelated to those of the optimum fuel to air ratios and the modifying effects of differing inlet manifold forms. An effort is being made to lay a proper foundation for the study of all of these problems, and it is believed that fundamental information will shortly be available. The data yielded by the testing of aircraft engines in the altitude chamber has been supplemented by tests of the carburetors alone in the carburetor test plant.

The program provides for the continuation of research in all of these problems, and in addition to study the question of the most advantageous form of poppet valves and ports as viewed from the standpoints of charging efficiency and minimum interference with mixture conditions.

The program also provides for a study of carburetion and manifolding for the purpose of securing fundamental data on the metering characteristics of aircraft carburetors with special reference to their performance at low-air densities. This will require also a study of the physical constants of aircraft fuels, such as vapor pressure, vapor volume, viscosities, etc., and of pulsating flow of manifolds. This work is to be carried on by—

(*a*) Study of various types of carburetors and the carburetor metering devices in the carburetor test plant.

(*b*) Subsequent checks on their performance when mounted on engines in the altitude laboratory.

(*c*) Study of physical constants of fuels by means of physical-chemical laboratory equipment, which has been developed for the purpose.

(*d*) Further mathematical analysis of the problem of melting at different air densities.

(*e*) Laboratory experiments to determine the effects of pulsating air flow on metering of fuel.

IGNITION SYSTEMS.

The work performed in this field has consisted largely of the more or less routine testing of new forms of spark plugs and spark-plug insulators. The mathematical theory of the electrical side of the ignition system has been worked out in detail and checked by laboratory experiments. The use of a series gap in the ignition secondary has received some additional attention and apparatus has been assembled for the continuation of research in this field.

The program for the future provides for the study of the effect of spark quality and intensity on the engine performance, the effect of the electrode temperature upon the breakdown voltage of the spark plug gap, as well as a continuation of the research into the auxiliary spark gap. The program also provides for the study of problems of ignition to secure information on the relative performance of different types of ignition systems, including spark plugs, magnetos, battery systems, etc., and the research will include:

(*a*) Laboratory and engine tests of spark plugs.

(*b*) Study of the effect of spark quality and intensity on engine performance.

(*c*) Study of the breakdown volume of spark gaps in normal operation.

(*d*) Further verification of the mathematical theory of the magneto.

Much of this work can be conveniently performed in connection with the research on the propagation of flame.

FUELS AND COMBUSTION.

During the past year experiments have been made with a number of compounded fuels, intended to reduce the tendency to knocking characteristics of the ordinary aviation gasoline. The results have demonstrated that there are a number of such compound fuels which will permit of a very considerable elevation of the compression ratio and compression temperature without difficulty. The effect of the admixture with gasoline or benzol, alcohol, or other substances, hydrocarbon or not, is now reasonably well known. It is possible to specify the proportions of a compound fuel to withstand any reasonable compression ratio, without developing disagreeable knocking conditions. The future program provides only for the testing of such fuel blends as may be introduced from time to time, it being considered that the fuel injection type of engine will ultimately eliminate the necessity for such compounded fuels as are now necessarily used in the very high compression carbureted engine.

The measurement of the rate of flame propagation in aircraft engine cylinders has been continued during the past year, through the medium of a specially equipped single cylinder Liberty engine. While the results of the investigation indicate a velocity of flame propagation in the order of 20 to 40 feet per second for ordinary operating conditions, it is felt that the data secured is not yet sufficient to justify the issuance of a general report upon the subject. To a large extent, this work is tied up with the investigation of the problem of detonation of charge, and the general problem of the chemical relations due to combustion. It is felt that investigation of all three general phases should be conducted simultaneously, in order that the resulting information may be of a fundamental instead of a particular nature. The program provides for the continuation of this work during the coming year, as it is believed that these are subjects of which too little is now known, in view of their vital bearing upon the desired continuous increase of brake mean effective pressure and thermal efficiency.

This work is being continued at the Bureau of Standards, and in conducting the tests it is hoped that data bearing on the following items of special importance will be obtained:

(*a*) The relation between rate of combustion and pressure distribution in engine cylinders.

(*b*) The velocities of propagation of flame in engine cylinders and in explosive mixtures as measured by laboratory methods, with special reference to the relative velocities for different fuels and different mixture ratios.
(*c*) Measurements of the apparent flame temperatures in engine cylinders under various conditions of operation by direct observation and spectrum analysis, if possible.
(*d*) The ignition temperatures for mixtures of fuel and air under various conditions.
(*e*) Analysis of the intermediate products of combustion secured by means of a sampling valve at different points in the combustion cycle.

LUBRICANTS AND LUBRICATION.

During the past year a series of tests were run for the military authorities to determine the relative merits of oils refined from different types of crude, giving results of a comparative nature only. Very little work has been done upon the fundamentals of cylinder lubrication, so that the real reason for the slight superiority of one oil over another has not as yet been determined. The future program provides for the comprehensive investigation of lubrication phenomena and the properties of oils suitable for aircraft engine lubrication, especially important because existing specifications do not guarantee an oil that will always answer the requirements.

CYLINDER PRESSURE INDICATORS.

A very satisfactory step-by-step diaphragm type indicator has been perfected during the past year and a number of instruments built and used in the regular work of the laboratory. The characteristics of preignition or knocking, as well as the normal operating cycles, have been studied with these instruments and pressures have been recorded of nearly 1,000 pounds per square inch. This instrument has also been developed in an automatic form, in which the balancing of the pressures on the two sides of the diaphragm is automatically provided for. This modification permits of more rapid operation and is probably the type which will be settled on for further development in the laboratory. Two other types of indicators are in the preliminary stage and will be further investigated. The program provides for the continuous perfection of such instruments in order that the study of the pressure changes in the cylinder may be carried out with the greatest degree of accuracy. It is intended that such studies shall form a part of all important investigations involving the operation of aircraft engines.

MISCELLANEOUS PROBLEMS.

Owing to the pressure of more important work, practically nothing has been done during the past year with reference to the development of mufflers for aeronautic engines, except as is necessarily involved in the general problem of the condensation of water from the exhaust gas. Considerable experimental work has been done on the latter problem and the general requirements have been roughly determined. Apparatus has been designed for the continuation of these experiments under free flight conditions upon the power plants of dirigibles.

The program provides that this work shall be continued and also for the further development of mufflers for those installations which do not involve water condensation.

EXTENSION OF LABORATORY FACILITIES.

During the past year the new engine laboratory has been completed at the Bureau of Standards, thus making available to the committee extensive facilities for the conduct of its work. A total of six dynamometers are at present provided, and these are so arranged in connection with the altitude chambers that engines as large as 800 horsepower may be tested therein. A complete complement of auxiliary equipment is included, and the installation as a whole is extremely convenient.

During the year some progress was made in the installation of the committee's own dynamometer laboratory equipment at its Langley Field laboratory. The small amount of available funds has, however, so limited the size of the staff that the work of installation is not

yet complete. For the same reason it has not been possible to carry out much actual research work in this laboratory, the work so far accomplished being largely in the nature of preliminary investigations to serve as a basis for the research being conducted on the two types of supercharger. It is expected that the funds available for this laboratory during the forthcoming year will permit the very considerable extension of the staff and the conduct of work on a scale commensurate with the amount of apparatus available. The equipment consists mainly of five dynamometers ranging from 2 to 450 horsepower capacity, together with a certain amount of auxiliary apparatus. The equipment is temporarily housed in a standard Army steel hangar, a building not at all suited for the purpose, and it is hoped that in the near future it will be possible to replace this with a permanent building of more substantial construction.

The engine laboratory facilities of the United States Army Air Service station at McCook Field have been increased so that that organization is well provided with means for the conduct of the experimental development of power plants.

The Navy Department maintains a small dynamometer laboratory in the navy yard at Washington the equipment of which has been augmented by making available two electric dynamometers coupled together. This laboratory has been largely engaged on the problem of water recovery from the exhaust.

REPORT OF COMMITTEE ON MATERIALS FOR AIRCRAFT.

Following is a statement of the organization and functions of the committee on materials for aircraft:

ORGANIZATION.

Prof. Charles F. Marvin, chairman.
Dr. G. K. Burgess, Bureau of Standards, vice chairman.
Capt. H. W. Flickinger, United States Army.
Dr. Henry A. Gardner, Institute of Industrial Research.
Prof. George B. Haven, Massachusetts Institute of Technology.
Commander J. C. Hunsaker, United States Navy.
Dr. Zay Jeffries, Aluminum Co. of America.
Prof. William Walker, Massachusetts Institute of Technology.
Prof. E. P. Warner, Massachusetts Institute of Technology.
Prof. H. L. Whittemore, Bureau of Standards, acting secretary.

FUNCTIONS.

1. To aid in determining the problems relating to materials for aircraft to be experimentally attacked by governmental and private agencies.

2. To endeavor to coordinate, by counsel and suggestion, the research and experimental work involved in the investigation of such problems.

3. To act as a medium for the interchange of information regarding investigations of materials for aircraft, in progress or proposed.

4. The committee may direct and conduct research and experiment on materials for aircraft in such laboratory or laboratories, either in whole or in part, as may be placed under its direction.

5. The committee shall meet from time to time on call of the chairman and report its actions and recommendations to the executive committee.

The committee on materials for aircraft, through its personnel acting as a medium for the interchange of information regarding investigations on materials for aircraft, is enabled to keep in close touch with research in this field of aircraft development.

Much of the research, especially in the development of light alloys, must necessarily be conducted by the industries interested in the particular development, and both the Aluminum

Co. of America and the American Magnesium Corporation are represented on the committee. In order to cover effectively the large and varied field of research on materials for aircraft three subcommittees were formed, as follows:

Subcommmittee on metals (Dr. G. K. Burgess, chairman).

Subcommittee on woods and glues (Prof. H. L. Whittemore, chairman).

Subcommittee on coverings, ropes, and protective coatings (Dr. Henry A. Gardner, chairman).

Most of the research in connection with the development of materials for aircraft is financed directly by the Bureau of Construction and Repair of the Navy Department and the Engineering Division of the Army Air Service.

The Bureau of Construction and Repair not only conducts research at its aerodynamical laboratory at the Washington Navy Yard and at the naval aircraft factory in Philadelphia, but also apportions and finances research problems to the Bureau of Standards, the Langley Memorial Aeronautical Laboratory, the Institute of Industrial Research, and the Forest Products Laboratory.

SUBCOMMITTEE ON METALS.

In a report of the progress that has been made in regard to the manufacture and utility of light alloys it is well to state that much of this progress is the result of investigations either started during the war and completed since the armistice or of work that has been a natural consequence of experience gained during the war. Of those light aluminum alloys which can be worked, duralumin or material of similar composition, because of its inherent possibilities, is probably the most widely used and naturally has received the most attention at the hands of the investigators. Dr. Merica and his associates at the Bureau of Standards have greatly increased the knowledge of the manufacture and heat treatment of duralumin, as reported in Bureau of Standards Scientific Paper No. 347. This paper shows that it is advisable to preheat the ingots previous to rolling somewhat higher than was customary, namely, to preheat to 500° C. and then roll to 450° C. The best quenching temperature was found to lie between 510° and 515° C., and quenching should be in hot water. The mechanical properties of the finished material are quite dependent upon the artificial aging process, but for most purposes it was found best to age at 100° C. for about five to six days. A theory of the mechanism of the hardening of duralumin was developed, which has been further amplified by Dr. Zay Jeffries.

Duralumin may be drop-forged as well as rolled, and some interesting tests on drop-forged connecting rods are given by Rollason, who found that the aluminum alloy rods withstood impact fatigue better than ordinary steel forgings.

Gibson has also investigated the fatigue resistance of various duralumins and concludes that, weight for weight, forged and heat-treated duralumin is equal to, if not superior to, forged steel in its fatigue-resisting properties. He also states that under certain limitations as to stresses involved that it is comparable with steel on a volume-for-volume basis.

As an example of the increasing use of duralumin there might be cited the all-metal airplanes, such as the Larsen or others similar to the German Junker models. These airplanes use duralumin for wing surface coverings in place of fabric as well as for structural members. For the latter purpose seamless tubing is essential, although to date satisfactory sources of supply have not become available in the United States.

Many of the light casting alloys have been studied by Merica and Karr, as reported in Bureau of Standards Technologic Paper No. 139, and they determined the tensile properties, hardness, resistance to corrosion, and resistance to the action of alternating stresses of a number of compositions. The effect of various additional elements, such as copper, zinc, manganese, magnesium, and nickel were studied, and these investigators showed that certain of the casting alloys were also subject to beneficial results from heat treatment. This practice was commended to the manufacturers of castings for realization of its commercial possibilities.

Jeffries and Gibson also investigated the effect of heat treatment upon cast aluminum alloys and suggested that more uniform results could be obtained by heating the castings in a bath of fused niter followed by quenching in oil, thus reducing to a minimum the tendency for the atmosphere to permeate and oxidize the interior of porous castings.

R. J. Anderson has published several articles on aluminum castings and foundry practice, particularly with a view to producing sound castings, free from blowholes and hard spots.

The metallography of aluminum and its alloys has also received some attention; Merica, Waltenberg, and Freeman studied aluminum and its alloys with copper and with magnesium. The various constituents were identified and the temperature solubility curves of $CuAl_2$ and of Mg_4Al_3 determined. Anderson studied the metallography of commercial aluminum and aluminum in ingot form and compared the microstructure, macrostructure, and fracture of tough and brittle ingots.

For a comprehensive investigation of the constitution and positive identification of the constituents in aluminum it is necessary to start with pure aluminum. The best aluminum now obtainable is seldom better than 99.8 per cent pure. Efforts to produce aluminum of greater purity have not been successful thus far.

The corrosion of the rolled light alloys was investigated by Merica, Waltenberg, and Finn, using three ternary series, Al–Mg–Cu, Al–Mg–Mn, and Al–Mg–Ni. The alloys of the Al–Mg–Mn series resisted corrosion in general better than the others. Hard-rolled commercial aluminum corrodes much more than any of the alloys, annealed aluminum was more resistant to corrosion than the hard-rolled aluminum, but did not compare favorably with the alloys. Bureau of Standards Technologic Paper No. 132 also gives the mechanical properties of the various alloys in the cold-rolled, annealed, and heat-treated conditions.

The Bureau of Standards, in cooperation with the Navy Department, also conducted tests on the corrosion of aluminum and its alloys by sea water, both unprotected and with various protective coatings. Presence of oil on the water where the plates were exposed lends some doubt to the results, but the indications were that unprotected duralumin has practically the same resistance to corrosion as that which has been protected. Other findings were practcially as above.

Among the new light alloys which have been brought out, "Dow metal" is quite interesting. This alloy is said to contain over 90 per cent magnesium and to have a specific gravity of 1.79. Castings have a tensile strength of from 22,000 to 25,000 pounds per square inch; yield point, 12,000 to 14,000 pounds per square inch; elongation, 3.5 per cent in 2 inches; reduction of area, 3.5 per cent; and Brinell hardness of 55 to 75. The sand castings are subject to heat treatment, such procedure increasing the tensile strength to 30,000 pounds per square inch and elongation and reduction of area to 6 per cent each. The alloy may also be worked, drop forgings having a tensile strength of 50,000 pounds per square inch and Brinell hardness of 70. No data are given in the literature on this alloy as to the method of casting which heretofore has been a great drawback in producing magnesium rich alloys, due to the affinity of magnesium for oxygen, nitrogen, etc.; Waltenberg and Coblentz in preparing aluminum magnesium alloys resorted to vacuum casting in order to produce sound material.

In this connection, in an article by Thomas on the casting of elektronmetall containing about 80 per cent magnesium and the balance aluminum and zinc, it is stated that great care must be exercised in selecting the sand for molding and that the molds must be thoroughly dried to get rid of all moisture. The alloy is melted in wrought-iron or cast-steel crucibles, as magnesium will take up the silica of graphite crucibles. The crucibles are covered with an iron cover to reduce oxidation, the pouring temperature must be closely controlled (just above melting point) and the melt poured directly after reaching the proper temperature. The alloy is brittle down to 100° C. and the casting must not be disturbed until cold. He gives illustrations of very sound castings produced in this manner.

The naval aircraft factory conducted a test of the comparison of the effect of punched and drilled rivet holes on the physical properties of duralumin sheet, and to determine whether

punching is harmful to the material. The general conclusion of this investigation is that rivet holes made by punching will not weaken duralumin sheets more than those made by drilling, and comparative tests made on duralumin sheets in the four states in which it is likely to be handled showed the punched specimens to be slightly stronger than the drilled specimens.

The Bureau of Construction and Repair of the Navy has also authorized an investigation of the strength and fatigue value of duralumin as affected by heat treatment and working. This investigation will be conducted at the Bureau of Standards along lines that have been proposed by the committee on materials for aircraft.

A special report is being prepared by A. M. Hunt, director of research of the American Magnesium Corporation, Niagara Falls, N. Y., on the present status of magnesium alloys, with special reference to their possible use in connection with aircraft.

The Engineering Division of the Air Service of the Army is making extensive preparations to carry on experiments in connection with the development of aluminum and magnesium alloys for use in the construction of aircraft.

SUBCOMMITTEE ON WOODS AND GLUES.

Most of the research in connection with the development of woods and glues to be used in construction of aircraft is carried on at the forest-products laboratory at Madison, Wis. The forest-products laboratory is developing for the Bureau of Construction and Repair a 17-foot strut for a Navy flying boat, and is also engaged in the development of a streamline strut and the determination of the dimensions of wide struts of uniform strength. Tests have also been made upon airship girders. Improvement has been made in the formula for a waterproof glue developed by the forest-products laboratory for use in gluing wood. Increasing or decreasing greatly the proportion of sodium silicate shortens the life of the glue, although this has little effect on the strength or water resistance. The life of the glue is less the greater the amount of lime content. Water resistance increases with the increase of the amount of lime; very rapidly at first, but later more slowly. With very high lime content the water resistance falls off slightly.

Weathering tests were also conducted on all-veneer wing sections. The all-veneer wing used was built and previously tested by sand loading, and was cut into five sections and covered with varnish and aluminum leaf for protection. The usefulness of both veneers appeared to be equal, but the surface of the one protected by aluminum leaf was in better condition than the one protected by spar varnish.

Strength tests were also conducted on screw fastenings of plywood, with the result that it was found that round-head screws with washers give about the same strength at flat-headed screws of the same size without washers. The tests resulted in recommendations for screw sizes, margin, and spacing for use with various species and thicknesses of plywood.

The Bureau of Construction and Repair also authorized the investigation of the strength of plywood. The object of these tests was to secure comparative data on properties of different species, effect of varying ratio of thickness of core to total panel thickness, and the effect of low density species in core to high density species on faces. Stiffness in bending, tensile strength, resistance to splitting, and toughness increase with the increase of the density of the plywood, and column bending and tensile strength of 3-ply wood are greater when force is applied perpendicular. Strength and stiffness in column bending decrease as the number of plies increase when the force is applied parallel to the grain and increase when force is applied perpendicular to the grain.

SUBCOMMITTEE ON COVERINGS, DOPES, AND PROTECTIVE COATINGS.

The Bureau of Construction and Repair is developing a substitute for goldbeater's skin. It is intended to use this material in the construction of gas cells for rigid airships, and possibly in the construction of envelopes for nonrigid airships. The material has properties approach-

ing the properties which made goldbeater's skin of value, namely, high resistance to the passage of hydrogen gas, and lightness. It is capable of being produced in sheets of large area, and cemented with comparative ease.

The Bureau of Construction and Repair has requested the Bureau of Standards to develop a cement for the attaching of goldbeater's skin to cotton cloth. The material should resist the action of moisture, and at the same time remain permanently flexible and adherent. It is intended to replace the rubber cement at present in use. The latter is understood to have a slow deteriorating effect on goldbeater's skin. The new cement should have no deteriorating effect, and if possible actually have a preservative action.

The Bureau of Standards is also developing a substitute for rubber proofing in balloon fabrics. It is believed that other substances may have the properties of being impermeable to hydrogen in equal or better degree than rubber, and may also be made flexible and capable of application in very much the same manner as that used with rubber.

TECHNICAL PUBLICATIONS OF THE COMMITTEE.

During the past year, the committee on publications and intelligence has recommended the publication of 28 technical reports, a summary of which follows. The reports cover a wide range of subjects on which research has been conducted under the supervision and cognizance of the various subcommittees, each report being approved by the subcommittee interested and recommended for publication to the executive committee. The technical reports presented represent fundamental research in aeronautics carried on at different aeronautical laboratories in this country, including the Langley Memorial Aeronautical Laboratory, McCook Field, the Aeronautical Laboratory at the Washington Navy Yard, the Bureau of Standards, and the Leland Stanford Junior University.

Considerable technical information is obtained by the committee that is of immediate interest to those interested in experimental and research problems in connection with aeronautics. To make this information immediately available, the National Advisory Committee for Aeronautics has authorized the committee on publications and intelligence to issue a series of "Technical Notes." In accordance with this authorization, the committee has issued 16 technical notes, on subjects that were of immediate interest, not only to research laboratories, but also to airplane manufacturers. A list of the technical notes issued during the year follows the general summary of the technical reports.

The first annual report of the National Advisory Committee for Aeronautics contained technical reports Nos. 1 to 7; the second annual report, Nos. 8 to 12; the third annual report, Nos. 13 to 23; the fourth annual report, Nos. 24 to 50; the fifth annual report, Nos. 51 to 82; and since the preparation of the fifth annual report, the committee has issued the following technical reports, Nos. 83 to 110:

Report No. 83, entitled "Wind Tunnel Studies in Aerodynamic Phenomena at High Speed," by F. W. Caldwell and E. N. Fales, Engineering Division, Air Service, McCook Field.—A great amount of research and experimental work has been done and fair success obtained in an effort to place airplane and propeller design upon an empirical basis. However, one can not fail to be impressed by the apparent lack of data available toward establishing flow phenomena upon a rational basis, such that they may be interpreted in terms of the laws of physics.

With this end in view it was the object of the authors to design a wind tunnel differing from the usual type especially in regard to large power and speed of flow. This involved features whose suitability could not be predicted; for, after all available information has been secured on full-size and model wind tunnels in various parts of the world, there remains much obscurity about the air flow phenomena. It is the assumption of Dr. George de Bothezat that the type of air flow which establishes itself is governed by the stresses set up in the air passing the aerofoil. The stresses increase as the velocity rises, and it is easy to conceive that a given type of flow is possible only so long as the shearing stresses developed in the fluid do not exceed a certain magnitude which depends on the value of the viscosity coefficient.

Experimental investigation of the flow has heretofore been rather unsuccessful because of lack of adequate methods. The writers laid out the design of the McCook Field wind tunnel to investigate the scaling effect due to high velocities of propeller aerofoils. During the course of the experiments, however, it was found impossible to visualize the air flow. The velocities of the air flow obtained by the writers offers a solution to one of the fundamental problems of aerodynamics, namely, the quantitative empirical measurement of the phenomena of fluid dynamics pertaining to flight and air flow. The method described in the report for visualizing air flow depends upon the fact that the moisture in the air condenses as a fog when the temperature is reduced to the dew point, provided that there is a solid or liquid nucleus to start the condensation. In the McCook Field wind tunnel the temperature drop is brought about through expansion of the air during acceleration due to a drop of pressure of 100 inches of water. The relative humidity of the atmosphere can be artificially raised if too low, and the necessary nucleus for condensation is provided by the model tested. Flow vortices become readily visible, and the report contains many photographs showing the air flow past an aerofoil under different conditions.

Report No. 84, entitled "Data on the Design of Plywood for Aircraft," by Armin Elmendorf, Forest Service.—This report gives the results of investigations made by the Forest Products Laboratory of the United States Forest Service at Madison, Wis., for the War and Navy Departments. Sufficient discussion on the mechanical and physical properties of plywood is included so that the data may be intelligently used. The data, although primarily intended for aircraft design, have a broader field of application. The report makes available data which will aid the designer in determining the plywood that is best adapted to various aircraft parts. The results expressed in the report were determined through a comprehensive test of the mechanical and physical properties of plywood and of the way these properties vary with the density, number, thickness, and arrangement of the plies and the direction of grain of the plies.

Report No. 85, entitled "Moisture Resistant Finishes for Airplane Woods," by M. E. Dunlap, Forest Service.—This report describes briefly a series of experiments made at the Forest Products Laboratory, Madison, Wis., to determine the comparative moisture resistance of linseed oil, impregation treatments, condensation varnishes, oil varnishes, enamels, cellulose varnishes, rubber, electroplated and sprayed metal coatings, and metal-leaf coatings when applied to wood.

All coatings except the rubber and electroplated metal coatings, which were not developed sufficiently to make them practical, admitted moisture in varying degrees. The most effective and at the same time most practical coating was found to be that of aluminum leaf.

Tests were made by applying coatings to panels of yellow birch, care being taken that the panels were carefully smoothed and the corners rounded. In general, a coat of filler was first applied, followed by three coats of the coating material being studied; and in some cases the material applied required special methods of application. After the panels had dried thoroughly they were subjected to an atmosphere of the humidity of 95 to 100 per cent for 17 days.

Report No. 86, entitled "Properties of Special Types of Radiators," by S. R. Parsons, Bureau of Standards.—This report discusses the general performance characteristics of three special classes of radiators: Those with flat-plate water tubes, fin and tube types, and types that whistle in an air stream. Curves and tables show the performance of representative radiators of each class and compare the flat-plate and whistling types. Empirical equations are given for estimating the performance of flat-plate radiators of various dimensions.

The report also contains a brief discussion, with curves, showing the effect of yawing on the properties of a radiator.

It was found that a careful distinction should be made between radiators whose water tubes are smooth and other types using perforated plates or deep and narrow tubes placed

in rows one behind the other. Holes in water tubes or spaces between them in the direction of the air flow caused a great increase in head resistance and a decrease in mass flow of air, although the heat transfer per square foot of cooling surface was increased by the great turbulence caused by the use of perforations. The net result was a decrease in the figure of merit. The same result has been found in the case of turbulent vanes in cellular radiators, and, indeed, no type of radiator is known to the writer in which an artificial increase of turbulence is not accompanied by a decrease in the figure of merit.

Report No. 87, entitled "Effects of Nature of Cooling Surface on Radiator Performance," by S. R. Parsons and R. V. Kleinschmidt, Bureau of Standards. This report discusses the effects of roughness, smoothness, and cleanness of cooling surfaces on the performance of aeronautic radiators, as shown by experimental work, with different conditions of surface, on (1) heat transfer from a single brass tube and from a radiator; (2) pressure drop in an air stream in a single brass tube and in a radiator; (3) head resistance of a radiator; and (4) flow of air through a radiator. It is shown that while smooth surfaces are better than rough, the surfaces usually found in commercial radiators do not differ enough to show marked effect on performance, *provided the surfaces are kept clean.*

An accumulation of oil and dust on the surface will have a very harmful effect on the performance of the radiator. The heat transfer from an ordinary smooth surface may be increased 17 per cent in a good air flow by giving the surface a high polish, or it may be decreased 10 per cent or more by smoking the surface, and the figure of merit of the radiator then may be somewhat increased by polishing the surface, 6 to 10 per cent being observed in one case.

In general, the performance of a radiator may be improved by polishing the surfaces; and if they are fairly smooth and clean, a high polish is required to produce an appreciable change in the property of the radiator, and there is a question whether or not such a method of improvement is practicable.

Report No. 88, entitled "Pressure Drop in Radiator Air Tubes," by S. R. Parsons, Bureau of Standards. This report describes a method for measuring the drop in static pressure in air flowing through a radiator and shows (1) a reason for the discrepancy noted by various observers between head resistance and drop in pressure; (2) a difference in degree of contraction of the jet in entering a circular cell and a square cell; (3) the ratio of internal frictional resistance to total head resistance for two representative types; (4) the effect of smoothness of surface on pressure gradient; and (5) the effects of supplying heat to the radiator on pressure gradient.

The fact that the pressure gradients are found to be approximately proportional to the square of the rate of flow of air appears to indicate turbulent flow, even in the short tubes of the radiator.

It was found that in general the drop in the static pressure in the air stream through a cellular radiator and the pressure gradient in the air tubes are practically proportional to the square of the air flow in a given air density; that the difference between the head resistance per unit area and the fall of static pressure through the air tubes in radiators is apparent rather than real; and that radiators of different types differ widely in the amount of contraction of the jet at entrance. The frictional resistance was found to vary considerably, and in one case to be two-thirds of the head resistance in the type using circular cells and one-half of the head resistance of the radiator type using square cells of approximately the same dimensions.

Report No. 89, entitled "Comparison of Alcogas Aviation Fuel with Export Aviation Gasoline," by V. R. Gage, S. W. Sparrow, and D. R. Harper, of the Bureau of Standards.—Mixtures of gasoline and alcohol when used in internal-combustion engines designed for gasoline have been found to possess the advantage of alcohol in withstanding high compression without "knock," while retaining advantages of gasoline with regard to starting characteristics. Tests of such fuels for maximum power-producing ability and fuel economy

at various rates of consumption are thus of practical importance, with especial reference to high-compression engine development.

Aviation alcogas, prepared by the Industrial Alcohol Co., of Baltimore, Md., for trial by the Navy Department and by the latter submitted to the Bureau of Standards for test, was a mixture apparently of about 40 per cent alcohol, 35 per cent gasoline, 17 per cent benzol, and 8 per cent other ingredients. This is not the alcogas prepared for commercial or passenger car use. The exact composition and methods of manufacture are a trade secret.

The tests made for the Navy Department consisted in a direct comparison, in a 12-cylinder Liberty engine, between alcogas and standard "X" (export grade) aviation gasoline with respect to maximum power obtainable and fuel consumption with the leanest mixture giving maximum power. The tests were made in the altitude laboratory at the Bureau of Standards, where controlled conditions simulate those of any altitude up to 30,000 feet. The speed range covered was from 1,400 to 1,800 revolutions per minute and the altitude range from ground level to 25,000 feet. Two series of comparisons were made, one with 5.6 compression ratio pistons and one with 7.2 compression ratio pistons.

The results of the tests showed the following performance of alcogas in comparison with X gasoline as a standard:

(1) At 5.6 compression the same maximum power production at ground level and a general average of 4 per cent more power at altitude, the maximum difference being about 6 per cent at 6,400 feet and 1,800 revolutions per minute.

(2) At 7.2 compression an average and fairly uniform increase of 4 per cent in power at altitude, no comparative figure for X gasoline at ground level being determined with this compression.

(3) A fuel consumption per brake horsepower of from 10 per cent to 15 per cent more by weight to secure this maximum power at any altitude or speed with either compression ratio. Owing to 12 per cent higher density of alcogas, the fuel consumption in terms of volume per brake horsepower is practically the same as with X gasoline.

(4) Thermal efficiency superior by about 15 per cent. A pound of alcogas contains about 22 per cent less heat units than a pound of gasoline, so that in securing more power with 15 per cent greater weight of fuel it is evident that the available energy of alcogas is more fully utilized than that of gasoline.

Report No. 90, entitled "Comparison of Hecter Fuel with Export Aviation Gasoline," by H. C. Dickinson, V. R. Gage, and S. W. Sparrow, of the Bureau of Standards.—Aviation engine developments for attaining higher power at altitude are following two principal lines, supercharging and increase in compression ratio. For the latter fuels have been demanded which are capable of operating under compressions too high for gasoline. Among the fuels which will operate at compression ratios up to at least 8.0 without preignition or "pinking" is Hecter fuel, whence a careful determination of its performance is of importance.

The Hecter fuel supplied by the Bureau of Mines for use in these tests was a mixture of 30 per cent benzol (C_6H_6) and 70 per cent cyclohexane (C_6H_{12}), having a low freezing point, and distilling from first drop to 90 per cent at nearly a constant temperature, about 20° C. below the average distillation temperature ("mean volatility") of the X gasoline.

This comparison of the performance of the two fuels in an aviation engine was made in the altitude chamber at the Bureau of Standards, duplicating altitude conditions up to about 25,000 feet, except that the temperature of the air entering the carburetor was maintained nearly constant at about 10° C. A Liberty 12-cylinder aviation engine was used, supplied with special pistons giving a compression ration of 7.2 (the compression pressure measured by check-valve gauge was 170 pounds per square inch). Stromberg carburetors were used and were adjusted for each change of fuel, speed, load, and altitude so as to give the maximum possible power with the least fuel for this power. The tests covered a speed range of 1,400 to 1,800 r. p. m.

The results of these experiments show that the power developed by Hecter fuel is the same as that developed by export aviation gasoline at about 1,800 r. p. m. at all altitudes.

At lower speeds differences in the power developed by the fuels become evident. At 1,400 r. p. m. and 25,000 feet Hecter gives a little less power than X gasoline, at 15,000 feet about the same, and at 6,000 feet perhaps 6 per cent more. Comparisons at ground level were omitted to avoid any possibility of damaging the engine by operating with open throttle on gasoline at so high a compression. The fuel consumption per unit power based on weight, not volume, averaged more than 10 per cent greater with Hecter than with X gasoline, considering all conditions. The thermal efficiency of the engine when using Hecter is less than when using gasoline, particularly at higher speeds, a generalization of the difference for all altitudes and speeds being 8 per cent. The general deduction from these facts is that more Hecter is exhausted unburnt. Undoubtedly Hecter can withstand high compression pressures and temperatures without preignition. This characteristic was proved by operating the engine (compression ratio 7.2) with full throttle at 1,500 r. p. m. on the ground, carburetor air temperature 42° C. (107.6° F.) and jacket water temperature, leaving engine, at 90° C. (194° F.). No signs of preignition or "pinking" were noted.

Report No. 91, "Nomenclature for Aeronautics," by the National Advisory Committee for Aeronautics.—This nomenclature and list of symbols were approved by the executive committee of the National Advisory Committee for Aeronautics, for publication as a technical report, on April 1, 1920, on recommendation of the subcommittee on aerodynamics.

The purpose of the committee in the preparation and publication of this report is to secure uniformity in the official documents of the Government and, as far as possible, in technical and other commercial publications. This report supersedes all previous publications of the committee on this subject.

The subcommittee on aerodynamics had charge of the preparation of the report. It was materially assisted by the Interdepartmental Conference on Aeronautical Nomenclature and Symbols, organized by the executive committee, with the approval of the War and Navy Departments, for the purpose of giving adequate representation to the divisions of the Army Air Service and to the bureaus of the Navy Department most concerned. The first meeting of the interdepartmental conference was held on October 23, 1919, and the second meeting on January 15, 1920, at which meeting this report was unanimously approved and recommended to the subcommittee on aerodynamics, with the reservation that stability terms and power plant terms be given further and special consideration.

The stability terms were accordingly referred for special consideration to Messrs. E. B. Wilson, J. C. Hunsaker, A. F. Zahm, E. P. Warner, and H. Bateman, and the power plant terms were referred to the subcommittee on power plants for aircraft. The complete report was adopted by the subcommittee on aerodynamics on March 8, 1920, and recommended to the executive committee for approval and publication.

Report No. 92, entitled "Analysis of Wing Truss Stresses," by E. P. Warner and Roy G. Miller, Langley Memorial Aeronautical Laboratory. Airplane wing trusses are generally designed to contain redundant members (stagger wires and external drag wires) which, according to common practice, are not taken into account in calculations, so as to simplify the stress analysis by rendering the structure statically determinate. A more accurate method, in which the redundancies are included, involves a solution by means of Castigliano's method of least work.

For the purpose of demonstrating the practical application of the method of least work the stresses for several cases of loading were worked out for a structure similar to that of the Curtiss JN-4H.

Case I was taken as the condition of velocity of 100 miles per hour combined with the angle of attack of maximum lift. Case Ia assumed the same loading but neglected the distortion of wooden members in the least-work analysis. So little error was involved in Case Ia (nowhere exceeding 5 per cent of the ultimate strength) that this simplified method was employed for each succeeding case.

Case II assumed a diving speed of 120 miles per hour and an angle of attack of no lift.

Case III was worked out for the conditions imposed by the sand load recommended in N. A. C. A. Technical Note No. 6.

An analysis for each case was also carried through with the stresses corrected for the worst initial tensions which tensiometer readings on service machines indicated were probable.

It was concluded that—

(i) The making of a least-work analysis of a new design for at least one case is thoroughly justified.

(ii) The wooden members may be omitted from consideration in the work equations without causing any serious error.

(iii) The effect of the stagger wires is unimportant when the load is approximately equally distributed between the front and rear trusses. The stagger wires are subjected to their worst stresses while diving.

(iv) Only very rarely are both external drag wires stressed simultaneously.

(v) The initial tensions are almost always excessive, particularly in the stagger wires.

The following recommendations were made:

I. Only one external drag wire should be used on each side of the plane of symmetry and none are required if the front flying wires are strengthened and their attachment to the fuselage carried forward.

II. The long stagger wire is generally the more severely stressed. If a steel tube is used for stagger bracing it should form the short diagonal of the panel.

III. Airplanes should be rigged, whenever possible, by means of a tensiometer and in accordance with a schedule of initial tensions to be provided by the designer.

Report No. 93, entitled "Properties of Aerofoil Sections," by the National Advisory Committee for Aeronautics.—The object of this report is to bring together the investigations of the various aerodynamic laboratories of this country and Europe upon the subject of aerofoils suitable for use as lifting or control surfaces on aircraft. The data have been so arranged as to be of most use to designing engineers and for purposes of general reference. It is the purpose of the committee to publish all existing tests on aerofoil sections, and present this information in a new form.

The absolute system of coefficients has been used, since it is thought by the National Advisory Committee for Aeronautics that this system is the one most suited for international use, and yet is one for which a desired transformation can be easily made. For this puropse a set of transformation constants is included in this report.

Each aerofoil section is given a reference number, and the test data are presented in the form of curves from which the coefficients can be read with sufficient accuracy for design purposes. The dimensions of the profile of each section are given at various stations along the chord in per cent of the chord using as datum the line shown on the curves. The shape of the section is also shown in reasonable accuracy to enable one to more clearly visualize the section under consideration together with its characteristics. The more accurately to obtain the dimensions of the profile of each section, a separate data sheet for each section has been included, which gives an additional decimal place for the greater portion of the ordinates.

The authority for the results here presented is given as the name of the laboratory at which the experiments were conducted with the size of model, wind velocity, and date of test.

Three separate indices are given—a chart index which makes it possible for a designer to select the wing section most suitable for the particular design he is interested in: a group index which is arranged in the same order as the curve sheets; that is, by countries and laboratories at which tests were conducted, each section also being designated by a reference number; and an alphabetical index.

In order that the designer may easily pick out a wing section which is suited to the type of machine on which he is working, four index charts are given which classify the wings according to their aerodynamic and structural properties.

Report No. 94, entitled "The Efficiency of Small Bearings in Instruments of the Type Used in Aircraft," by F. H. Norton, Langley Memorial Aeronautical Laboratory.—This report deals with the construction and properties of bearings and pivots for use in instruments. The static and running friction for both thrust and radial loads was determined for a number of conical pivots and cylindrical and ball bearings. The static rocking friction was also measured for several conical and ball bearings under a heavy load, especially to determine their suitability for use in an N. P. L. type wind tunnel balance. In constructing conical pivots and sockets it was found that the pivots should be hardened and highly polished, preferably with a revolving lap, and that the sockets should be made by punching with a hardened and polished punch. It was found that for a light load the conical pivots give less friction than any other type, and their wearing qualities when hardened are excellent. When the load exceeds about 1,000 grams ball bearings give less friction than pivots and will stand shocks and wear better. Very small ball bearings are unsatisfactory because the proportional accuracy of the balls and races is not high enough to insure smooth running. For rocking pivots under heavy loads it was found that a ball-and-socket bearing, consisting of a hemispherical socket and a sphere of smaller diameter concentric with it with a row of small balls resting between the two, was superior to a pivot resting in a socket. It was found that vibration such as occurs in an airplane will greatly reduce the static friction of a pivot or bearing, in some cases to as little as one-twentieth of its static value.

Report No. 95, entitled "Diagrams of Airplane Stability," by H. Bateman, California Institute of Technology.—In this report a study is made of the effect on longitudinal and lateral oscillations of an airplane of simultaneous variations in two resistance derivatives while the remainder of the derivatives are constant. The results are represented by diagrams in which the two variable resistance derivatives are used as coordinates, and curves are plotted along which the modulus of decay of a long oscillation has a constant value. The same type of analysis is also carried out for the stability of the parachute. For longitudinal stability it is concluded that a decrease in η is unfavorable to stability, but it may be offset by a variation in the other derivatives. The effect of a spring flap is discussed that will change ζ from its usual value of 0 to either a positive or negative quantity. It is found that a positive value of ζ is unfavorable to stability. It is also found that an increase in the value of ξ is unfavorable to stability, and that, if ξ is made positive, the time of damping is decreased. In lateral stability it is found that the greater the value of ξ the greater is the effect on the damping of a change in w, and that an increase in w decreases the time of damping, but does not greatly alter the period. When $\xi=0$ an increase in η decreases the time of damping and increases the period, but when ξ is positive the effect seems to be reversed. An increase in z widens the gap between the curves $t=a$ constant, and to greatly increase the period when $\xi=1$ and $\xi=2$. The chief effect of a decrease in η seems to be a slight change in curvature of the curve $t=a$ constant. In discussing the stability of a helicopter it is concluded that the gyroscopic effect on stability will be greater than in the case of the airplane.

Report No. 96, entitled "Statical Longitudinal Stability of Airplanes," by E. P. Warner, Langley Memorial Aeronautical Laboratory.—This report, which is a continuation of the "Preliminary Report on Free Flight Testing" (No. 70), presents a detailed theoretical analysis of statical stability with free and locked controls and also the results of many free flight tests on several types of airplanes.

In developing the theory of stability with locked controls an expression for pitching moment is derived in simple terms by considering the total moment as the sum of the moments due to wings and tail surface. This expression, when differentiated with respect to angle of incidence, enables an analysis to be made of the factors contributing to the pitching moment. The effects of slip stream and down wash are also considered and it is concluded that the C. G. location has but slight effect or stability, and that stability is much improved by increasing the efficiency of the tail surfaces, which may be done by using an "inverted" tail plane.

The results of free flight tests with locked controls are discussed at length and it is shown that the agreement between the experimental results and theory is very satisfactory.

The theory of stability with free controls is not amenable to the simple mathematical treatment used in the case of locked controls, but a clear statement of the conditions enables several conclusions to be drawn, one of which is that the fixed tail surfaces should be much larger than the movable surfaces.

The discussion of flight tests with free controls covers the effect of C. G. position, tail setting, and slip stream on the JN-4H and gives an analysis of the curves of forces on control stick for the VE-7, U. S. A. C-11, and Martin transport.

Report No. 97, "General Theory of the Steady Motion of an Airplane," by George de Bothezat, Engineering Division, Air Service of the Army.—The writer points out briefly the history of the method proposed for the study of the steady motion of an airplane, which is different from other methods now used. M. Paul Painlevé has shown how convenient the drag-lift curve was for the study of airplane steady motion. His treatment of this subject can be found in "La Technique Aeronautique," No. 1, January 1, 1910. In the author's book "Etude de la Stabilité l'Aeroplane," Paris, 1911, he has added to the drag-lift curve the curve called the "speed curve" which permits a direct checking of the speed of the airplane under all flying conditions. But the speed curve was plotted in the same quadrant as the drag-lift curve. Later, with the progressive development of aeronautical science, and with the continually increasing knowledge concerning engines and propellers, the author was brought to add the three other quadrants to the original quadrant, and thus was obtained the steady motion chart which is described in detail in this report.

This chart therefore permits one to read directly for a given airplane its horizontal speed at any altitude, its rate of climb at any altitude, its apparent inclination to the horizon at any moment, its ceiling, its propeller thrust, revolutions, efficiency, and power absorbed, that is, the complete set of quantities involved in the subject, and to follow the variations of all these quantities both for variable altitude and for variable throttle. The chart also permits one to follow the variation of all of the above quantities in flight as a function of the lift coefficient and of the speed.

The author also discusses in this report the interaction of the airplane and propeller through the slipstream and the question of the properties of the engine-propeller system and its dependence upon the properties of the engine considered alone and of the propeller considered alone will be found treated here in the general manner demanded by actual aeronautical engineering practice. There is also a discussion of the question of a standard atmosphere.

In Part IV the general theory of the steady motion of an airplane is developed, and after the basic equations have been established and the methods to be used described a general survey of the properties of airplane steady motion is given. A detailed discussion of climbing phenomena will be found and the general formulas established for the rate of climb and time of climb, which quantities under the simplest assumptions appear as hyperbolic functions of the ceiling. It is also shown as a consequence under what conditions one can drive the law of linear variation of the rate of climb with altitude as observed practically.

Report No. 98, entitled "Design of Wind Tunnels and Wind Tunnel Propellers," by E. P. Warner and F. H. Norton, Langley Memorial Aeronautical Laboratory.—This report is a continuation of National Advisory Committee for Aeronautics' Report No. 73. The variations in velocity and direction of the wind stream were studied by means of a recording air speed meter and a recording yawmeter. The work was carried on both in a 1-foot diameter model tunnel and in a 5-foot full-sized tunnel, and wherever possible comparison was made between them. It was found that placing radial vanes directly before the propeller in the exit cone increased the efficiency of the tunnel to a considerable extent and also gave a steadier flow. The placing of a honeycomb at the mouth of the experimental portion was of the greatest aid in straightening the air flow, but at the same time this decreased the efficiency of the tunnel. Several types of diffuser were tried in the return air stream, but only a slight improvement resulted in the steadiness of the flow. Some experiments were tried on the effect of the shape of exit cone and it was found that a straight cone in all cases gave the highest

efficiency. The effect of placing a closed room about the model tunnel of the same proportional size as the building on the 5-foot tunnel decreased the speed for the same power 14½ per cent. Several spinners were placed about the propeller in the model tunnel in the hope that they would give increased efficiency and a steadier flow, but in no case was there any improvement.

Report No. 99, entitled "Accelerations in Flight," by F. H. Norton and E. T. Allen, Langley Memorial Aeronautical Laboratory.—This report deals with the accelerations obtained in flight on various airplanes at Langley Field. The instrument used in these tests was a recording accelerometer of a new type designed by the technical staff of the National Advisory Committee for Aeronautics. The instrument consists of a flat steel spring supported rigidly at one end so that the free end may be deflected by its own weight from its neutral position by any acceleration acting at right angles to the plane of the spring. This deflection is measured by a very light tilting mirror caused to rotate by the deflection of the spring, which reflected the beam of light onto a moving film. The motion of the spring is damped by a thin aluminum vane which rotates with the spring between the poles of an electric magnet. Records were taken on landings and take-offs, in loops, spins, spirals, and rolls. It was found that the loading in a fairly heavy landing reached a maximum of 5 g., in a loop it reached a maximum of about 3.7 g., in a spin a maximum of about 3 g., while in a roll it attained the value of 4.2 g., showing that this maneuver puts a greater strain on the airplane than any other. A JN-4H was pulled as suddenly as possible out of a dive at 50, 60, 70, and 80 miles an hour. The records show that the time elapsed between pulling the stick back and reaching the maximum acceleration was independent of the air speed and amounted to about 0.9 seconds. These accelerations are slightly lower than the theoretical accelerations that would be obtained if the airplane were suddenly turned to the angle of maximum lift. It was also found that an airplane had a certain definite period of vibration which could be excited by the engine, but which was not at all dependent upon it, as the vibrations were nearly as evident when the airplane was gliding with a dead stick. This period of vibration appeared to be inversely proportional to the weight of the airplane. It is concluded from these tests that in no reasonable stunting would the load in flying ever exceed a factor of four and one-half times the normal stress.

Report No. 100, entitled "Accelerometer Design," by E. P. Warner and F. H. Norton, Langley Memorial Aeronautical Laboratory.—In connection with the development of an accelerometer for measuring the loads on airplanes in free flight a study of the theory of such instruments has been made, and the results of this study are summarized in this report. Portion of the analysis deals particularly with the sources of error and with the limitations placed on the location of the instrument in the airplane. The discussion of the dynamics of the accelerometer includes a study of its theoretical motions and of the way in which they are affected by the natural period of vibration and by the damping, together with a report of some experiments on the effect of forced vibrations on the record.

Report No. 101, entitled "The Calculated Performance of Airplanes Equipped with Supercharging Engines," by E. C. Kemble, Harvard University.—In Part I of this report are presented the theoretical performance curves of an airplane engine equipped with a supercharging compressor. In predicting the gross power of a supercharging engine, the writer uses temperature and pressure correction factors based on experiments made at the Bureau of Standards (cf. Reports No. 45 and 46 of this committee). Means for estimating the temperature rise in the compressor are outlined. Since the compressor will be designed for a definite normal pressure ratio, the gross power output under normal conditions is easily computed when the intake temperature is known. In the case of a gear-driven compressor, the net power is obtained by subtracting the power absorbed by the compressor from the gross power. For use in determining the size and power absorption of the compressor needed in a given case, a formula for the variation of the volumetric efficiency of the motor with intake temperature and exhaust pressure is derived.

In calculating the power output of an engine fitted with a turbine-driven compressor, it is assumed that the back pressure created by the turbine is equal to the increase in the carbureter pressure produced by the blower.

A graphical method is outlined whereby performance curves for either type of engine-compressor unit at all speeds and altitudes may be laid out with the aid of assumed compressor characteristics. Comparative performance curves for a Liberty engine operating with a turbine-driven compressor, a gear-driven compressor, and without supercharging, are derived in an illustrative calculation. A discussion of the relative fuel consumption of supercharging and nonsupercharging engines when the carbureter is adjusted for maximum power is appended.

Part 2 of this report presents an estimation of the performance curves of an airplane fitter with a supercharging engine. If the heat leak from gas turbine and exhaust pipes to the water jackets is prevented, and the cooling system is kept under constant pressure, no additional radiator equipment should be required when a supercharging compressor is fitted to an airplane engine.

A method of estimating airplane performance at altitudes with the aid of curves for the "reduced" thrust horsepower available and required, is developed. This method simplifies the graphs of the thrust horsepower required at altitudes, and is particularly useful in comparing the performance of airplanes of different sizes, wing loadings, and propelling plant characteristics, which have the same lift and drift coefficients. Two methods for drawing curves of the thrust horsepower available with a variable pitch propeller are indicated.

In an illustrative example horizontal flight speed and maximum climbing speed curves are worked out for the Lepere two-seater fighter when equipped with supercharging and nonsupercharging engines, and with both fixed and variable pitch propellers. These are supplemented by altitude-time curves at maximum climbing rate and curves showing the relative fuel economy (i. e., relative distance traversed per pound of fuel) in horizontal flight with the engine wide open at all altitudes.

A supercharging installation suitable for commercial use is described, and it is shown that with the aid of the compressor a great saving in fuel and a considerable increase in carrying capacity can be effected simultaneously.

In an appendix the writer derives a theoretical formula for the correction of the thrust coefficient of an airscrew to offset the added resistance of the airplane due to the slip-stream effect.

Report No. 102, entitled "Performance of Liberty-12 Engine," by S. W. Sparrow and H. S. White, Bureau of Standards.—In cooperation with the Engineering Division of the Air Service of the United States Army, a Liberty-12 engine has been tested at the Bureau of Standards. The program of tests was planned to yield that information considered most important in determining the value of the engine for aviation. Full power runs were made at the ground, at 25,000 feet, and at several intermediate altitudes. To determine the mechanical efficiency of the engine, friction horsepower was measured at the ground and at 15,000 feet. As a basis for predicting engine performance with a propeller, a series of tests was made in which the dynamometer load and engine throttle were adjusted at each speed to simulate the engine load which would be imposed at that speed by a propeller operating under normal full load at 1,700 r. p. m.

Among the quantities calculated from the test measurements are: Brake horsepower; brake mean effective pressure; fuel consumption; mixture ratio; mechanical, thermal, and volumetric efficiency; and the percentage of the heat in the fuel appearing in the jacket water and in the exhaust. Jacket water temperature, oil temperature, manifold pressure, etc., are recorded to show the conditions under which the test was made.

The provision on the carburetor for adjusting the mixture ratio is shown to be inadequate at altitudes above 15,000 feet. Improving the mechanical efficiency of the engine and making such changes as will prevent the present decrease of volumetric efficiency with increase of altitude are suggested as two possibilities of improving the altitude performance of the engine.

Report No. 103, entitled "Performance of Hispano-Suiza 300-horsepower Engine," by S. W. Sparrow and H. S. White, Bureau of Standards.—A 300-horsepower Hispano-Suiza engine has been tested at the Bureau of Standards. The program of tests was planned in cooperation with the Engineering Division of the Air Service of the United States Army and was intended primarily to determine the characteristic performance of the engine at various altitudes. The engine was operated at the ground, at 25,000 feet, and at intermediate altitudes, both at full load and at loads similar to those that would be imposed upon the engine at various speeds by a propeller whose normal full-load speed was 1,800 r. p. m. Friction horsepower also was determined in order that the mechanical efficiency of the engine might be calculated.

From the test data there were computed the brake horsepower; brake mean effective pressure; specific fuel consumption; mixture ratio; jacket loss; exhaust loss; and thermal, mechanical, and volumetric efficiencies. A record of jacket water temperatures, oil temperatures, manifold pressures, etc., shows the conditions under which the test was made.

A brake horsepower of 352 was obtained at 2,200 r. p. m. and a maximum brake mean effective pressure of 128 pounds per square inch at about 1,600 r. p. m. The mechanical efficiency varied from 88 per cent to 83 per cent from speeds of 1,400 r. p. m. to 2,200 r. p. m., while the brake thermal efficiency, based on the lower calorific value of the fuel, was about 26 per cent over this speed range. At 1,800 r. p. m. and at an air density of 0.040 pounds per cubic foot the brake horsepower was about 42 per cent and the indicated horsepower about 47 per cent of that at the ground.

Report No. 104, entitled "Torsion of Wing Trusses at Diving Speeds," by Roy G. Miller, Langley Memorial Aeronautical Laboratory.—It is the purpose of this report to indicate what effect the distortion of a typical loaded wing truss will have upon the load distribution. The case of high angle of incidence may be dismissed immediately from consideration as the loads on the front and rear trusses are nearly balanced, and consequently there will be little angular distortion. A given angular distortion will have the maximum effect upon load distribution in the region of the angle of no-lift, because the slope of the lift curve is highest here, and it is here that the greatest angular distortion will occur, because the load on the front truss acts downward while the load on the rear truss acts upward.

The RAF–15 aerofoil was chosen as most typical of present-day wing sections and serves for an illustrative example. This was combined with the JN–4 wing truss, a biplane with overhanging upper wings. Starting with the assumption of a loading for a rigid structure, the wing truss and the deflections were calculated. The assumption of loading for the second trial was based upon the deflections as determined by the first trial. After several approximations it was possible to compute accurately the angular distortion at each panel point.

It was found that no great angular distortion occurred at panel points where there was adequate stagger bracing but that it was considerable at the tip of the overhanging portion of the upper wing. In conclusion, it may be said that it is not worth the added complication to correct the load distribution on the conventional biplane for wing truss distortion but that it would be highly advisable in the case of a monoplane, where the wires of the lift truss make an acute angle with the spars and where there can be nothing to take the place of stagger bracing. It would also be advisable in the case of the internally braced wing where the relative deflection is likely to be high.

Report No. 105, entitled "Angles of Attack and Air Speeds During Maneuvers," by E. P. Warner and F. H. Norton, Langley Memorial Aeronautical Laboratory.—In seeking further information as to the nature of maneuvers and as to the maneuverability characteristics of airplanes, continuous measurements of the angles of attack and air speeds at several points along the wings have been made during spins and loops. Very striking results have been obtained with reference to the rolling velocity and the distribution of load in spins and the variation of the angle of attack in loops, a surprisingly large range of angle being experienced during slow loops. This work is fully described in Technical Report No. 105.

Report No. 106, entitled "Turbulence in the Air Tubes of Radiators for Aircraft Engines," by S. R. Parsons, Bureau of Standards.—The existence of turbulent flow in the air passages of aircraft radiators and of variations in character or degree of turbulence with different types of construction is shown by the following experimental evidence:

(1) Pressure gradients along the air tubes are roughly proportional to the 1.7 power of the speed, which is characteristic of turbulent flow in long circular tubes of the same diameters.

(2) The surface cooling coefficients of radiators vary widely (0.002 to 0.007) when expressed as heat dissipated per unit time, per unit cooling surface, per unit temperature difference between air and water, and at a given average linear speed through the tubes.

(3) A fine wire electrically heated shows different cooling coefficients in the air tubes of different radiators.

(4) Temperature gradients in the air tubes are of the form characteristic of turbulent flow and fail to show sudden breaks such as might indicate a dividing line between regions of viscous and of turbulent flow.

The use of special devices for increasing turbulence may increase the heat transfer per unit surface for a given flow of air through the radiator but such practice decreases that flow for a given speed of flight and increases head resistance. At very low flying speeds, or in cases where the radiator is mounted in the nose of the fuselage, turbulence devices may sometimes be used to advantage, but every type known to the writer is detrimental to the general performance of the radiator at high speeds.

Report No. 107, entitled "A High-Speed Engine Pressure Indicator of the Balanced Diaphragm Type," by H. C. Dickinson and F. B. Newell, Bureau of Standards.—This report describes a pressure-measuring device especially adapted for use in mapping indicator diagrams of high-speed internal-combustion engines. The cards are obtained by a point-to-point method giving the average of a large number of engine cycles. The principle involved is the balancing of the engine cylinder pressure against a measured pressure on the opposite side of a metal diaphragm of negligible stiffness. In its application as an engine indicator the phase of the engine cycle to which a pressure measurement corresponds is selected by a timing device. The report discusses briefly the errors which must be avoided in the development of an indicator for light high-speed engines, where vibration is serious, and outlines the principles underlying the design of this instrument in order to be free of such errors. A detailed description of the instrument and accessories follows, together with operating directions. Specimen indicator diagrams are appended. The indicator has been used successfully at speeds up to 2,600 r. p. m., the highest speed engine available for trial. Its sensitivity is approximately that of a standard 6-inch dial gauge of the Bourbon tube type.

Report No. 108, entitled "Some Factors of Airplane Engine Performance," by Victor R. Gage, Bureau of Standards.—This report was prepared for the National Advisory Committee for Aeronautics and is based upon an analysis of a large number of airplane-engine tests made at the Bureau of Standards. This report contains the results of a search for fundamental relations between many variables of engine operation.

The data used came from over 100 groups of tests made upon several engines, primarily for military information. The types of engines were the Liberty 12 and three models of the Hispano-Suiza. The tests were made in the altitude chamber, where conditions simulated altitudes up to about 30,000 feet, with engine speeds ranging from 1,200 to 2,200 r. p. m. The compression ratios of the different engines ranged from under 5 to over 8 to 1. The data taken on the tests were exceptionally complete, including variations of pressure and temperature, besides the brake and friction torques, rates of fuel and air consumption, the jacket and exhaust heat losses.

With the Liberty engine operating at from 500 to 2,000 r. p. m. and with the Hispano-Suiza 300 horsepower operating from 1,400 to 2,200 r. p. m. it is found that the friction torque

increases approximately as a linear function of engine speed at a given air density, and approximately as a linear function of density at a constant speed. This means that the friction horsepower increases approximately as the square of the speed. Actually the relation of torque and speed is such that the friction horsepower increases with speed raised to a power between the first and second, this power increasing with speed, approaching the square. The relation depends upon the engine design, the speed, and density of the air. Any statements as to the distribution of the friction losses are based upon incomplete evidence; the indications are, however, that the pumping losses are about half of the total friction.

There is no doubt that for a given process of combustion and at a constant speed the engine power is directly proportional to the weight of charge supplied; in other words, proportional to the charge density at the beginning of compression. As a consequence, if operating conditions are sensibly constant except for altitude, the engine power will be closely proportional to the air density. The volumetric efficiency increases with increase of air temperature at constant pressure, so that power does not decrease as fast as the air density when the temperature is raised, due to changes in vaporization and heat transfer.

Report No. 109, entitled "Experimental Research on Air Propellers, IV," by W. F. Durand and E. P. Lesley, Leland Stanford University.—This report is a continuation of a report on the same subject published in the fifth annual report. The research was conducted in the aerodynamical laboratory of Leland Stanford Junior University, and the report prepared under the direction of Dr. W. F. Durand and Prof. E. P. Lesley. The report states the results of investigations made upon numerous propeller models at the request of the subcommittee on aerodynamics, and contains valuable data to those interested in the design of air propellers. The discussion accompanying the report is necessarily somewhat brief, as the report is to be a part of the general report which will include a review of all the propeller investigations that have been conducted at Leland Stanford Junior University. This general report will be ready for publication with the seventh annual report of the committee.

Report No. 110, entitled "The Altitude Effect on Air Speed Indicators," by M. D. Hersey, Franklin L. Hunt, and Herbert N. Eaton, of the Bureau of Standards.—The object of this paper is to present the results of a theoretical and experimental study of the effect, on the performance of air speed indicators, of the different atmospheric conditions experienced at various altitudes. This matter has ordinarily been handled in a very simple way by following the PV^2 law and therefore correcting the observed reading of the air speed indicator by assuming the differential pressure developed to be directly proportional to the density and independent of any other physical property of the air.

The failure of certain types of air speed indicator to follow the simple PV^2 law at very low or very high speeds is already well recognized. For example, in the case of the Pitot tube, more accurate results can be deduced at high speeds by considering, in addition to the density of the air, its adiabatic compression. Again, in the case of the Venturi, a departure from the law of proportionality to the square of the speed has been recognized also at low speeds; consequently, as shown in this paper, a corresponding departure must be expected at a sufficiently high altitude, even without going to the lower speeds.

Thermodynamic formulas are available indicating the probable performance of Pitot tubes at high speeds where compressibility has to be considered, but all efforts which have thus far been made to arrive at a sufficiently complete formula for the Venturi tube by purely deductive reasoning have proven impracticable, on account of the difficulty of treating viscosity and turbulence. An adequate method of analysis for such problems has, however, been found in dimensional reasoning, for by this means the minimum number of experimental data needed for providing an absolutely complete inductive rather than deductive solution can be determined. In this way in the present paper the general form of the relation expressing the pressure generated in terms of the size of the instrument, its velocity through the air, and the density, viscosity, and elasticity of the medium has been derived. It is shown how all of the last five physical quantities can be reduced to only two independent variables, one involving the viscosity and the other the elasticity or compressibility of the air. Thus the equation be-

comes simply that of an ordinary surface in three coordinates. By such a surface or family of curves the experimental observations can be represented graphically.

The experiments reported all relate to Venturi tubes. They include water channel experiments to determine the degree of dynamical similarity attainable between air and water and to discover whether compressibility has to be taken into account; observations in a wind stream at reduced pressure, i. e., a vacuum wind tunnel, to determine the effect of density and viscosity; airplane observations as a practical check on the laboratory results; also ordinary wind tunnel tests.

The results by these various experimental methods are all in qualitative agreement and have been reduced to a common basis for quantitative comparison by the graphical method outlined above. At the conclusion of the paper a chart is given containing the most probable results available to date for the relative performance characteristics of five well-known types of air-speed nozzle both American and foreign, involving Venturi tube combinations. This chart provides the necessary experimental basis for computing altitude corrections.

This investigation is primarily of importance in connection with low speed or high altitude flight, for the altitude correction under the conditions of high-speed flight near sea level is sufficiently well given for most instruments by the simple PV^2 law.

LIST OF TECHNICAL NOTES ISSUED BY NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS DURING THE PAST YEAR.

No. 1. Notes on Longitudinal Stability and Balance. By Edward P. Warner.
2. Airplane Performance as Influenced by the Use of a Supercharged Engine. By George de Bothezat.
3. Notes on the Theory of the Accelerometer. By Edward P. Warner.
4. The Problem of the Helicopter. By Edward P. Warner.
5. Relation of Rib Spacing to Stress in Wing Planes. By A. F. Zahm.
6. Static Testing and Proposed Standard Specifications. By Edward P. Warner.
7. Notes on the Design of Supercharged and Overdimensioned Aircraft Motors. Translated from Technische Berichte, Vol. III, sec. 5. By Schwager.
8. Duralumin. By E. Unger and E. Schmidt. Translated from Technische Berichte, Vol. III, sec. 6.
9. Abstract of Theory of Lifting Surfaces, Part I. By L. Prandtl, 1918. Prepared by Paris office.
10. Abstract of Theory of Lifting Surfaces, Part II. By L. Prandtl, 1919. Prepared by Paris office.
11. The Problem of the Turbo-Compressor. By René Devillers.
12. Recent Efforts and Experiments in the Construction of Aviation Engines. Translated from Technische Berichte, Vol. III, sec. 5. By Schwager.
13. Soaring Flight in Guinea. By P. Idrac.
14. Increase in Maximum Pressures Produced by Preignition in Internal Combustion Engines. By S. W. Sparrow.
15. Tests of the Daimler D IVa Engine at a High Altitude Test Bench. By W. G. Noack. Translated from Technische Berichte, Vol. III, sec. 1.
16. Experience with Geared Propeller Drives for Aviation Engines. By Kutzbach.
17. Italian and French Experiments on Wind Tunnels. By W. K. Knight.
18. The Dynamometer Hub. By W. Stieber. Translated from Technische Berichte, Vol. III, sec. 6.
19. The Steadiness Factor in Engine Sets. By W. Margoulis.
20. Notes on Specifications for French Airplane Competitions. By W. Margoulis.
21. Drag or Negative Traction of Geared-Down Supporting Propellers in the Downward Vertical Glide of a Helicopter. By A. Toussaint.

RESEARCH PROGRAM AND ESTIMATES.

For the year 1922 the National Advisory Committee for Aeronautics has planned a comprehensive program of aeronautical research, which in the opinion of the committee covers the most important features that have to do with the further development of power plants for aircraft, aerodynamical improvements in aircraft, and new materials for aircraft.

Aerodynamical research.—The program of aerodynamical research is to be carried out with a view to the successful development of an airplane incorporating an internally braced wing structure, in order to eliminate practically all of the structural resistance, a factor which greatly handicaps the performance of the present type of airplane. The program includes research on the aerodynamical characteristics of airplane structures, including wings and fuselage, that are applicable to all-metal and internally braced types of construction. The research is to be carried on both in the wind tunnel and in free flight, so that by an examination of the performance of full-scale airplanes using the new type of construction as compared with the performance indicated by experiments on models in the wind tunnel further knowledge of the scale factor between model and full-scale performance may be obtained. The aerodynamical research program also contains provision for the determination of the variation of loading along the span for the thick wing sections which are likely to be used in all-metal, internally braced designs. This research will supply data very much needed in the design of these new types of machines, which, because of their structural permanency, their high load carrying capacity, and their high maximum speed, will undoubtedly be the airplanes of the future.

In free flight testing the program provides for the complete performance tests of airplanes to determine accurately the aerodynamical characteristics of the airplane, especially with reference to their stability, so that information may be obtained that will aid the designing engineer to predict accurately the performance of a new airplane. The outstanding feature of the airplane over other means of transportation is the high speed at which it is possible to fly, and it is appreciated that if the airplane is to become an important factor in transportation the efficient operating speed of the airplane must be materially increased. The program contemplates a research with the aim of obtaining those characteristics of an airplane that make for high operating speeds and a large speed range. The research program also includes the development of new instruments to aid in air navigation and new instruments to be used in the accurate performance tests of airplanes. Experimental research for the determination of the pressure distribution over the surface of an airplane and its controls is provided for, and likewise the distribution over the surface of an airship and its controlling members. The data obtained will enable the engineer to design the structure more accurately, as he will know definitely the forces acting on the structure under all operating conditions.

The committee asks for the sum of $215,000 to carry out research work in connection with aerodynamics.

Materials research.—The subcommittee on materials for aircraft has brought to the attention of the main committee the fact that since the armistice all-metal construction of airplanes has received the careful attention of airplane manufacturers in Europe, with the result that apparently successful models have been constructed. The war was fought with machines constructed of wood, which from many standpoints is most unsatisfactory, especially from a constructional point of view. Wood has a nonhomogeneous structure, is uncertain in strength and weight, warps and cracks, and weakens rapidly when exposed to moisture. The advantages of using metal construction for airplanes are apparent, as the metal does not splinter, is more homogeneous, and the properties of the material are much better known and can be relied upon. Metal also can be produced in large quantities, and it is felt that in the future all large airplanes must necessarily be constructed of metal. The program for the year 1922 provides for experimental research in the development of light alloys of aluminum and magnesium base for use in aircraft. Aluminum alloys are now being produced that have the same physical properties as mild steel, with one-third the weight, and the program further pro-

vides for the development of light alloys, especially in connection with their heat treatment and method of fabrication. The physical properties of light alloys are not accurately known, especially with reference to the fatigue resistance properties of the material, and the program provides for experimental research covering this phase of the problem.

This research will be carried on under the direction of the subcommittee on materials for aircraft and will be conducted by private corporations, and also by the Navy and the Army Air Service interested in the production of the material.

Aeronautic Power Plant Research.—The future progress of civil and military aviation is so fundamentally dependent upon the development of highly reliable and economical power plants that the problems connected with increasing these features of aircraft power-plant operation are considered to be among the most important at present demanding the attention of aeronautical research laboratories. The capital investment, maintenance charges, and fuel cost are all very high in the case of the present aircraft engine and must be materially lowered before the cost of power can be reduced to figures which will make possible the extensive development of commercial and pleasure aviation. The shortage and high cost of aviation gasoline, as well as the complication and relative unreliability of the carburetion and ignition systems, emphatically indicate the necessity for the development of an engine which will operate by direct hydraulic injection of low-grade fuel, with compression sufficiently high to ensure automatic ignition. The committee feel that the early development of an engine of this type is one of the most important technical problems involved in the growth of commercial aviation in this country, and the research program for the coming year provides for extensive work in this field.

Perhaps the next most important power-plant problem is the elimination of the water-cooling system, it being at present agreed that the added complication, weight, and head resistance of the indirect cooling system are to be considered as fundamentally unecessary handicaps to power-plant performance and reliability, and that these must ultimately be overcome. Although considerable research has been conducted upon the direct cooling of engine cylinders, the results must be considered as merely indicative, and much yet remains to be done before the successful and economical direct cooling of aircraft engines will become possible, especially with cylinders of large dimensions and high specific power output. The program covers the requirements in this problem in a comprehensive manner.

The perfecting of supercharges, or other means for securing the maximum power output of aviation engines at all altitudes, is considered to be one of the vital problems, and the program provides for a continuation of the research examination of the many possibilities offered in this field. All of those applications of commercial, military, and pleasure aviation which depend upon high speed for their successful fulfillment can only reach their complete development through flying at high altitudes with power plants capable of maintaining a high percentage of their maximum power output and equipped with variable pitch or variable characteristic propellers.

The program also contains provision for continuing the performance tests of new types and improved forms of aircraft engines in the altitude chamber; the performance tests of all engine accessories such as carburetors, ignition appliances, lubrication appliances, and cooling appliances, including radiators in the form of complete units and also sample cores; and the study of other interesting developments of important engine details, such as pistons, valves, etc.

The estimate of the committee to cover the necessary power-plant research for the fiscal year of 1922 is $131,600.

In connection with the research on powerplant and aerodynamic problems at Langley Field the committee maintains shop facilities at the Langley Memorial Aeronautical Laboratory, the estimated expenses of which for the year 1922 are $58,666.

Summary.—The committee's estimates for the prosecution of the programs of aerodynamical research, materials research, and aeronautic power plant research, as outlined above, total $405,266. To this should be added, under the committee on publications and intelligence,

the work of the Office of Aeronautical Intelligence in the collection, classification, and dissemination of scientific and technical reports and data on aeronautics, requiring the sum of $59,800, and for the general administration of the Washington office with its present personnel, the sum of $24,540, making the total estimates for the fiscal year 1922, $489,906. The appropriation for the fiscal year 1920 was $175,000, and for the present fiscal year the appropriation was increased to $200,000. The continuous prosecution of a well organized plan of scientific research is an essential factor in the development of the science of aeronautics, and the increased estimates of the committee for the fiscal year 1922 are made necessary by the increasing relative importance of scientific research in the general scheme of a national aviation policy, as outlined in the closing section of this report.

FINANCIAL REPORT.

The appropriation for the National Advisory Committee for Aeronautics for the fiscal year 1920, as carried in the sundry civil appropriation act approved July 19, 1919, was $175,000, under which the committee reports expenditures and obligations during the year amounting to $174,296.75, itemized as follows:

Salaries (including engineering staff)	$65, 299. 58
Wages	23, 559. 51
Equipment	12, 539. 57
Supplies	19, 493. 22
Transportation and communication	1, 281. 42
Travel	6, 066. 58
Special investigations and reports	40, 716. 39
Construction of buildings	5, 340. 50
Total	174, 296. 75

CONCLUSION.

A NATIONAL AVIATION POLICY.

Aviation activities during the war were concentrated on the development and production of military aircraft. The selection of the landing fields that were established was necessarily guided by military considerations. The close of the war found us with an aeronautic industry at the stage of quantity production, a large amount of aircraft material on hand, a large number of trained flyers, and a few scattered landing fields. In brief, all this constituted the national inheritance from the investment of hundreds of millions of dollars for the hurried development of military aviation during the war. In the two years that have elapsed since the armistice a good proportion of the aircraft material has become obsolete. A majority of the technical personnel and trained flyers have returned to civil life and to pursuits not connected with aviation. The great aircraft industry has almost disappeared, and some of the landing fields have been surrendered. Those that have been retained really represent one of the most valuable physical assets salvaged from our aircraft expenditures.

As a nation we must seek to realize clearly the lessons of the war and to profit by them, Our efforts in the development of a military air force and the organization of an aircraft industry during the war were remarkable accomplishments in themselves, but the handicap of a negligible industry at the outbreak of the war and the general lack of technical knowledge were too great to be satisfactorily overcome in a short time, regardless of the money available. It is now our clear duty to take to heart the lessons and mistakes of the war period and to shape a national aviation policy that will be productive of the greatest possible structural development consistent with prudent economy.

The Government agencies actively concerned with the use of aviation at the present time are the Army Air Service, the Naval Air Service, and the Postal Air Service. Other agencies such as the Geological Survey, the Coast and Geodetic Survey, the Forest Service, etc., have more or less need for the use of aircraft in their work. The National Advisory Committee for Aeronautics is concerned not so much with the promotion of the uses of aviation as with the

scientific study of the problems involved and the technical development of the art for the benefit of governmental agencies and of the public generally, but the committee believes that the use of aircraft by the various governmental agencies should be encouraged where its efficient use is practicable; also that the general development of aviation for all purposes should be encouraged by the National Government. The faithful performance of our national duties in these respects becomes compelling from considerations of wise military preparedness.

In time of war aviation will probably be the first arm of offense and defense to come into action. For this there must be an established industry and a trained and active air service. Aerial supremacy at the outset of hostilities would be a tremendous military advantage. Ultimate victory would unquestionably incline to the side that could establish and maintain supremacy in the air. Huge expenditures of money in time of danger and frantic efforts to train personnel and to develop hastily an aircraft industry from almost nothing will not do. There must be wise preparedness; there must be in healthy existence at least a nucleus of an industry capable of adequate expansion; there must exist civil and commercial aeronautical activities in all parts of the country which would be the main support of the industry in time of peace. In pure self-defense the Government must encourage the development of commercial aviation. The alternative proposition is the creation and maintenance of a powerful standing military air service relatively self-reliant in time of war. We can not, however, afford the expense which such a policy would entail, and there would be no advantage in time of peace from such expenditures comparable in any way to the advantages to be gained from the support of civil aviation. We should maintain an active air service in time of peace, which should possess inherent strength and be something more than a mere nucleus for expansion in time of war. In the final analysis, however, we must depend upon civil aviation to furnish a military reserve force. The remarkable accomplishments of our Motor Transport Service during the war were only made possible by the healthy condition of our automobile industry. The problem is to place our aircraft industry in a healthy condition, and to do this we must enter without delay upon a sane, sound policy for the development of civil aviation. The relative cost of fostering an organized plan to develop commercial aviation would be much less than the waste that would inevitably result from unprepared entry into war. Aside from military considerations, the fostering of commercial aviation would in time yield adequate returns in itself in the form of promoting and strengthening our means of transportation, advancing the progress of civilization, and increasing the national wealth.

Aviation is a distinct advance in civilization given to the world by America. The importance of the development of aviation from a military standpoint was not fully appreciated before the war, with the consequent lack of encouragement of the development of the art. The handicap of years of comparative inactivity has not yet been overcome. We can not afford to repeat the mistakes of the past. We can not go backward, but must go forward with the intelligent development of aviation in all its branches.

Aviation is still in its infancy; its possibilities, while unknown, appeal to the imagination. The forced development during the war and some of the experimental development since have not been based upon scientific research and sound scientific principles that make for substantial progress. Technical training is necessary, including education in advanced aeronautical engineering, so is the actual training of a large body of men in the technique of the care and operation of aircraft. Broadly speaking, scientific research, technical training, and commercial aviation constitute, or should constitute, the backbone of a national policy.

Reducing to definite form the steps which in the opinion of the National Advisory Committee for Aeronautics are wise and timely, the committee, after careful consideration of all the facts within its knowledge, submits the following specific recommendations:

First. That legislation be enacted providing for Federal regulation of commercial air navigation, licensing of pilots, aircraft, landing fields, etc. At the present time there is no authority of law for any executive agency of the Government to perform such duties. The

committee believes that for the executive administration of these new duties of government there should be established in the Department of Commerce a bureau of aeronautics in charge of a commissioner of air navigation, who should also become a member of the National Advisory Committee for Aeronautics. Acting in cooperation with the War, Navy, and Post Office Departments, the committee has prepared a draft of legislation which appears in full in a preceding section of this report under the heading "Organization of Governmental Activities in Aeronautics," and which it strongly recommends for the immediate consideration of Congress. In this connection the committee recommends also the adoption of a policy of Federal aid to the States in the establishment of landing fields for general use in every State in the Union.

Second. That the Congress authorize an American airplane competition in order to stimulate private endeavor in the development of new and improved designs of aircraft, the competition to be under the direction of the National Advisory Committee for Aeronautics, the entries of the successful competitors to be purchased by the Government at a predetermined and announced figure and made available for the use of the Postal Air Service.

Third. That adequate appropriations be made for the military and naval air services in order to permit the continuous development of these exceedingly important arms of the two services, and to enable them to place orders in such a way as to maintain a nucleus of an aircraft industry capable of sufficient expansion to meet military needs in time of emergency. The committee considers this absolutely essential.

Fourth. That the control of naval activities in aeronautics be centralized under a naval bureau of aeronautics in charge of a director of naval aviation. At the present time responsibility for the development of naval aviation is divided between the Office of Operations and the numerous bureaus of the Navy Department. This basis of organization does not permit full cooperation with the Army Air Service or with other governmental and civil agencies, nor does it, in the opinion of the committee, promote the efficient development of aviation within the Navy.

Fifth. That the Air Mail Service of the Post Office Department be further extended and developed. This service has given the best demonstration of the practicability of the use of aircraft for civil purposes. It has been seriously handicapped by inability to secure suitable airplanes adapted to its work. The question is one of design, which should be handled by the industry. The remedy lies in the development of the industry, which can only be brought about at an early date by the indorsement and prosecution by the Government of a constructive, comprehensive policy.

Sixth. That the Congress approve the program of scientific research in aeronautics formulated by the committee and provide for the enlarged facilities necessary for its prosecution. Continuous scientific research is necessary for the real advancement of the science of aeronautics. The number and importance of problems requiring solution have increased greatly with the general development of aircraft, and the development of airplanes of all-metal construction will require a large increase in the aerodynamic research and engineering experimentation conducted by the committee at the Langley Memorial Aeronautical Laboratory at Langley Field, Va.

Respectfully submitted,

National Advisory Committee for Aeronautics,
Charles D. Walcott, *Chairman.*

REPORT No. 83

WIND TUNNEL STUDIES IN AERODYNAMIC PHENOMENA AT HIGH SPEED

IN THREE PARTS.

By F. W. CALDWELL and E. N. FALES
Engineering Division, Air Service of the Army, McCook Field, Dayton, Ohio

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TIP VORTEX ON MODEL ÆROFOIL.

REPORT No. 83.

WIND TUNNEL STUDIES IN AERODYNAMIC PHENOMENA AT HIGH SPEED.

PART I.

MODEL WIND TUNNEL EXPERIMENTS.

By F. W. Caldwell and E. N. Fales.

McCook Field.

INTRODUCTION.

This report was prepared for publication by the National Advisory Committee for Aeronautics by Messrs. F. W. Caldwell and E. N. Fales of the Engineering Division, Air Service of the Army, McCook Field, Dayton, Ohio, with the approval of Col. T. H. Bane, U. S. Army, Chief of the Engineering Division and Member of the National Advisory Committee for Aeronautics.

A great amount of research and experimental work has been done, and fair success attained, in an effort to place airplane and propeller design upon an empirical basis. One can not, however, fail to be impressed with the lack of data available toward establishing flight phenomena upon a rational basis, such that they may be interpreted in terms of the laws of physics. Almost the whole field of aeronautical experiment and design is based on the law of dynamic similarity.

In practical work it has been necessary to combine the results of model tests with empirical factors which are certainly limited in their application. The writers see no reason whatever for skepticism about the application of the law of dynamic similarity provided we really have similarity. It is certainly insufficient, however, to have geometrical similarity between the solid objects being studied. We must have in addition similarity of the character of the air flow.

Mathematical studies of first importance, which are now classical, on the nature of the flow about an aerofoil have been developed by Helmholtz, Kirchoff, Lord Rayleigh, Lanchester, Prandtl, Kutta, Karman, Greenhill, Lewis, and others. Dr. Georges de Bothezat has put forward some very interesting ideas about the effect of stresses in the fluid on the nature of the air flow, and he has consented to write a note on that subject at the end of this report.

It is Dr. de Bothezat's conception that the type of flow which establishes itself is governed by the stresses set up in the air passing the aerofoil. The unit stresses increase as the velocity rises. It is easy to conceive that a given type of flow is possible only so long as the shearing stress, developed in the fluid, does not exceed a certain magnitude which depends on the value of the viscosity coefficient.

When the stress reaches a certain critical value, two adjacent layers of air begin to slide past each other and the character of the flow is changed. Apparently such a change must bring with it a change of aerofoil characteristics since there is no longer flow similarity. This condition was actually encountered in most of the high speed tests referred to in this paper. The photograph of figure 2 shows the flow at a speed where such a change is incipient.

Experimental investigation of the flow has heretofore been rather unsuccessful because of lack of adequate methods. The writers laid out the design of the McCook Field wind tunnel to investigate the scaling effect due to the high velocity of propeller aerofoils. During the course of the experiments, however, it was found possible to visualize the air flow by means of the method described in Section II. This was made use of to study some changes in flow which affect the characteristics of the aerofoil in a very great degree.

The experiments were made under authority of Col. T. H. Bane, commanding officer of the field.

Acknowledgment is due the Bureau of Standards for furnishing laboratory facilities, and to Messrs. W. G. Gwynn and J. R. Randolph for carrying out the model wind-tunnel experiments. Acknowledgment is also due Messrs. C. P. Grimes, D. A. Dickey, and J. F. Piccard for assistance in carrying out the experiments at McCook Field.

The writers wish particularly, however, to express their appreciation of Prof. Gaetano Lanza, who in 1909–1912 assisted them in the researches which were a preliminary to those recorded in this paper. Prof. Lanza was at that time head of the mechanical engineering department at the Massachusetts Institute of Technology; he was the first active patron of aerodynamics on the staff of that institution, and prior to 1909, had erected a wind tunnel which became a stimulus to the authors' first aeronautical researches; he proposed a 12-foot wind tunnel for such work as early as 1910; and he put the entire shop and laboratory facilities of his department at the authors' disposal, for the wind tunnel, propeller, motor, and airplane material tests which were their major research subjects while students in his department.

OBJECT OF EXPERIMENTS.

The design of a wind tunnel differing from the usual type, especially with regard to large power and speed of flow, involves features whose suitability can not be predicted. After all available information has been secured on full size and model wind tunnels in various parts of the world, there remains much obscurity about the air-flow phenomena. The United States Army wind-tunnel designs, proposed as an item in the aircraft program of the recent war, have been developed toward the end of securing superior efficiency and steadiness of air flow. But it has been found that the conventional types fall short of the mark, and offer no precedent for many of the improvements conceived. Original experimentation has therefore become desirable for the purpose of comparing conventional and novel wind-tunnel arrangements.

DESCRIPTION OF APPARATUS.

The apparatus consisted of a one-twelfth scale model wind tunnel somewhat similar to the National Physical Laboratory type, but susceptible of a large number of variations. The "flue" was a sheet-iron cylinder 45 inches long by 8 inches diameter, provided at one end with a wooden intake bell and at the other with a 16° cone. For special tests the cone was modified at the large end either by constricting the discharge area, by affixing a 12-inch cylinder, by prolonging the cone, or by adding a large "vacuum chamber." Three types of intake were also tested.

The power plant consisted of a high-speed direct-current electric motor coupled to a long shaft, the whole properly mounted to avoid serious obstruction to the propeller discharge. The entire arrangement was adapted to the study of fan-cone arrangements, traverses in the flue, noise, etc. Five propellers were used comprising one 12-inch, 2-blade; two 12-inch, 4-blade; one 18-inch, 2-blade; one 18-inch, 4-blade. (Figs. 3, 4, 5.)

The conventional Pitot and impact tube apparatus was used for determining velocity, static, and dynamic heads. Flow lines were observed carefully by the smoke method and by the fine thread method.

METHOD OF TESTS.

The object of the tests was to compare specific arrangements of wind tunnels and to eliminate those proving inferior. It was therefore sufficient to establish relative rather than absolute efficiency in each test. While electrical power input was measured, it was found unnecessary to use the $\frac{\text{output}}{\text{input}}$ ratio as a means of interpreting the results; and the larger number of tests were made, not in terms of power, but in terms of a performance factor dependent on the effective pitch of the propeller blades.

A given set of tests, all made with the same fan, were then directly comparable with the others. The results have been represented graphically, in such a way as to show by diagrams the arrangement, the serial number, and the performance factor obtained in each test.

FIG. 1.—VIEW OF THE FLOW TAKEN FROM ABOVE THE MODEL.

FIG. 2.—THE DISTURBED AIR BEHIND THE CENTER SUPPORT HAS HERE SHIFTED INTO THE LEFT.

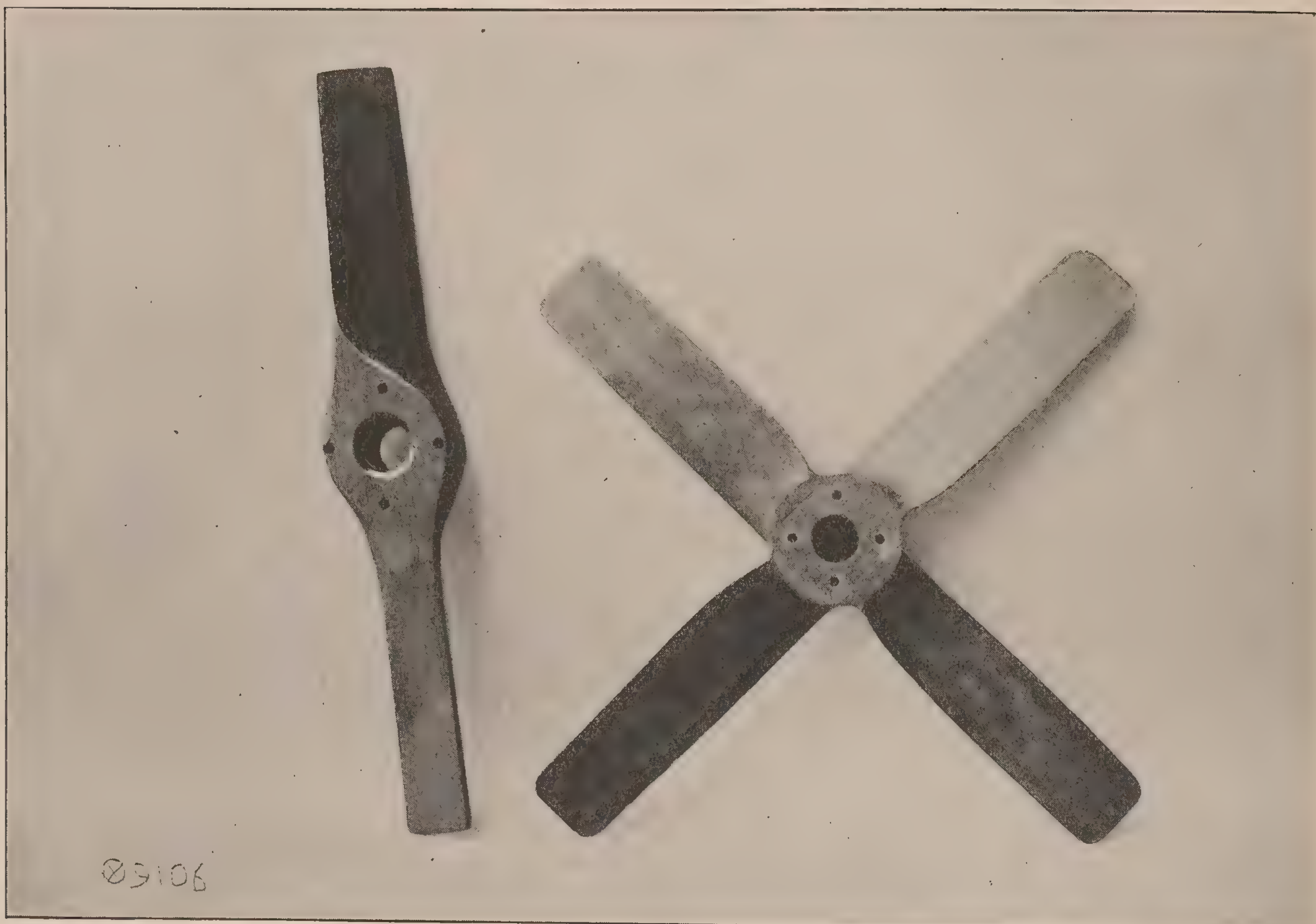


FIG. 5.

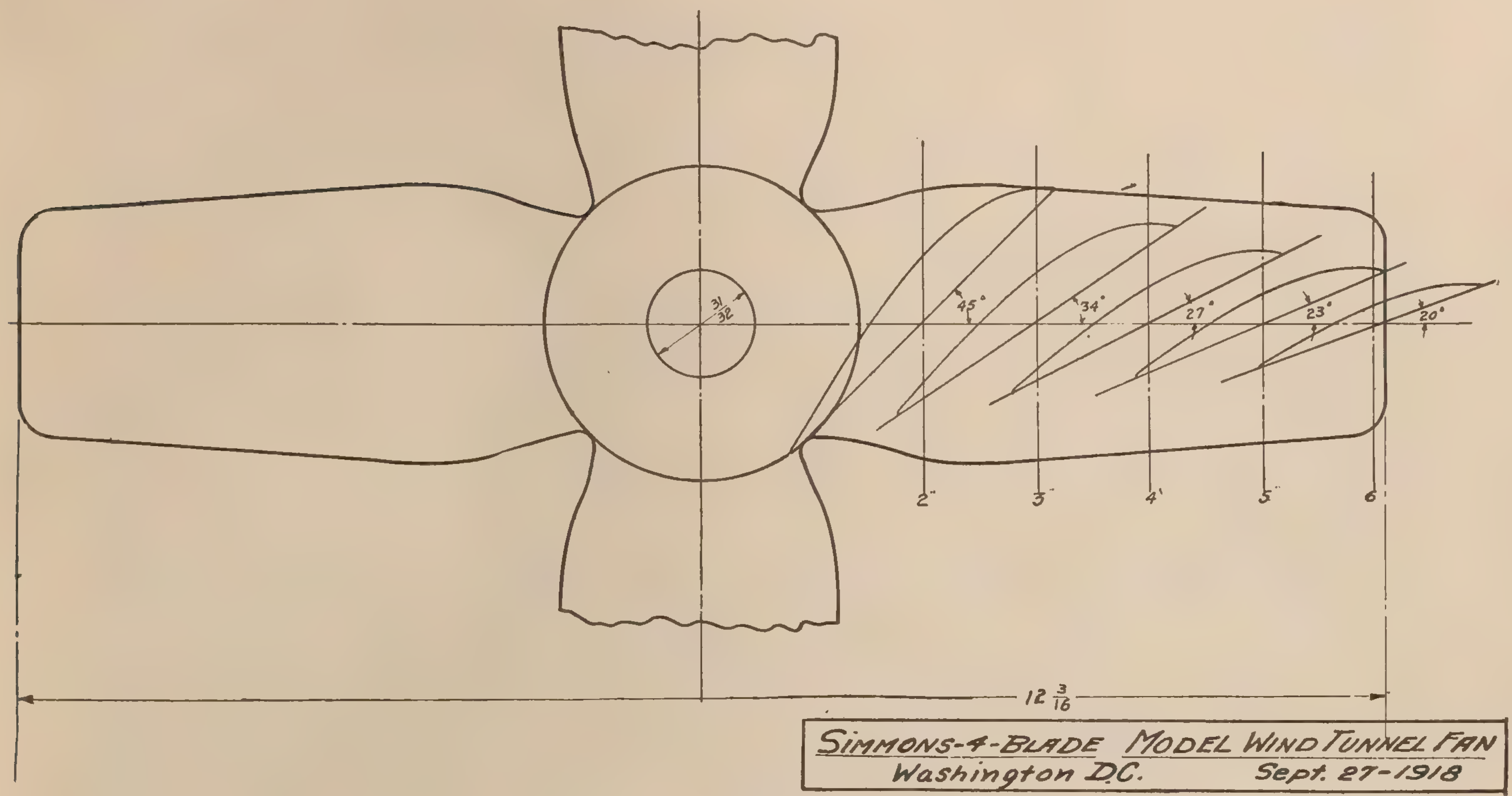

31/32
45°
34°
27°
23°
20°
2"
3"
4"
5"
6"
12 3/16
SIMMONS-4-BLADE MODEL WIND TUNNEL FAN
Washington D.C. Sept. 27-1918

FIG. 3.

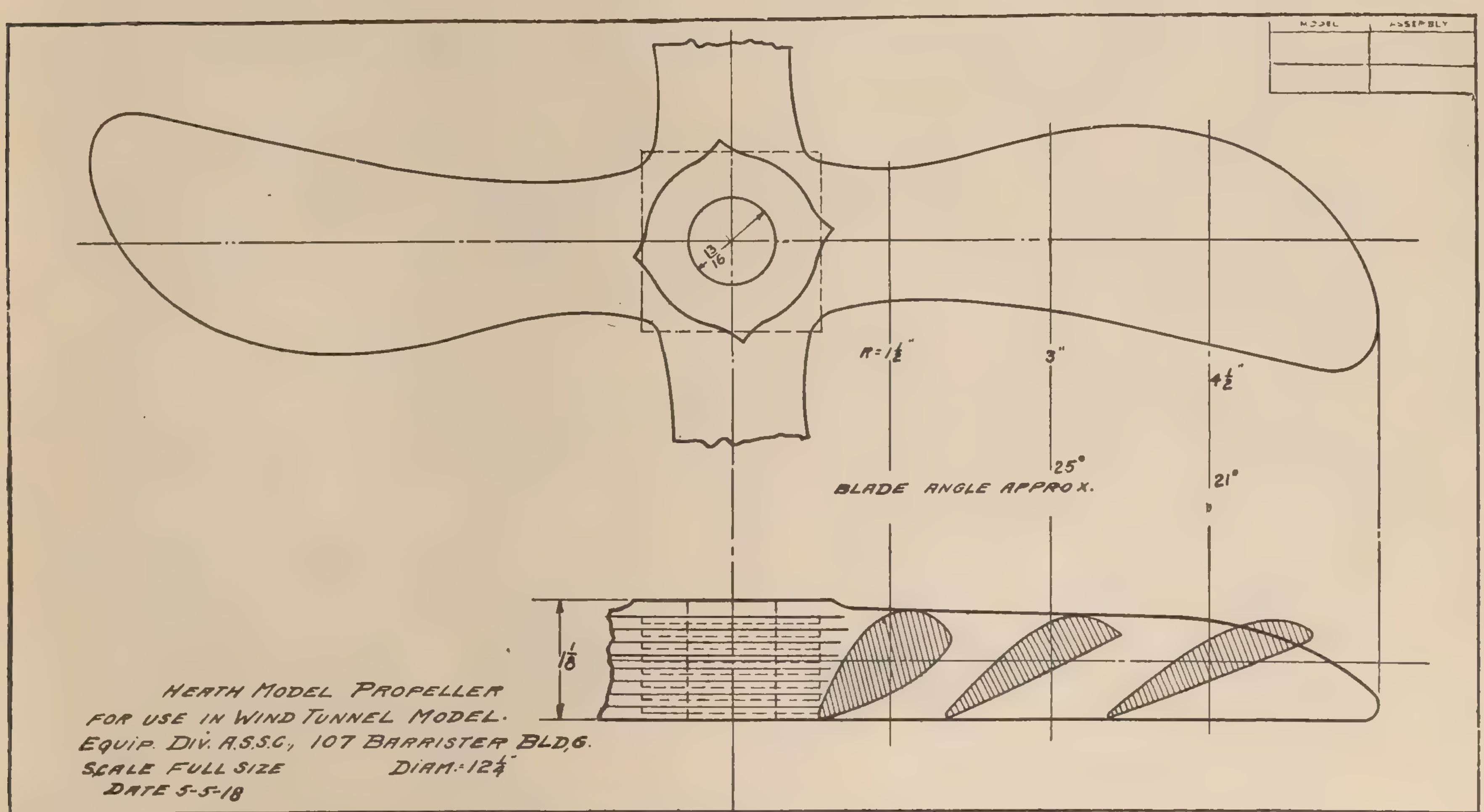

13/16
R=1½"
3"
4½"
25°
21°
BLADE ANGLE APPROX.
1/8
HEATH MODEL PROPELLER
FOR USE IN WIND TUNNEL MODEL.
EQUIP. DIV. A.S.S.C., 107 BARRISTER BLDG.
SCALE FULL SIZE DIAM.=12¼"
DATE 5-5-18

FIG. 4.

DISCUSSION OF RESULTS.

Few of the tests tabulated in this report are intended to represent complete wind-tunnel arrangements; the tests apply rather to investigation of detached details of air flow, noise, efficiency, etc., according as the latter were found to require study. Negative results are observed in some cases, but they are nevertheless included in this report in order that they may be available for study by those interested. No reference has been made in this report to the model wind-tunnel studies of Crocco, Costanzi, Eiffel, and others. The data obtained by these earlier experimenters has been accepted, without corroboration, as a valuable preliminary to the further developments recorded in this report.

From analysis of the data herewith presented various obscure phenomena of air flow have been better explained than heretofore and have become properly subject to interpretation for the design and operation of aerodynamic wind tunnels. Inasmuch as the graphs are so arranged as to afford their own analysis, no general comment upon the results is needed. A summary is given, however, of the studies for which the experiments were chiefly originated, namely, analysis of air flow through the cone and fan.

Angle of Cone.

The discharge cone used in a wind tunnel is, of course, analagous to the expansion cone of a venturi tube. If the angle of divergence of the walls is too great the cone will not fill, as shown by the graphs of the Second Series, runs Nos. 14, 43, 45, 58, 58a, and by traverse runs 66d and 66j. When the cone angle is too large the velocity recorded in it is greater than called for by the cross-sectional area at the point of observation. The area of uniform traverse in the cone is about the same as in the flue. In an arrangement of the "pushing" variety, the flow in the cone tends to hug the walls better, due to the centrifugal whirling of the air. (Refer to the Second Series, tests Nos. 52, 53, and 54.)

Whirls in the Wind Tunnel.

The whirling noticed in the wide-angle cone of the conventional type of wind tunnel may be analyzed from a study of the flow diagrams. In general, as indicated in a typical arrangement (Series 2, test No. 33), the air from the flue expands on an angle considerably less than the cone angle; and there is a whirling ring of air 1½ inches thick at the large end, separating the axial flow from the cone walls. The whirl ring takes up its motion from the fan tips and may communicate a spiral motion to the air flow at the center of the ring. Downstream from the fan there is a negative flow at the hub. Change of fan design, aside from cowling, does not prohibit this condition. (Refer to First Series, tests Nos. 20 and 21.) Even the blanking off of the hub does not completely eliminate the whirl. (See Series 2, test 47.) Regarding the matter of whirling, see also Series 2, tests 48, 51, 54, 58, 58a, 65.

Study of Air Flow in the Vacuum Chamber.

To better study the natural deceleration of an unconfined air blast, the cone was replaced by a spacious chamber called the "vacuum chamber." It was large enough to permit entrance by the observer, who then had before him all the cone phenomena save those dependent upon the presence of the cone walls.

The tests demonstrate the loss due to eddies wherever a flowing stream of air is surrounded not by container but by other air. Compare Second Series, tests Nos. 12 and 13, wherein the shorter flow produces the higher performance factor. Compare also Nos. 12 and 14, wherein the arrangements are identical except that No. 14 interposes a container between the discharge cone of air flow and the relatively inert air of the vacuum chamber. Compare similarly Second Series, tests Nos. 3, 11, 30, and 36.

Equilibrium in Conical Discharge.

It is seen from reference to First Series, tests Nos. 7 and 9, that the air blast leaving the wind tunnel with the cone temporarily in place can be maintained even after the cone is removed; although this conical air blast can not be brought into existence without the use of such a dis-

charge cone. (See also First Series, runs Nos. 22, 23, and 24.) The natural establishment of a virtual cone of air flow is shown in Series 2, test No. 59b, where the fan can be moved 5 inches away from the discharge opening and yet maintain fair flow of air. The existence of the virtual cone is shown by comparing the Second Series, tests Nos. 11 and 20; in No. 11 the air has not expanded sufficiently for the diameter of the fan, with the result that it is discharged with a higher radial than axial component. (See also No. 13, where the effect is exaggerated.)

The Fan.

A comparison of different fan housings shows the fan to be properly located when its plane of rotation is upstream from the large end of the cone, as in Series 2, test No. 31. (Compare Nos. 31 to 34, also 43, 45, 46.) Should a cylindrical housing be used, as in the National Physical Laboratory type, refer to test 55 for indication of the best position for the fan. A cone terminating in a restricted discharge opening is not superior to one terminating in a cylinder. (Compare Series 1, tests 56 and 57.)

Tip clearance was studied in Series 2, test 45, by moving the fan axially to successive positions. It is seen that an increase of clearance from $\frac{1}{48}$ of the radius to $\frac{4}{48}$ of the radius drops the performance factor 12 per cent. In the particular arrangement shown, the clearance should be a minimum; however, it does not follow that generous clearances are always detrimental, for certain other arrangements give very good results with comparatively large clearance

Tests 47 to 51, Second Series, deal with the matter of cores or cowling applied to the fan.

Various investigations were made of the discharge blast from the fan. The parasite whirl occurs as a ring separating the positive and negative air flow; or it occurs some times outside the positive air flow. (See Series 2, tests 31, 70, 43, and 59-a). The effect of confining the fan blast in a Venturi discharge is shown for various arrangements in Series 2, tests 28, 29, 30, 36, 36-a, 37-a, 37-b, 60, 61, 62. Further investigation of confined flow of the fan blast is made by setting up the fan as a "pusher" at the wind-tunnel intake. There results, of course, a helical flow in the flue which is excessive, as would be expected, only when flow through the fan is inefficient. (See Series 1, tests 2, 3, 4, 38, 39.) The characteristics of the pushing arrangement are shown in Series 1, tests 1 to 5, and Series 2, tests 52 to 54, and traverses 66-d and 66-e.

One of the important objects of these experiments was to investigate the question of noise made by the fan. (See Series 1, tests Nos. 9, 57, 59, 60, and Series 2, tests Nos. 55 and 59–b.)

RELATION BETWEEN VELOCITY AND EFFICIENCY.

Reference to run 71 indicates a practically constant efficiency from 0 to 85 ft./sec. Beyond 85 ft./sec. the efficiency decreases, instead of increasing, as would be expected.

EFFECT OF INTAKE BELL ON VELOCITY TRAVERSE OF VALUE.

Without an intake bell a "vena contracta" forms at the flue entrance, resulting in loss of energy due to eddy formation, and inferior velocity distribution in the flue. (See run 73.) The inflowing air can, in the early stages of its acceleration, follow a curve of sharp radius, and flows in with radial component as well as axially toward the intake. The inflow is analagous to the inflow to a static propeller; by means of experiment the proper shape of intake bell may be determined as well as the minimum clearance between intake and end wall of the building in which the wind tunnel is housed. The minimum practicable diameter may also be thus determined.

Runs 72 to 76 establish the comparative merits of various intakes, through a study of the velocity traverse in the flue afforded by each type. These tests measure the drop in energy between the room and the point of observation; they give an adequate indication of the velocity traverse, since the static head traverse is substantially uniform. (Run 77.)

Of the types tested the order of preference appears in sequence as follows (see Tests 72–76):

1. Grimes intake.
2. N.P.L. 7-foot intake } Practically the same.
3. N.P.L. 4-foot intake }
4. No intake.

Velocity Traverse Studies:

In these tests (72 to 76) it is apparent that the velocity traverse is better close to the intake than at a distance. For the Grimes' type intake uniformity exists over 0.93 diameter near the intake, as against 0.77 diameter 20 inches downstream. Run 77 shows that the flow near the entrance is substantially axial at all points of the traverse.

Two conclusions are obvious:

1. The model to be tested should be located as close to the intake as is permitted by local distortions in the airflow lines about the model.

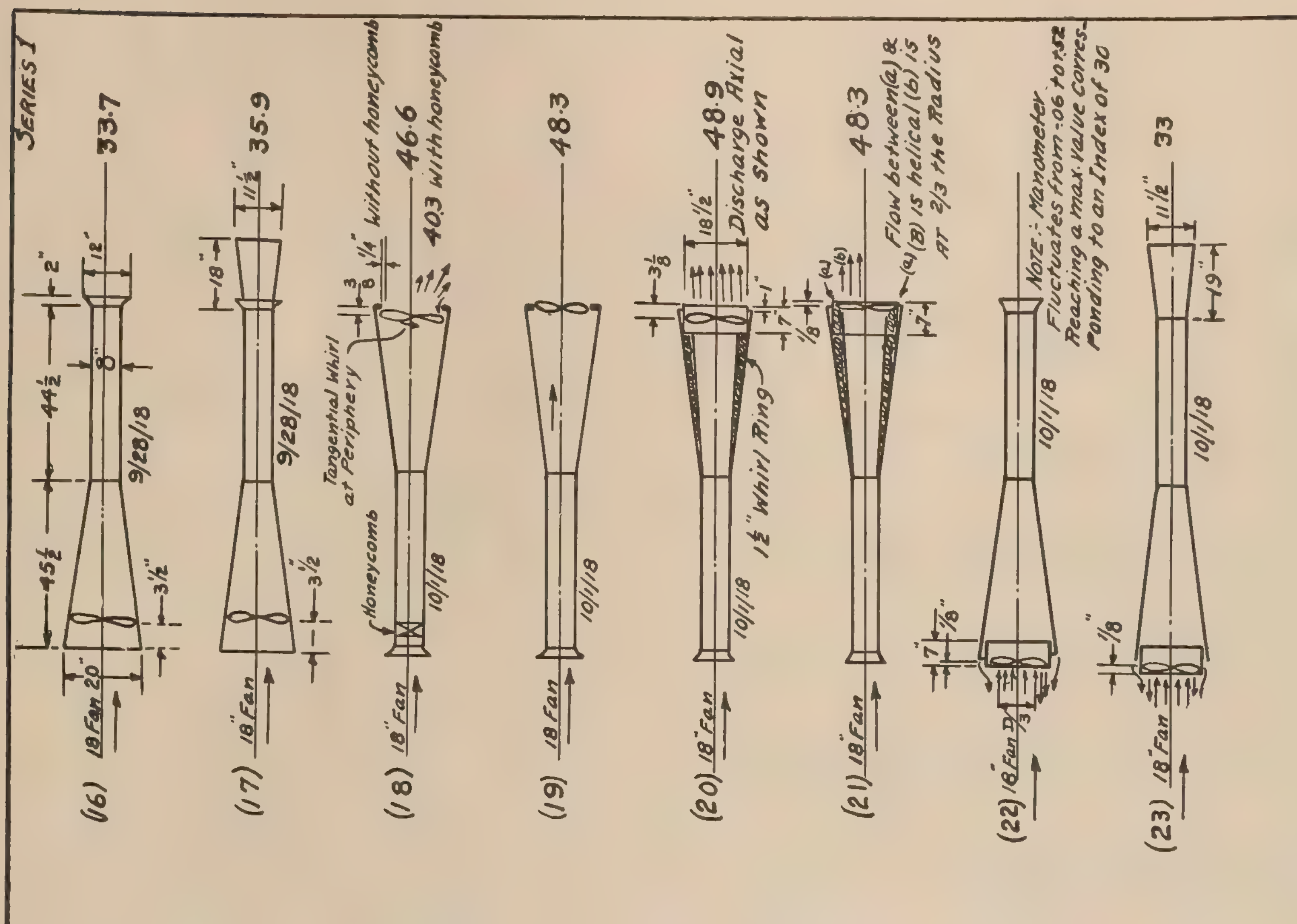
2. A honeycomb is not necessary for securing parallelism of the airflow filaments where proper intake is installed, and is to be considered only as a device for decreasing velocity fluctuations in the flue. Regarding velocity fluctuations or "pulsations" in a wind tunnel, the writers have not succeeded in solving the problem to their satisfaction. After taking account of variations in mean flue velocity due to inconstancy of the pressure causing the flow, it has been found that serious "pulsations" remain. This has been investigated by reading the differential dynamic pressure between any two points in the flue cross section; the readings indicate that in general pulsations at the two points are not simultaneous, but are of local extent. The pulsations are greatest near the flue walls, where they depend on the vortex phenomena associated with skin friction.

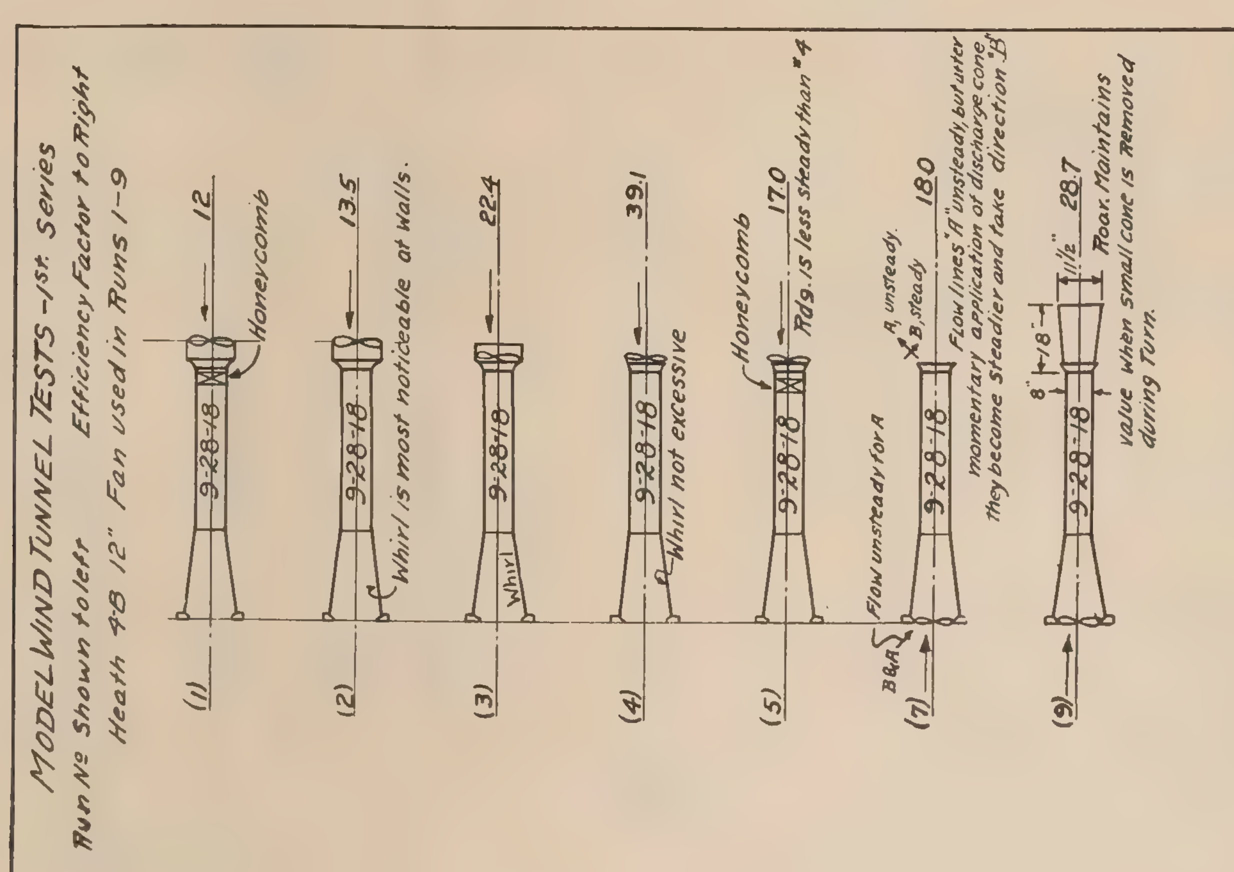
In further discussion of the relation between the honeycomb and the pulsations, it may be said that the pulsations in the McCook Field wind tunnel have been reduced from 15 per cent down to 3 per cent without the use of the conventional honeycomb. It has been assumed that the stream has a tendency to take up a spiral whirl, due to the radial flow at intake. Such whirl can be largely prevented by plane axial vanes located a considerable distance downstream. Hence the installation of the four-blade "straightener" 4 feet long and 4 feet downstream from the model.

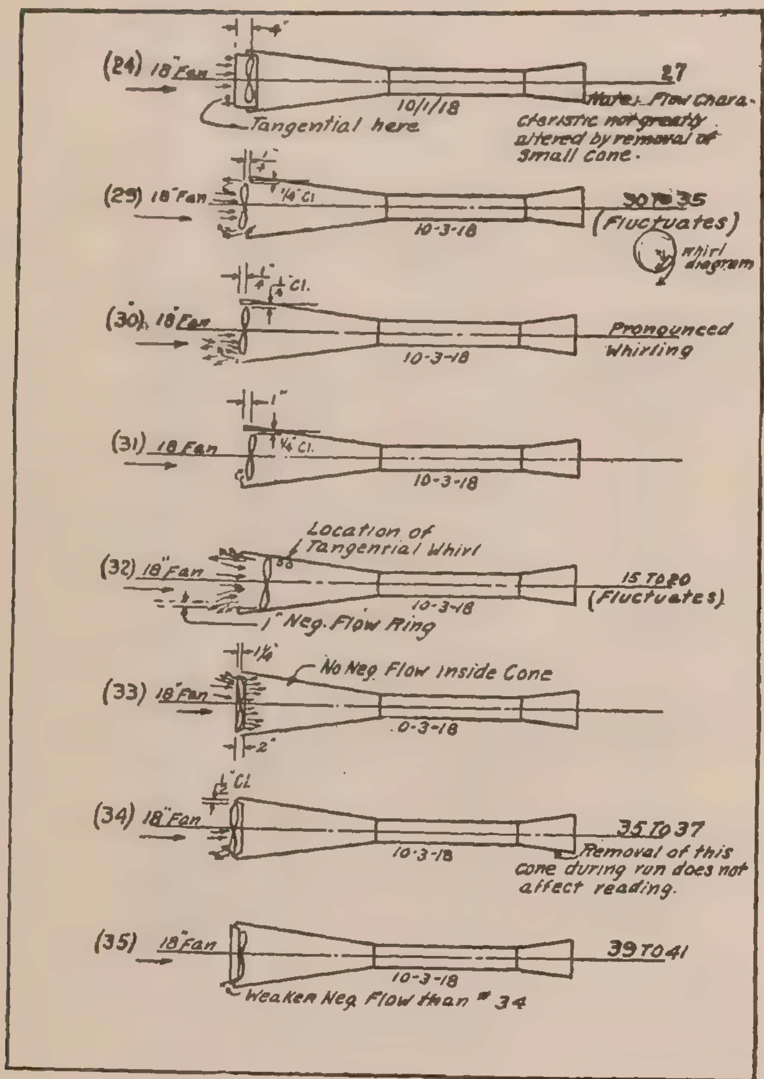
EFFECT OF FAN-CONE ARRANGEMENT ON VELOCITY TRAVERSE IN FLUE.

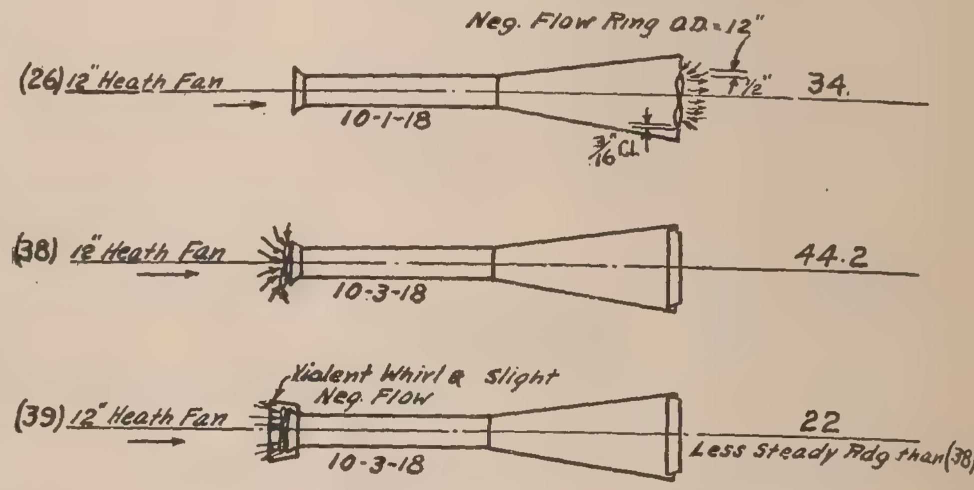
The velocity traverses afford valuable data on the relative merits of the fan-cone arrangement. Refer to Series 2 tests 72, 66–c, 66–f, 66–g, which are made with the same intake bell and flue and without honeycomb. Of these four tests the last three are made with fan propulsion; test 72 is made with a special arrangement, the cone being removed, and the flue being sealed into the intake of the 14-inch McCook Field wind tunnel so that its flow is in effect created by a uniform "suction" rather than by a propeller.

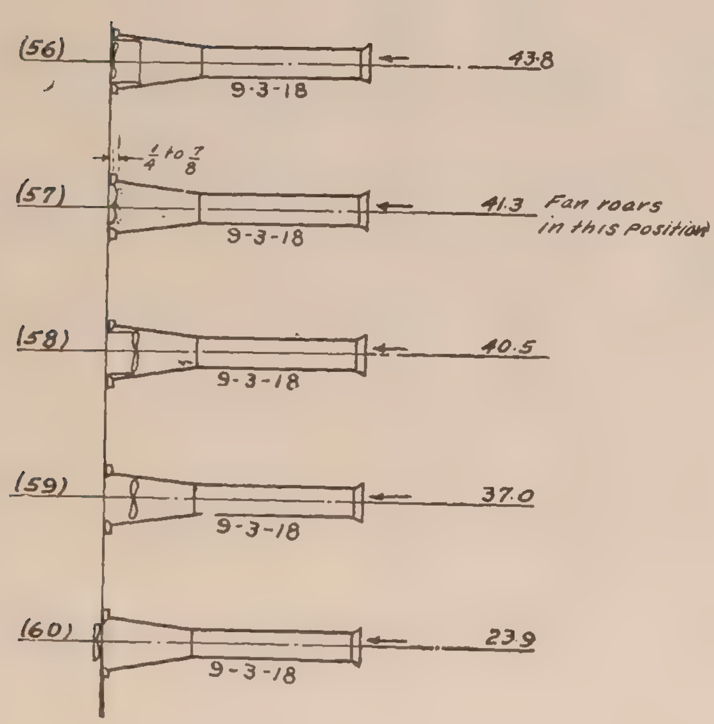
Comparison of these tests shows that the traverse uniformity varies with the fan-cone arrangement; it is best when the suction is uniform and inferior when the suction is less uniform. Of course the static head traverse at the fan is not uniform, and from tests 66–c, 66–f, and 66–g it appears that this nonuniformity of static head traverse affects the portion of the flue occupied by the model.

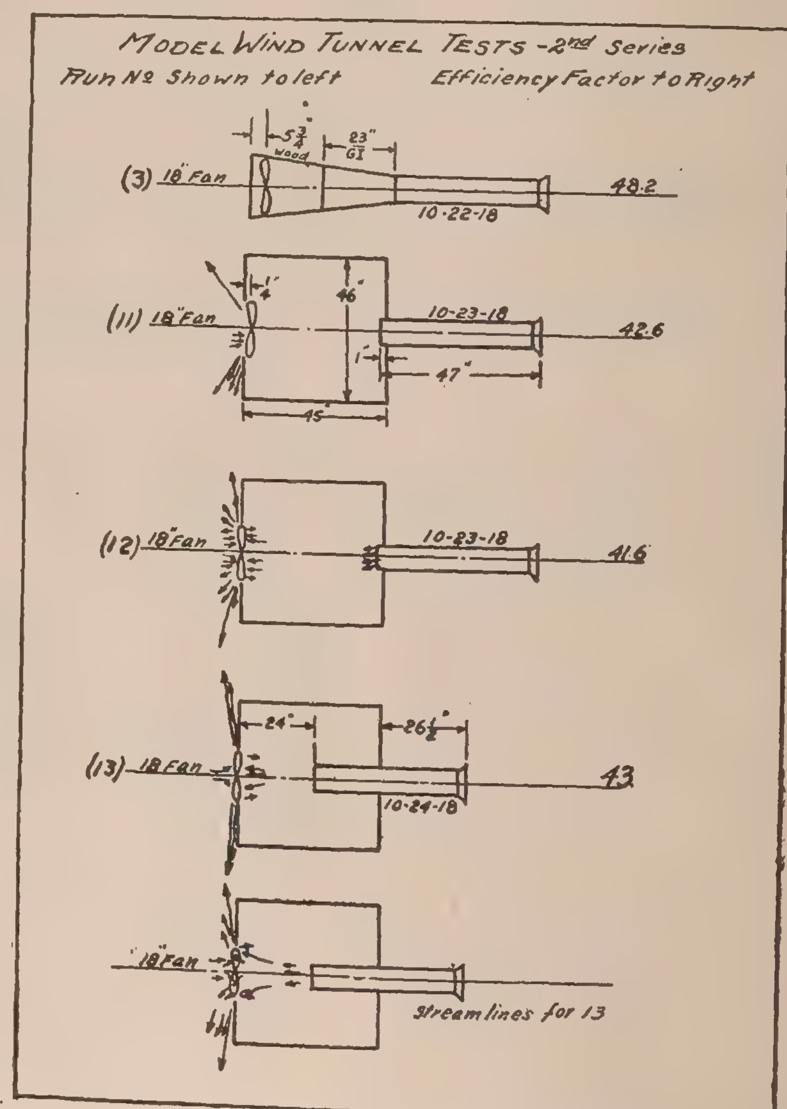

SERIES I
(16) 18"Fan 20" 45½ 44½ 8" 2" 12" 3½" 9/28/18 33.7
(17) 18"Fan 3½ 9/28/18 18" 11½" 35.9
(18) 18"Fan Honeycomb 10/1/18 Tangential Whirl at Periphery 3/8 1¼" Without honeycomb 46.6 40.3 with honeycomb
(19) 18"Fan 10/1/18 48.3
(20) 18"Fan 10/1/18 1½" Whirl Ring 3⅛ 1" 7" 18½" 48.9 Discharge Axial as Shown
(21) 18"Fan 1/8 7" (a) (b) 48.3 Flow between (a) & (b) is helical (b) is at 2/3 the Radius
(22) 18"Fan D D/3 7" 7/8" 10/1/18 Note: Manometer Fluctuates from .06 to .52 Reaching a max. value corresponding to an Index of 30
(23) 18"Fan 1/8" 10/1/18 19" 11½" 33

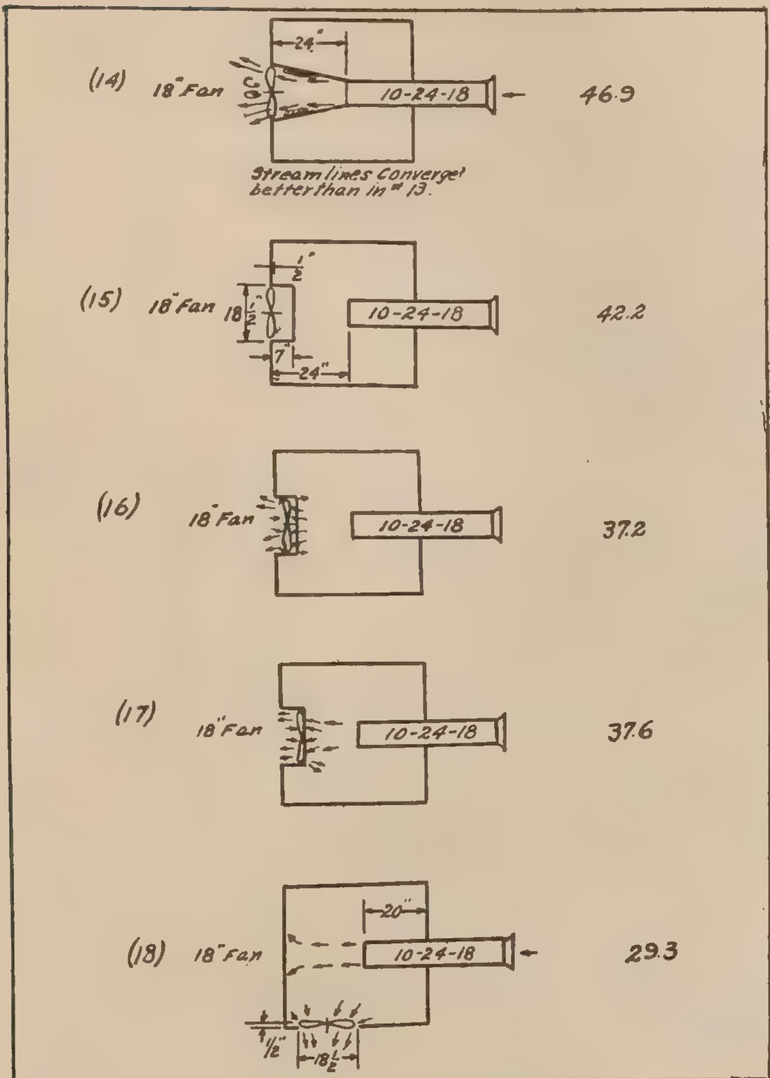

MODEL WIND TUNNEL TESTS - 1st Series
Run No Shown to left Efficiency Factor to Right
Heath 4B 12" Fan used in Runs 1-9
(1) 9-28-18 Honeycomb 12
(2) 9-28-18 Whirl is most noticeable at walls. 13.5
(3) Whirl 9-28-18 22.4
(4) 9-28-18 Whirl not excessive 39.1
(5) 9-28-18 Honeycomb 17.0 Rdg. is less steady than #4
(7) B&A Flow unsteady for A 9-28-18 A, unsteady. B, steady 18.0 Flow lines "A" unsteady, but after momentary application of discharge cone they become steadier and take direction "B"
(9) 9-28-18 8" 18" 11½" 28.7 Roar. Maintains value when small cone is Removed during Turn.

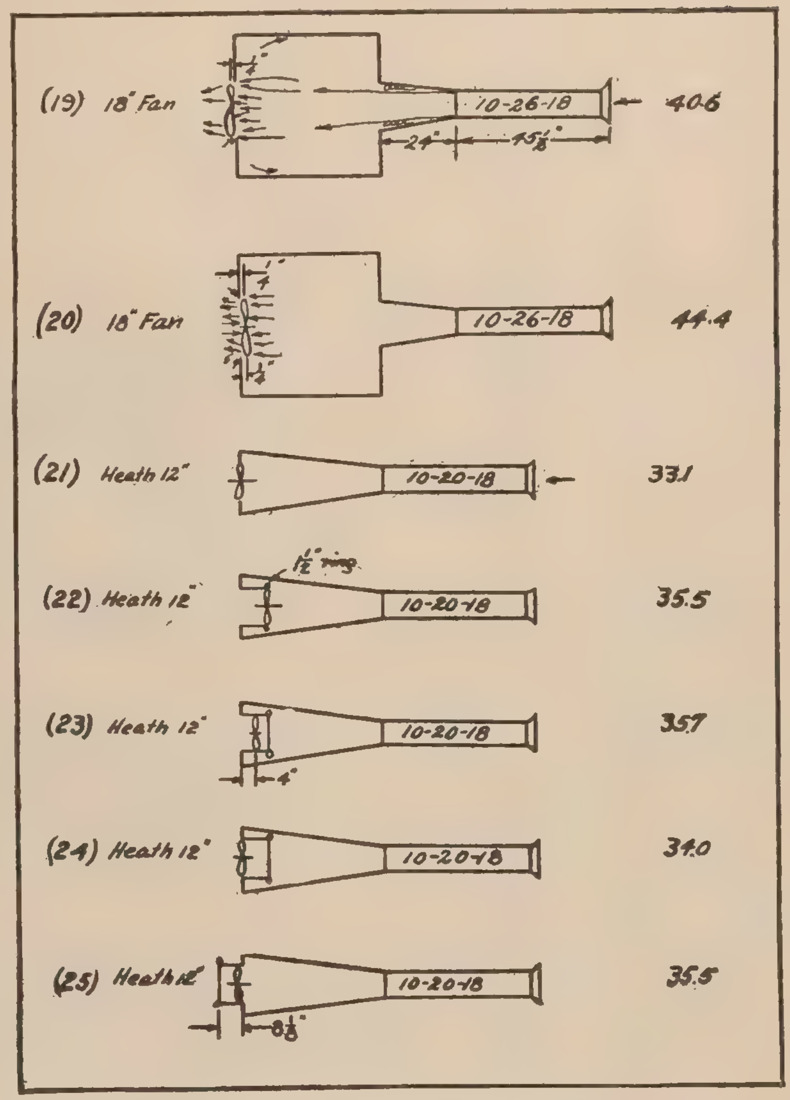

(24) 18" Fan
10/1/18
27
Note: Flow characteristic not greatly altered by removal of small cone.
Tangential here
(29) 18" Fan
1/4" Cl.
10-3-18
30 to 35 (Fluctuates)
whirl diagram
(30) 18" Fan
10-3-18
Pronounced Whirling
(31) 18 Fan
1/4" Cl.
10-3-18
(32) 18" Fan
Location of Tangential Whirl
10-3-18
15 to 20 (Fluctuates)
1" Neg. Flow Ring
(33) 18" Fan
No Neg. Flow Inside Cone
10-3-18
(34) 18" Fan
10-3-18
35 to 37
Removal of this cone during run does not affect reading.
(35) 18" Fan
10-3-18
39 to 41
Weaker Neg. Flow than # 34

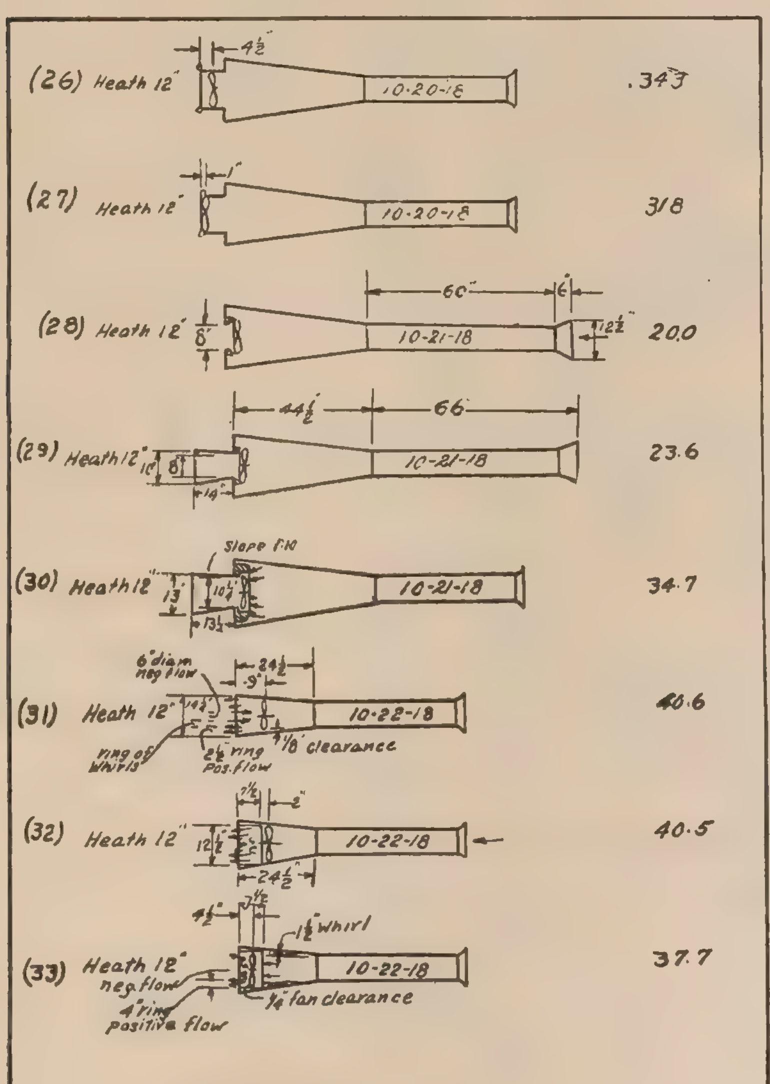

Neg. Flow Ring O.D.=12"
(26) 12" Heath Fan
10-1-18
3/16" CL
34.
(38) 12" Heath Fan
10-3-18
44.2
Violent Whirl & Slight Neg. Flow
(39) 12" Heath Fan
10-3-18
22
Less Steady Rdg than (38)

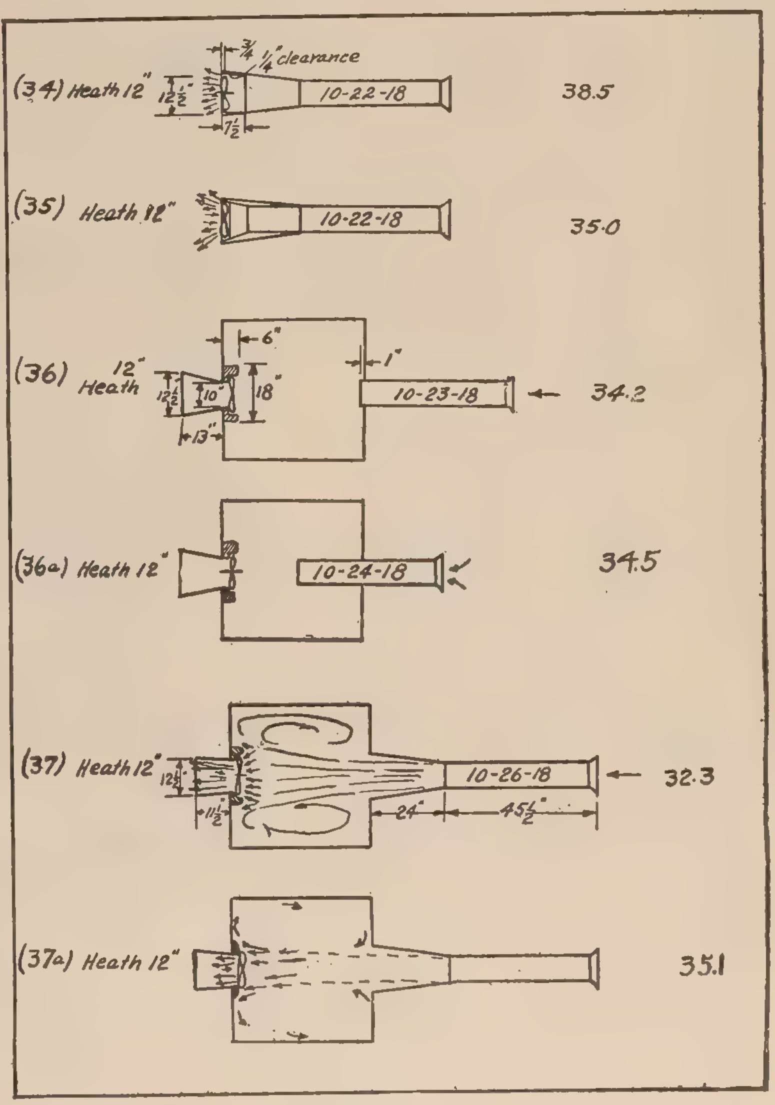

(56)
9-3-18
43.8
(57)
9-3-18
41.3 Fan roars in this position
(58)
9-3-18
40.5
(59)
9-3-18
37.0
(60)
9-3-18
23.9

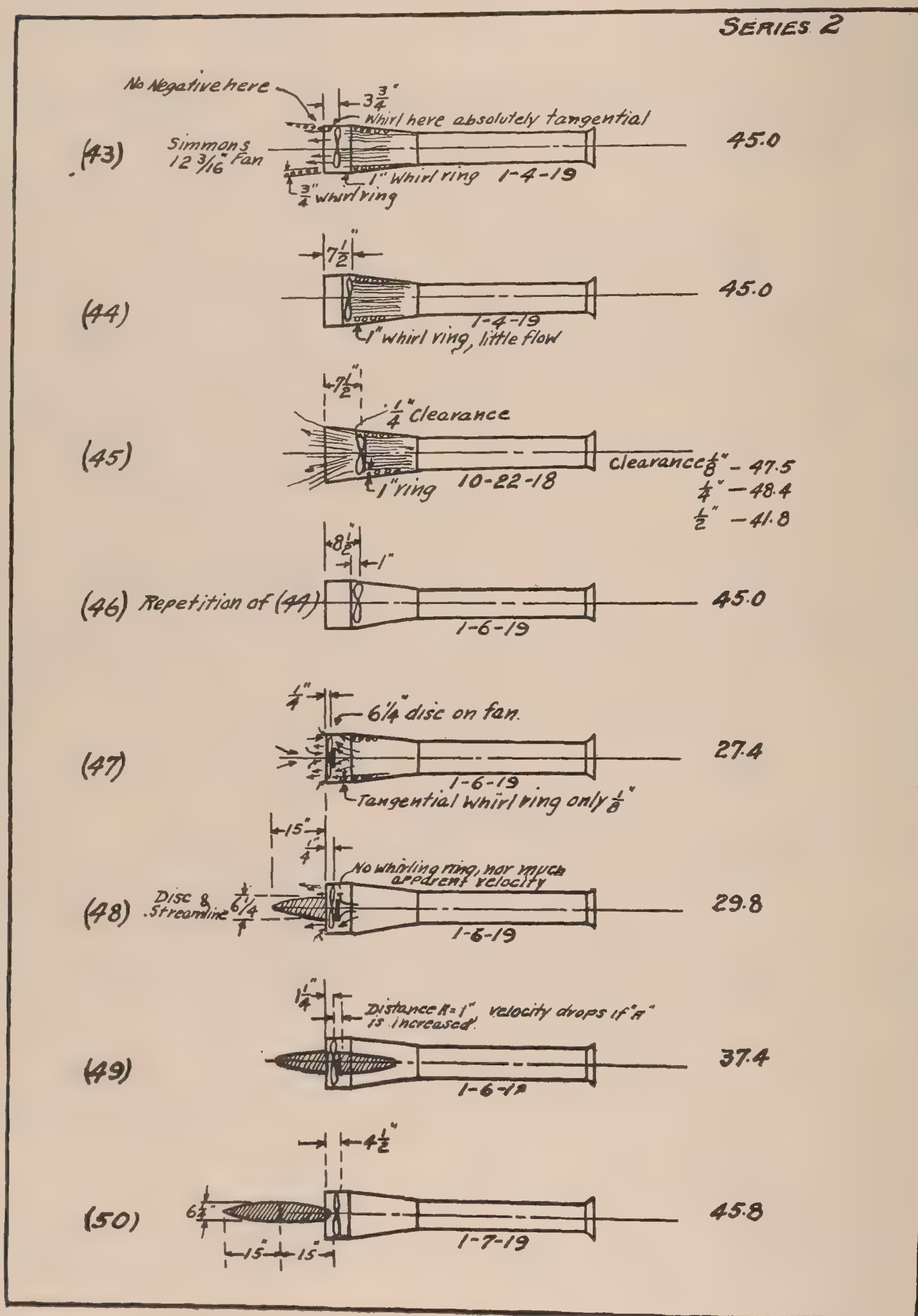

MODEL WIND TUNNEL TESTS -2nd Series
Run No. Shown to left
Efficiency Factor to Right
(3) 18" Fan
10-22-18
48.2
(11) 18" Fan
10-23-18
42.6
46"
47"
45"
(12) 18" Fan
10-23-18
41.6
(13) 18 Fan
24"
10-24-18
43
18" Fan
Streamlines for 13

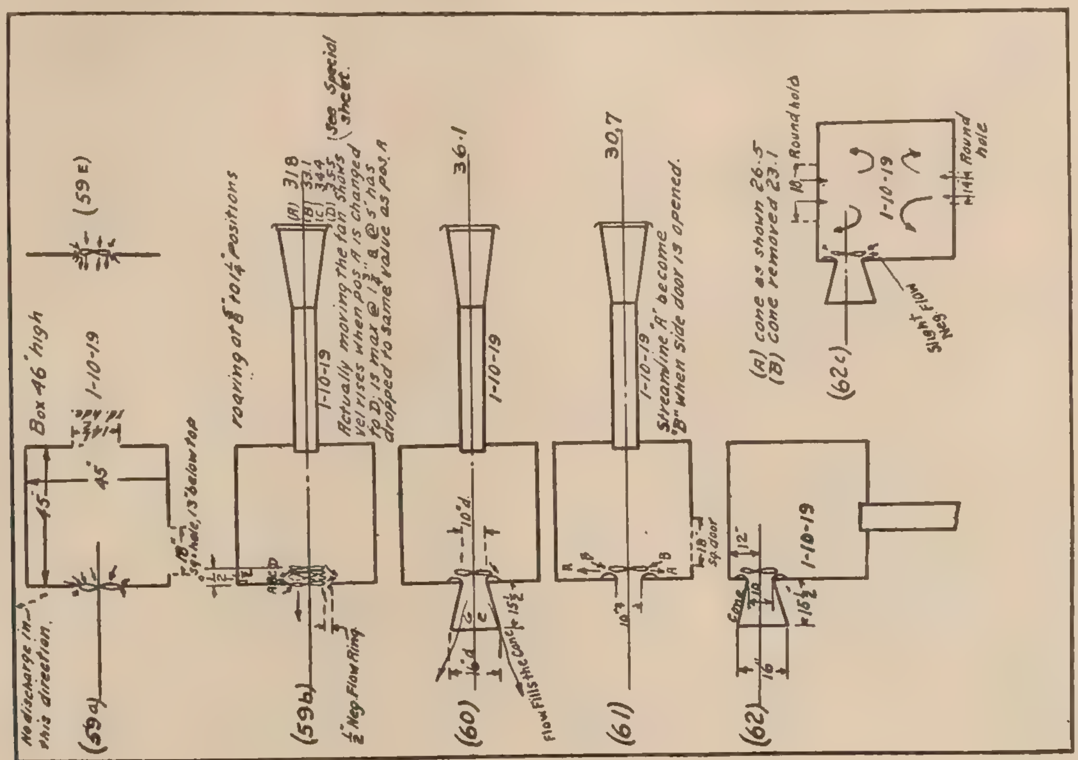

(14) 18" Fan
10-24-18
46.9
Streamlines Converge better than in #13.
(15) 18" Fan
10-24-18
42.2
(16) 18" Fan
10-24-18
37.2
(17) 18" Fan
10-24-18
37.6
(18) 18" Fan
10-24-18
29.3

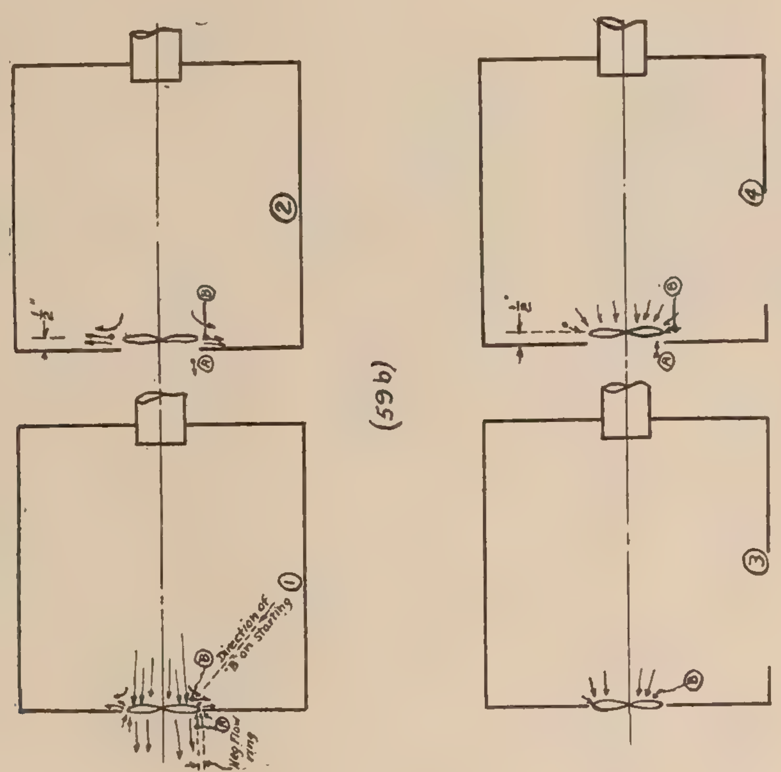

(19) 18" Fan
10-26-18
40.6
(20) 18" Fan
10-26-18
44.4
(21) Heath 12"
10-20-18
33.1
(22) Heath 12"
10-20-18
35.5
(23) Heath 12"
10-20-18
35.7
(24) Heath 12"
10-20-18
34.0
(25) Heath 12"
10-20-18
35.5

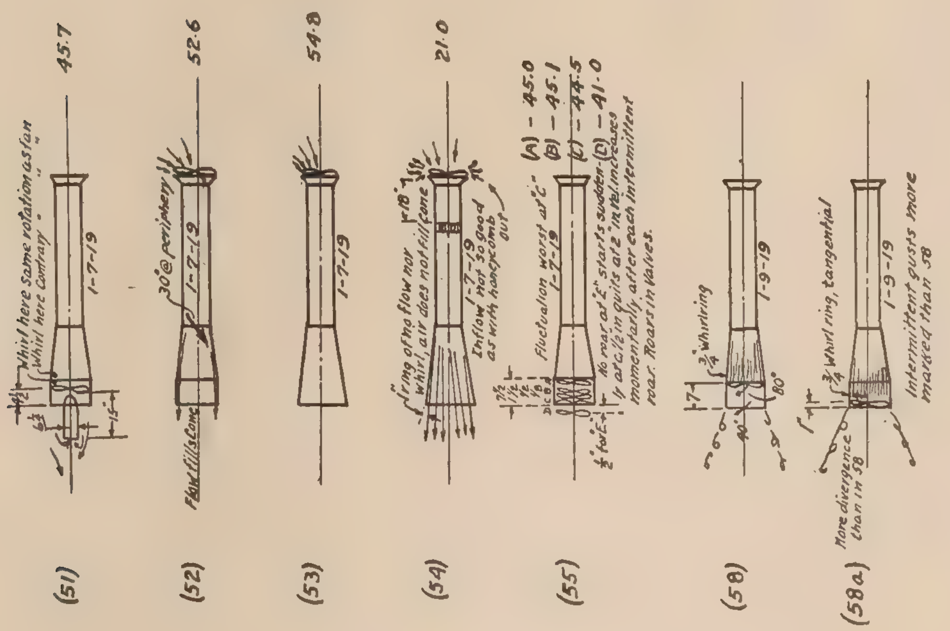

(26) Heath 12"
10-20-18
34.3
(27) Heath 12"
10-20-18
31.8
(28) Heath 12"
10-21-18
20.0
(29) Heath 12"
10-21-18
23.6
(30) Heath 12"
Slope 1:10
10-21-18
34.7
(31) Heath 12"
6" diam neg flow
ring of whirls
2½" ring Pos. flow
1/8" clearance
10-22-18
40.6
(32) Heath 12"
10-22-18
40.5
(33) Heath 12"
neg flow
4" ring positive flow
1½" whirl
¼" fan clearance
10-22-18
37.7

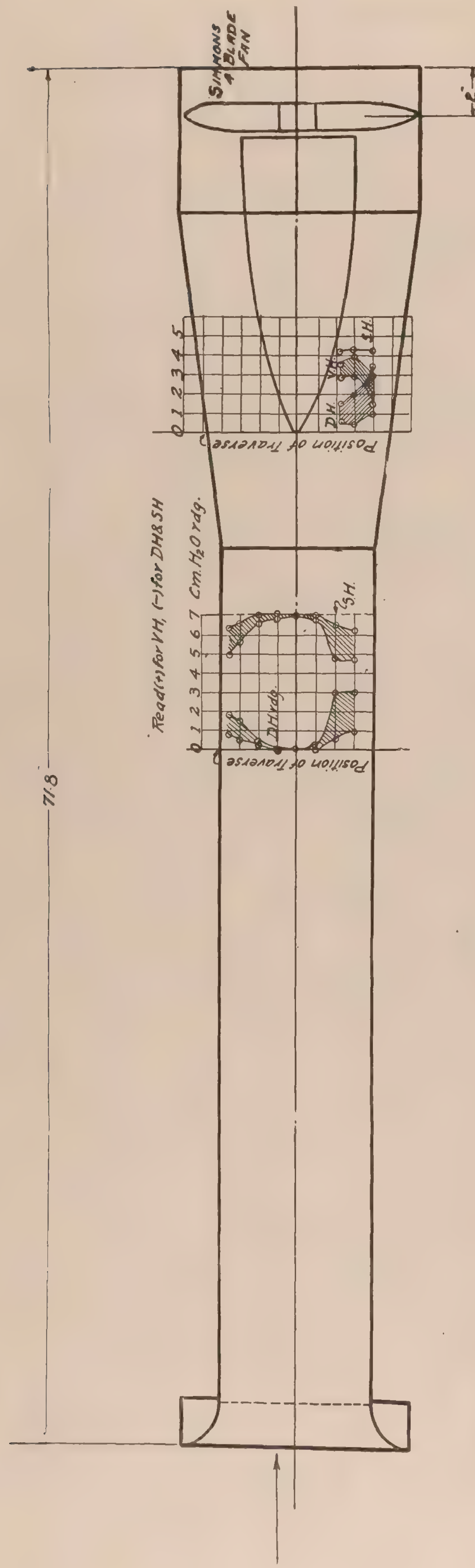

(34) Heath 12"
¼" clearance
10-22-18
38.5
(35) Heath 12"
10-22-18
35.0
(36) 12" Heath
10-23-18
34.2
(36a) Heath 12"
10-24-18
34.5
(37) Heath 12"
10-26-18
32.3
(37a) Heath 12"
35.1


SERIES 2
No Negative here
$3\frac{3}{4}$"
Whirl here absolutely tangential
(43)
Simmons
12 3/16" Fan
1" Whirl ring 1-4-19
3/4" whirl ring
45.0
$7\frac{1}{2}$"
(44)
1-4-19
1" whirl ring, little flow
45.0
$7\frac{1}{2}$"
1/4" Clearance
(45)
1" ring 10-22-18
Clearance 1/8" - 47.5
1/4" - 48.4
1/2" - 41.8
$8\frac{1}{2}$"
1"
(46) Repetition of (44)
1-6-19
45.0
1/4"
6/4" disc on fan.
(47)
1-6-19
Tangential whirl ring only 1/8"
27.4
15"
1/4"
No whirling ring, nor much apparent velocity
(48)
Disc & Streamline
6/4
1-6-19
29.8
$1\frac{1}{4}$"
Distance R = 1", velocity drops if "R" is increased.
(49)
1-6-19
37.4
$4\frac{1}{2}$"
(50)
$6\frac{1}{2}$"
15" 15"
1-7-19
45.8

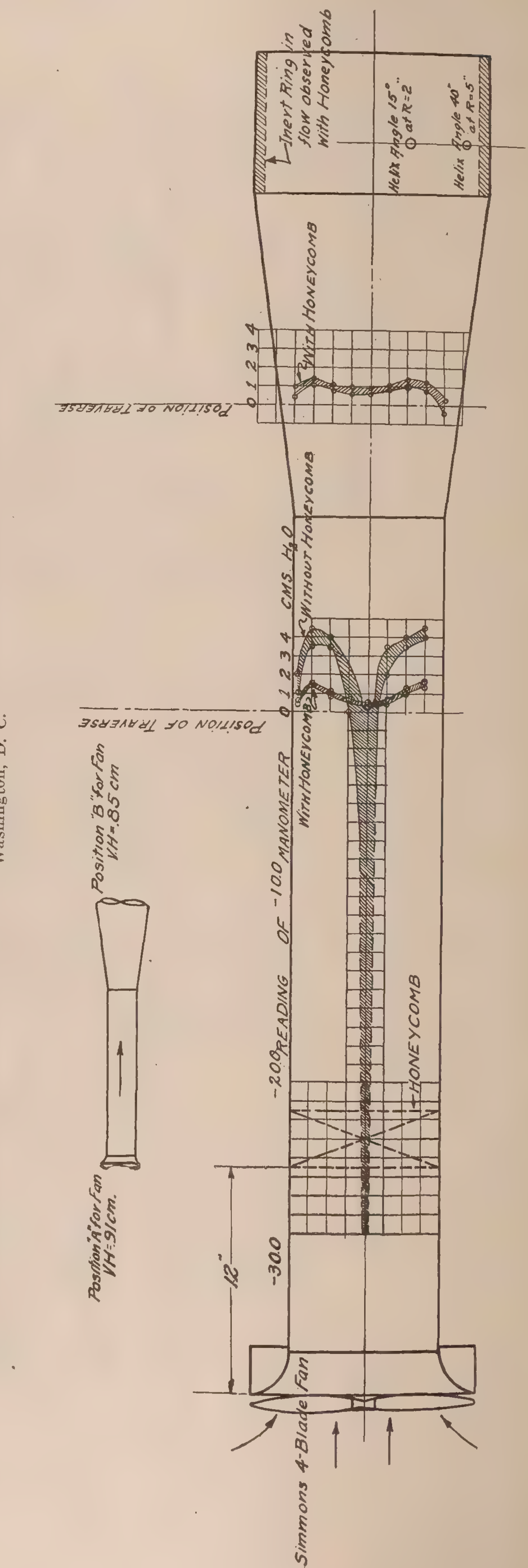

(59a)
(59E)
Box 46" high
1-10-19
No discharge in this direction.
roaring at 8" to 10¼" positions
(59b)
½" Neg. Flow Ring
1-10-19
Actually moving the fan shows vel. rises when pos. A is changed to D, is max @ 1¼" & @ 5" has dropped to same value as pos. A
(A) 31.8
(B) 33.1
(C) 34.4
(D) 35.5
See Special Sheet.
(60)
36.1
1-10-19
Flow Fills the Cone
(61)
30.7
1-10-19
Streamline "A" become "B" when side door is opened.
Sq. door
(62)
1-10-19
(62c)
(A) Cone as shown 26.5
(B) Cone removed 23.1
Round hole
Round hole
Slight Flow Neg.

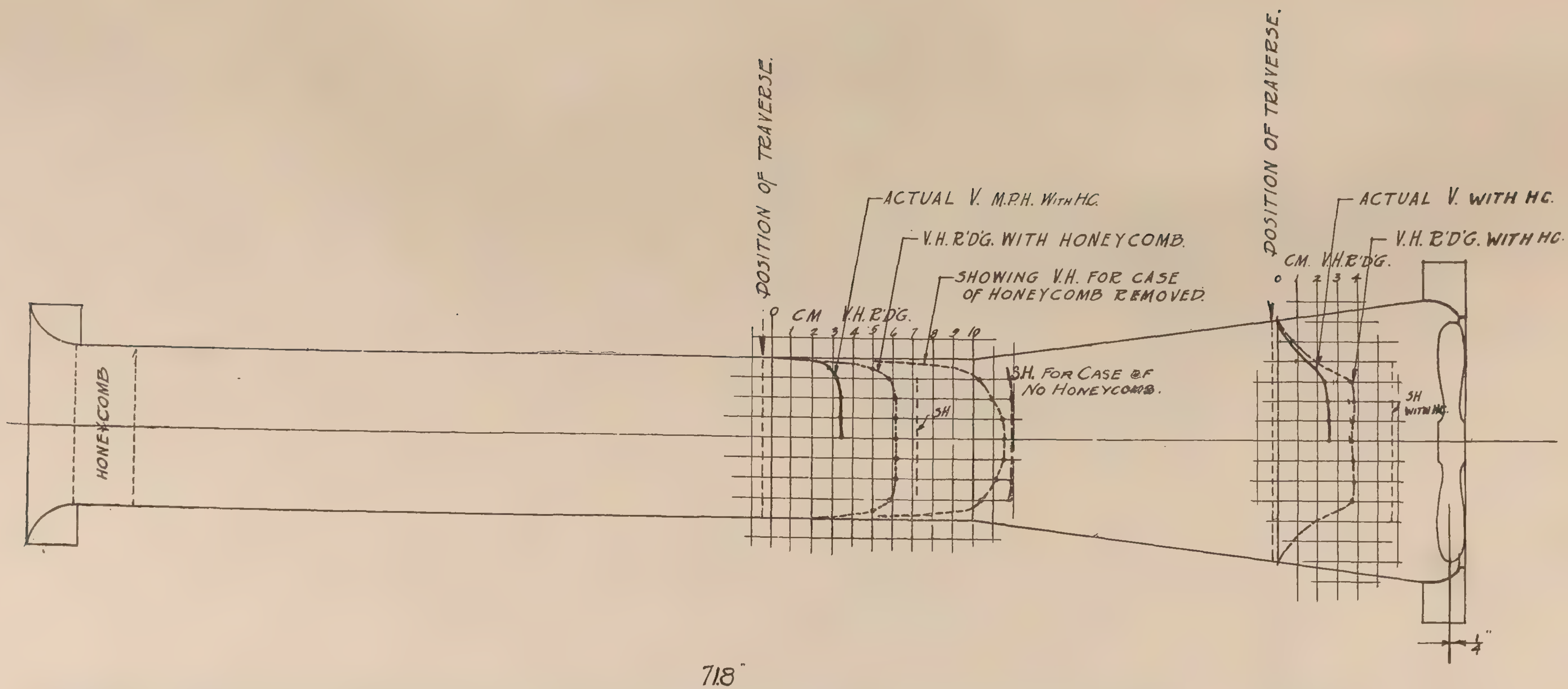

(59b)
1
2
3
4
Direction of "B" on Starting
Neg. Flow ring


(51)
45.7
1-7-19
Whirl here same rotation as fan
Whirl here Contrary
(52)
52.6
1-7-19
30° @ periphery
Flow Fills Cone
(53)
54.8
1-7-19
(54)
21.0
1-7-19
Ring of no flow nor whirl, air does not fill cone
Inflow not so good as with honeycomb out
(55)
(A) - 45.0
(B) - 45.1
(C) - 44.5
(D) - 41.0
1-7-19
Fluctuation worst at "C"
Roars in valves.
(58)
1-9-19
¾" Whirl ring
(58a)
1-9-19
¾" Whirl ring, tangential
Intermittent gusts more marked than 58
More divergence than in 58

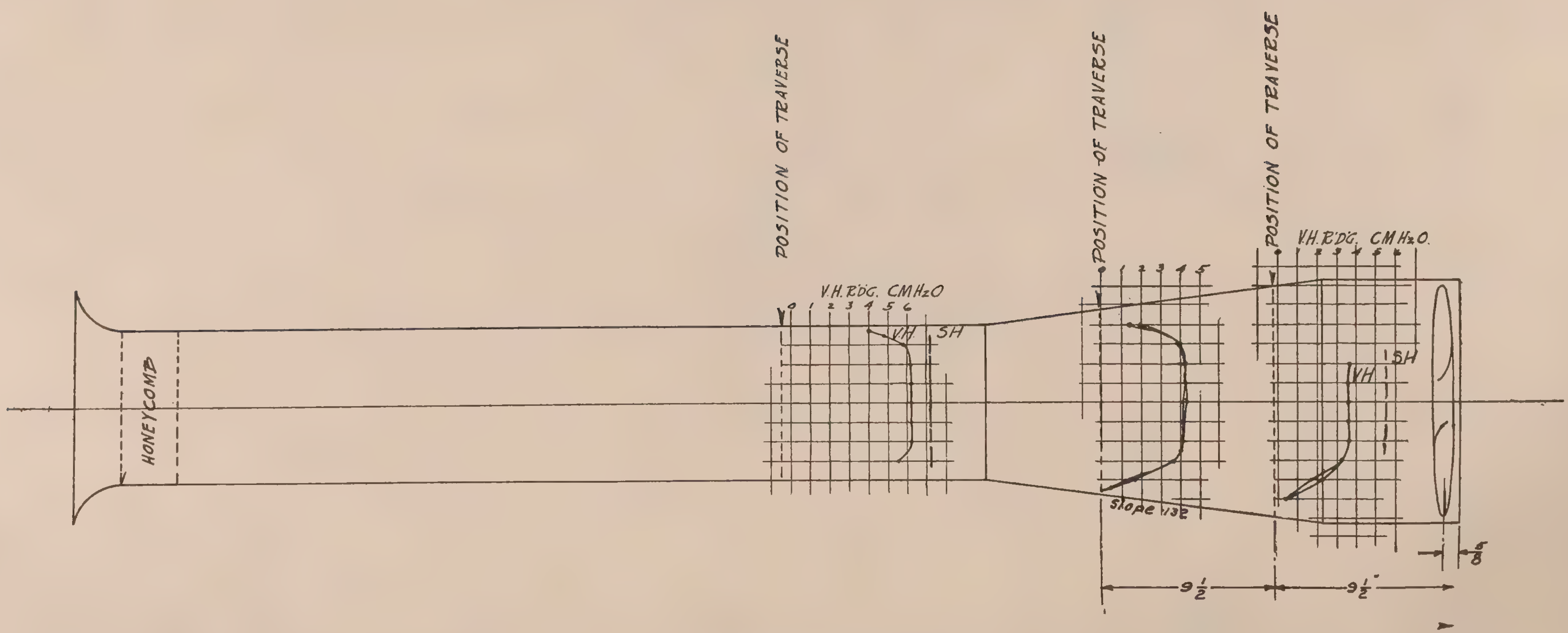


Traverse of air flow. Model wind tunnel having core in cone. Test No. 66 (c) 1–8–19. Fan 2" inside discharge place. No honeycomb; discharge sleeve in place. Washington, D. C.

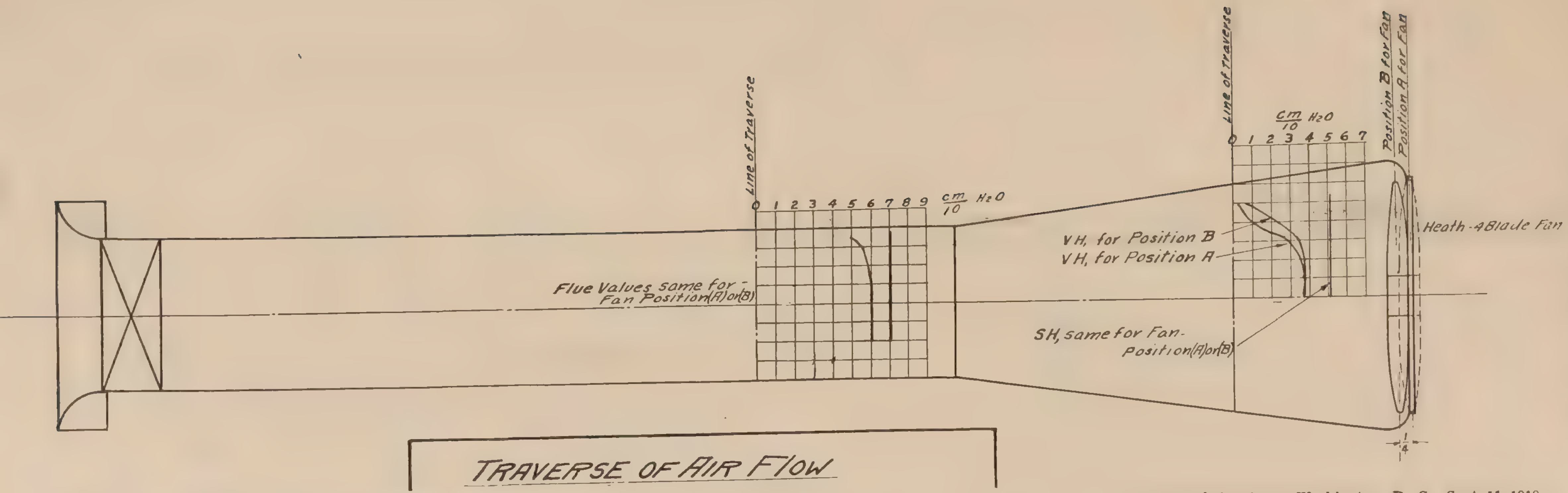




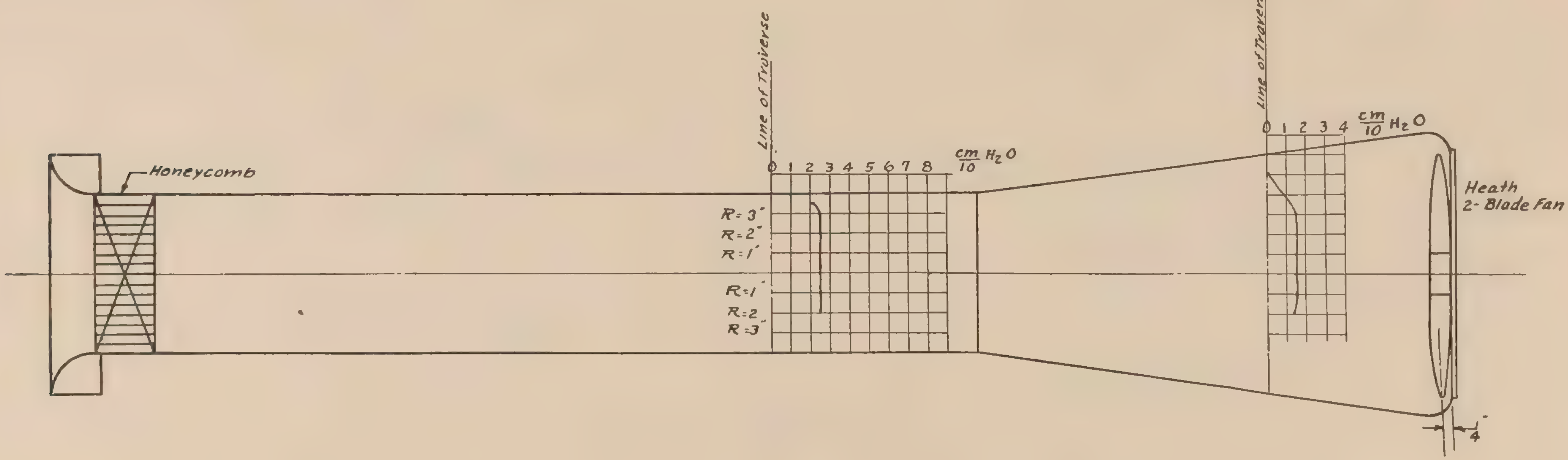
Run 66 (f). Traverses of typical wind tunnel on $\frac{1}{12}$ scale model. Heath 4-blade, 12" propeller, 2500 R. P. M. Comparison of traverse with and without honeycomb. Sci. and Res. Department. B. A. P. Washington—1:19:19—Fales.

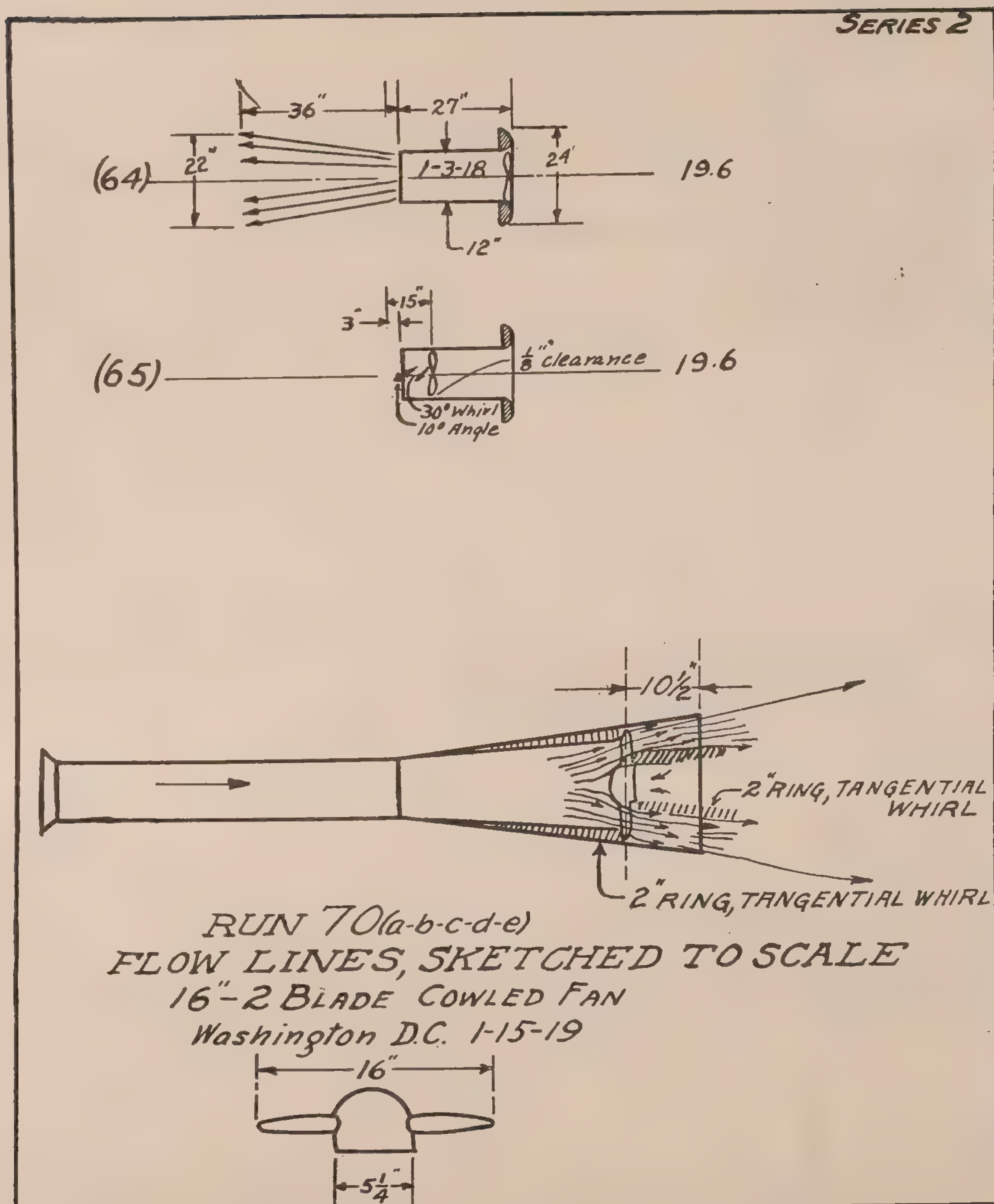


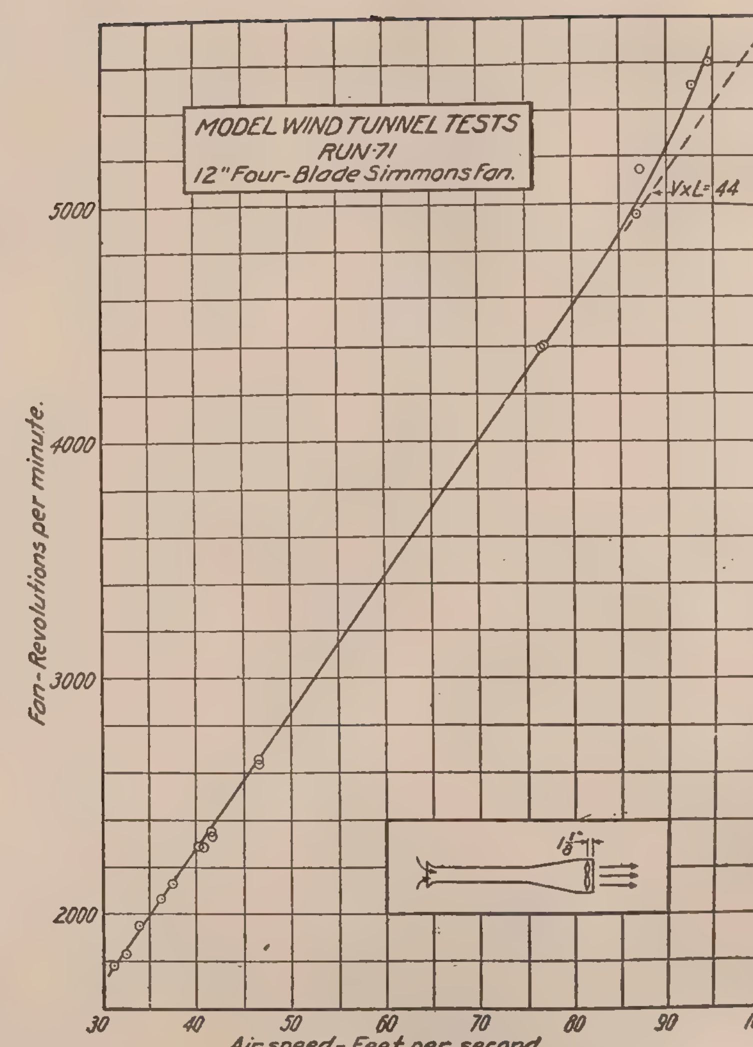
Run 66 (g). Traverses of typical wind tunnel on $\frac{1}{12}$ scale model. Simmons 4-blade, $12\frac{3}{16}''$ propeller, 2240 R. P. M. Honeycomb in place. M. P. H.=13.1 x V. H. R' D' G. Sci. & Res. Department. B. A. P. Washington—1:9:19—Fales.

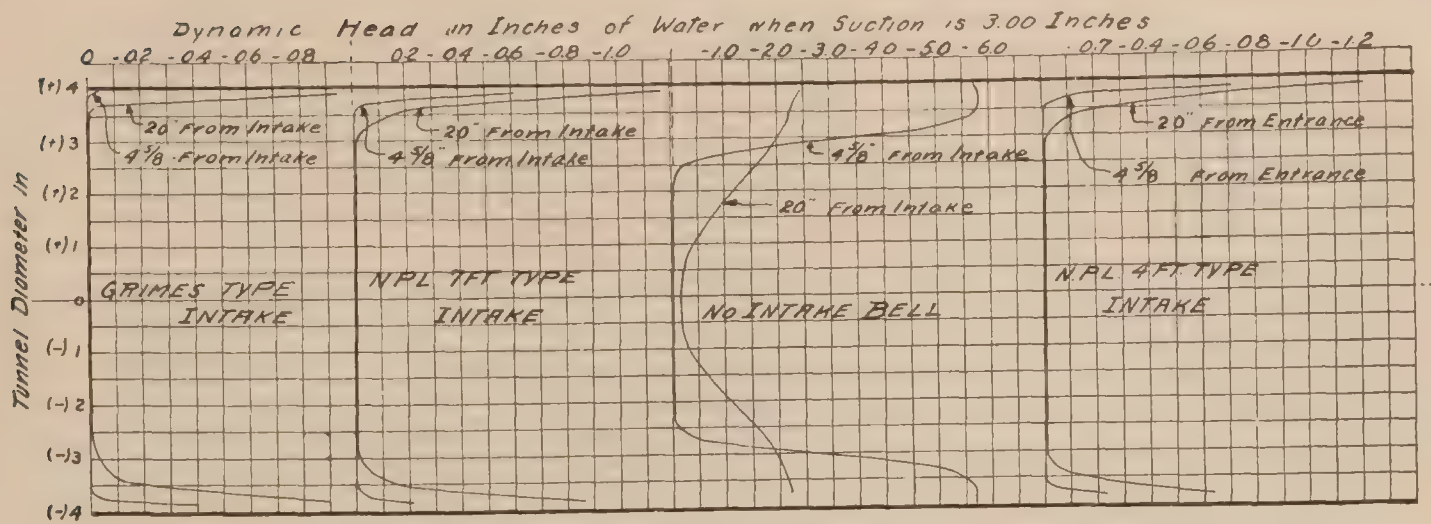


Traverse of air flow, giving comparison of fan positions, Model wind tunnel test Nos. 66(h) and 66(i). Heath 4-bladed 12-inch fan, 2,500 R. P. M. Honeycomb in place. Washington, D. C. Sept. 11, 1918.




SERIES 2
36"
27"
(64)
22"
1-3-18
24"
19.6
12"
15"
3"
(65)
1/8" clearance
19.6
30° Whirl
10° Angle
10½"
2" RING, TANGENTIAL WHIRL
2" RING, TANGENTIAL WHIRL
RUN 70(a-b-c-d-e)
FLOW LINES, SKETCHED TO SCALE
16"-2 BLADE COWLED FAN
Washington D.C. 1-15-19
16"
5¼"

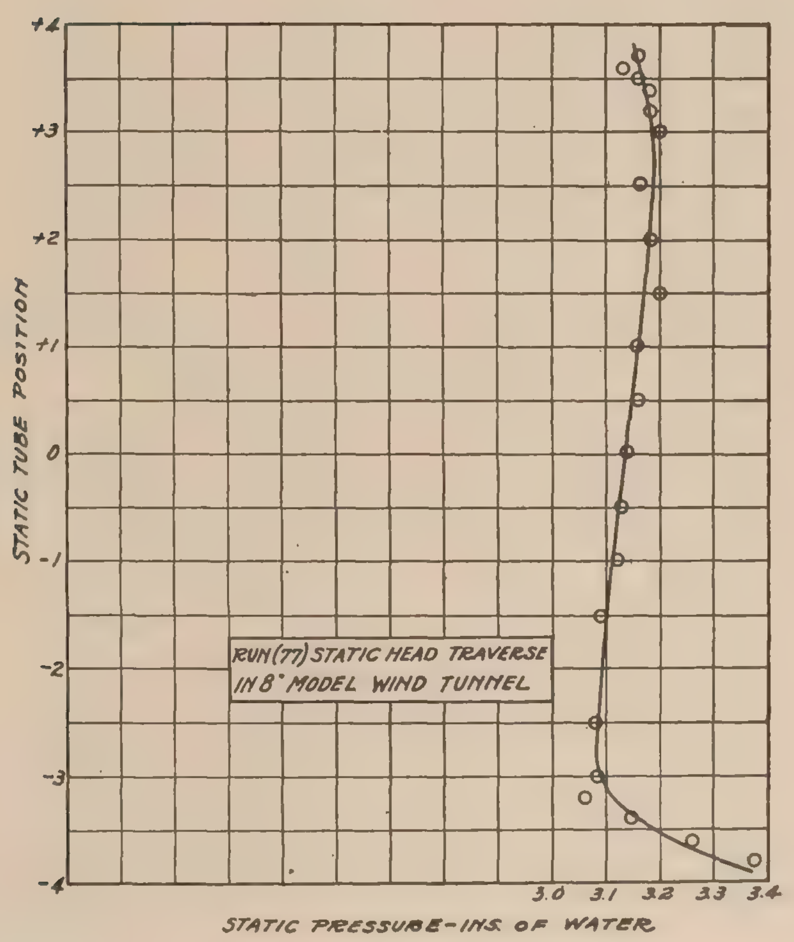

MODEL WIND TUNNEL TESTS
RUN 71
12" Four-Blade Simmons Fan.
V×L=44
Fan-Revolutions per minute.
5000
4000
3000
2000
30
40
50
60
70
80
90
100
Air speed-Feet per second.



DYNAMIC HEADS IN MODEL WIND TUNNEL FLUE

FOR COMPARING VELOCITY TRAVERSES OF DIFFERENT INTAKES

2nd SERIES TESTS 72 TO 76 no Honeycomb used
Flue connected to Large Wind Tunnel No Cone used
10-2-19 DAYTON OHIO



REPORT NO. 83.

WIND TUNNEL STUDIES IN AERODYNAMIC PHENOMENA AT HIGH SPEED.

PART II.

THE McCOOK FIELD WIND TUNNEL.

By F. W. Caldwell and E. N. Fales.

The information available on propeller aerofoils in the past has been comparatively meager, and most of it has been obtained at air speeds of about 30 to 60 miles per hour. It has always been a matter of a good deal of concern among aeronautical engineers as to whether the information obtained at these low speeds is reliable when applied to propellers whose velocities are many times greater.

As a matter of interest, tip speeds of some of the propellers in actual use are given below:

	Miles per hour.
USD-9 airplane with Liberty-12 engine	650
VE-7 airplane with Hispano-Suiza 150-horsepower engine	545
Thomas-Morse airplane with 80-horsepower LeRhone engine	380
Verville Chasse airplane with 300-horsepower Hispano-Suiza engine	600
Roché XB-1-A airplane with 300-horsepower Hispano-Suiza engine	625
Curtiss JN-4 airplane with Curtiss OX-5 engine	420
DH-9 airplane with Rolls-Royce 375-horsepower engine	430

It is evident that the speeds given above are so far in excess of the usual wind-tunnel speed as to justify a little skepticism in applying results obtained in the slow-speed tunnel.

The results of many static tests of propellers have shown that the horsepower does not vary directly as the cube of the revolutions per minute, but increases much more rapidly than the cube at very high tip speeds. It has consistently been shown, as the result of a large series of static tests carried out by the writers, that the ratio of thrust to torque varies considerably with revolutions per minute. At the same time wind-tunnel tests on propellers have indicated that the experimental no-thrust pitch increases somewhat with the revolutions per minute of the propeller.

These considerations have resulted in the inauguration of a series of wind-tunnel tests at very high speeds in order to investigate scaling effect on the lift and drag coefficients due to such speeds.

During the winter of 1918 it was proposed by the engineering division at McCook Field to erect a small tunnel, primarily for the calibration of instruments. (See fig. 6.) The writers, working in conjunction with Mr. C. P. Grimes, determined the design of this tunnel for the attainment of the greatest possible speed with the power available, so as to adapt the apparatus to model tests of propeller aerofoils.

DESCRIPTION OF McCOOK FIELD 14-INCH WIND TUNNEL.

As in other wind tunnels, air is sucked through a horizontal tube, where it blows against a small model at known velocity. The model is supported by a rod projecting from a suitable balance into the tunnel, and the forces concerned in flight can thus be measured. The air after passing the model is decelerated in an expanding cone and exhausted into the room by a propeller fan. Description of the McCook Field tunnel need include only those features which differ from the standard type.

The intake trumpet, tube, and expanding cone have the general form of a Venturi tube with a length of 18⅔ feet; the intake trumpet has a diameter of 3½ feet and radius of curvature of 22¾ inches; the throat diameter is 14 inches and the fan diameter is 5 feet. (See fig. 10.) The location of the test section close to the intake is advantageous, as discussed

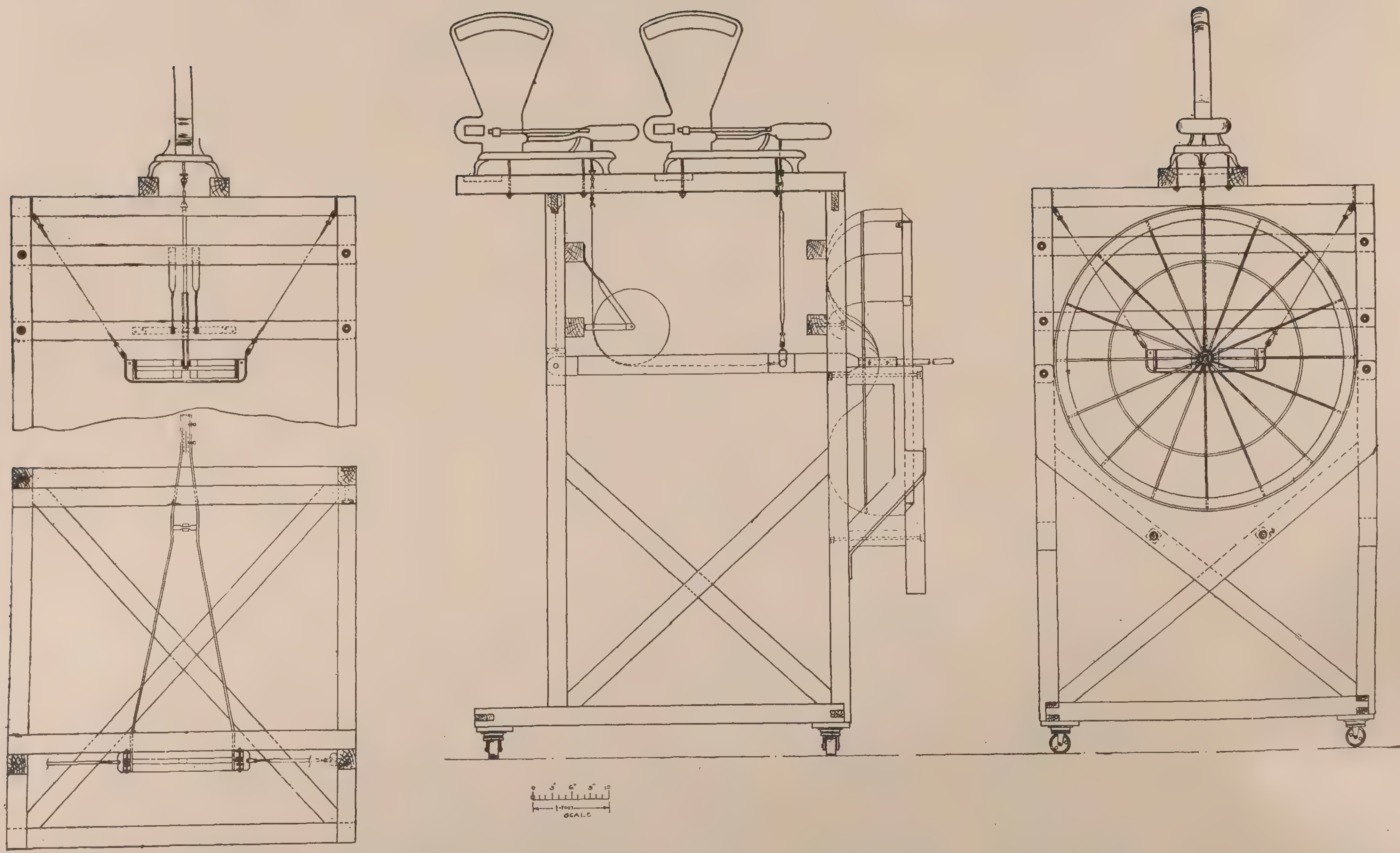


FIG. 9.—Weighing mechanism and entrance vanes for high-speed wind tunnel.


McCook Field High Speed Wind Tunnel
General View

FIG. 6.

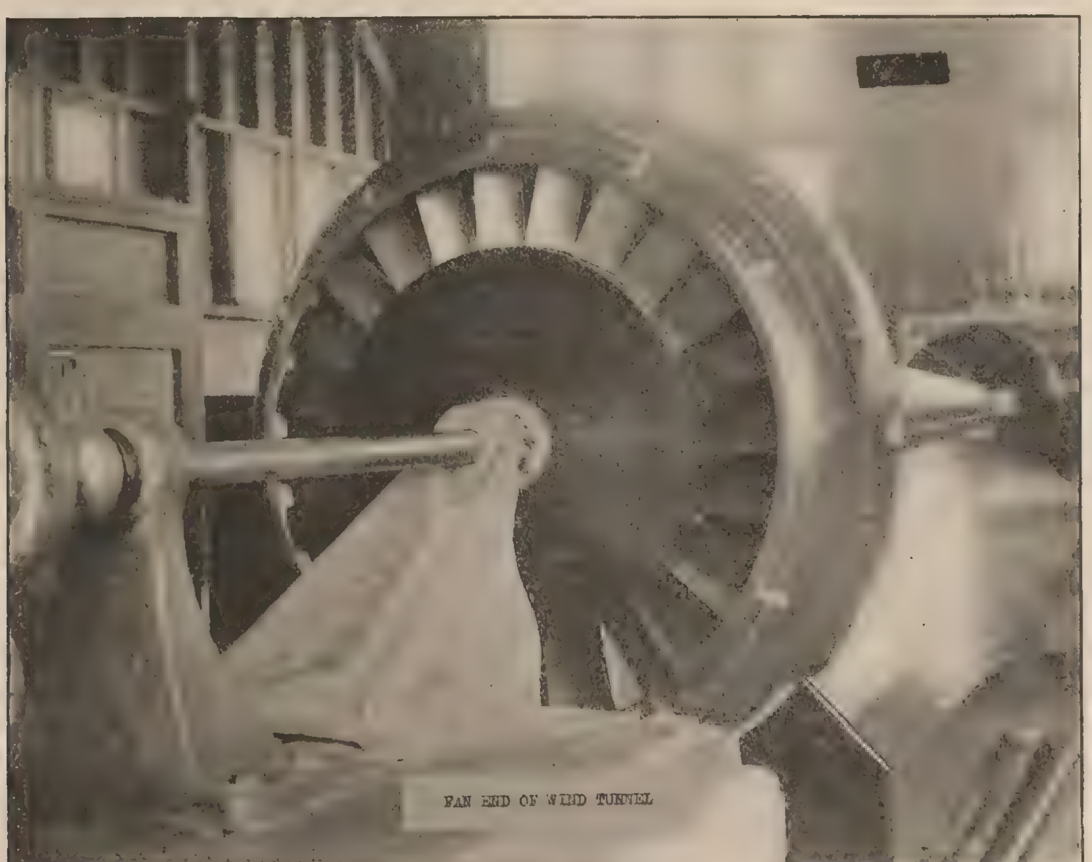
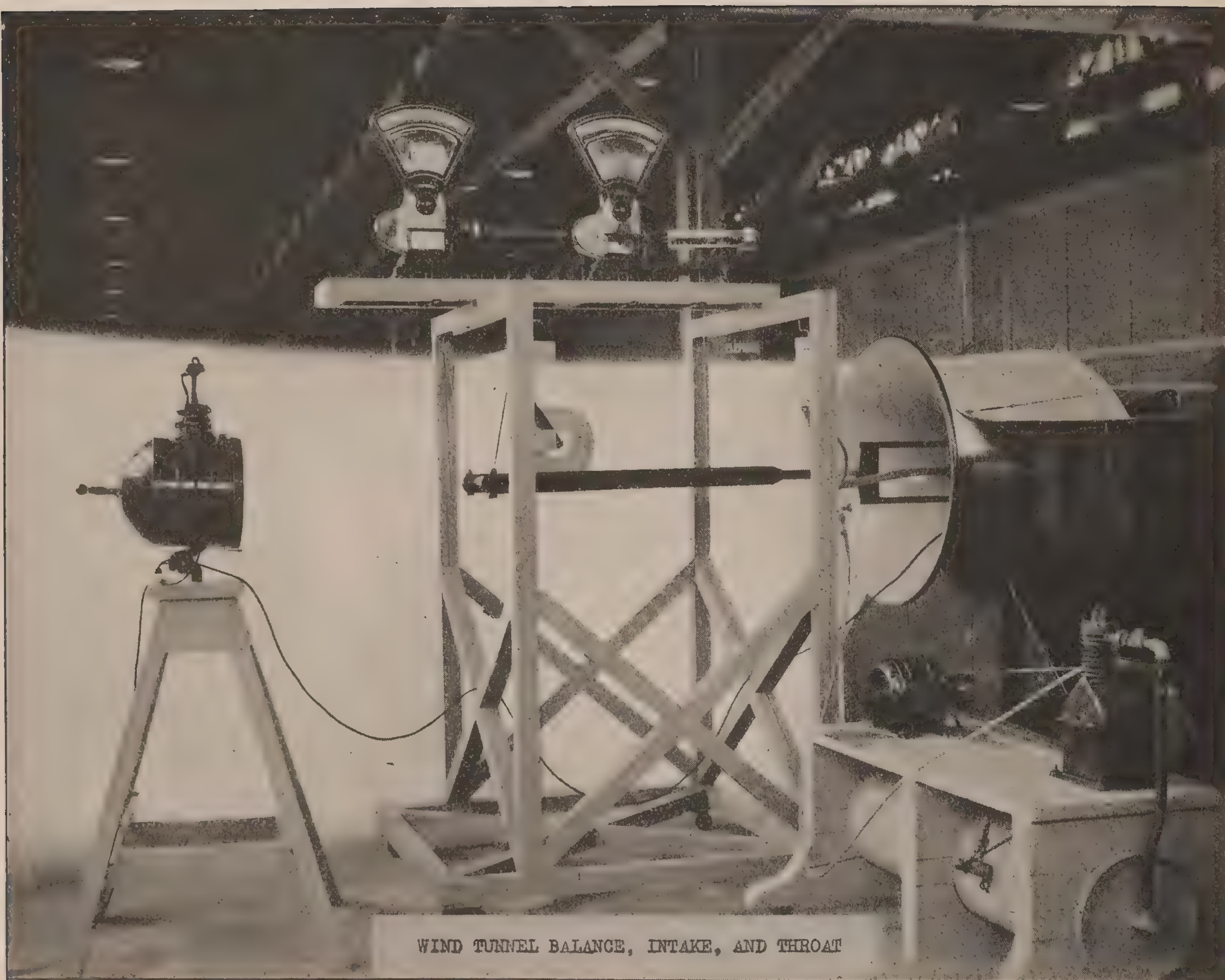

FAN END OF WIND TUNNEL

FIG. 7.



WIND TUNNEL BALANCE, INTAKE, AND THROAT

FIG. 8.

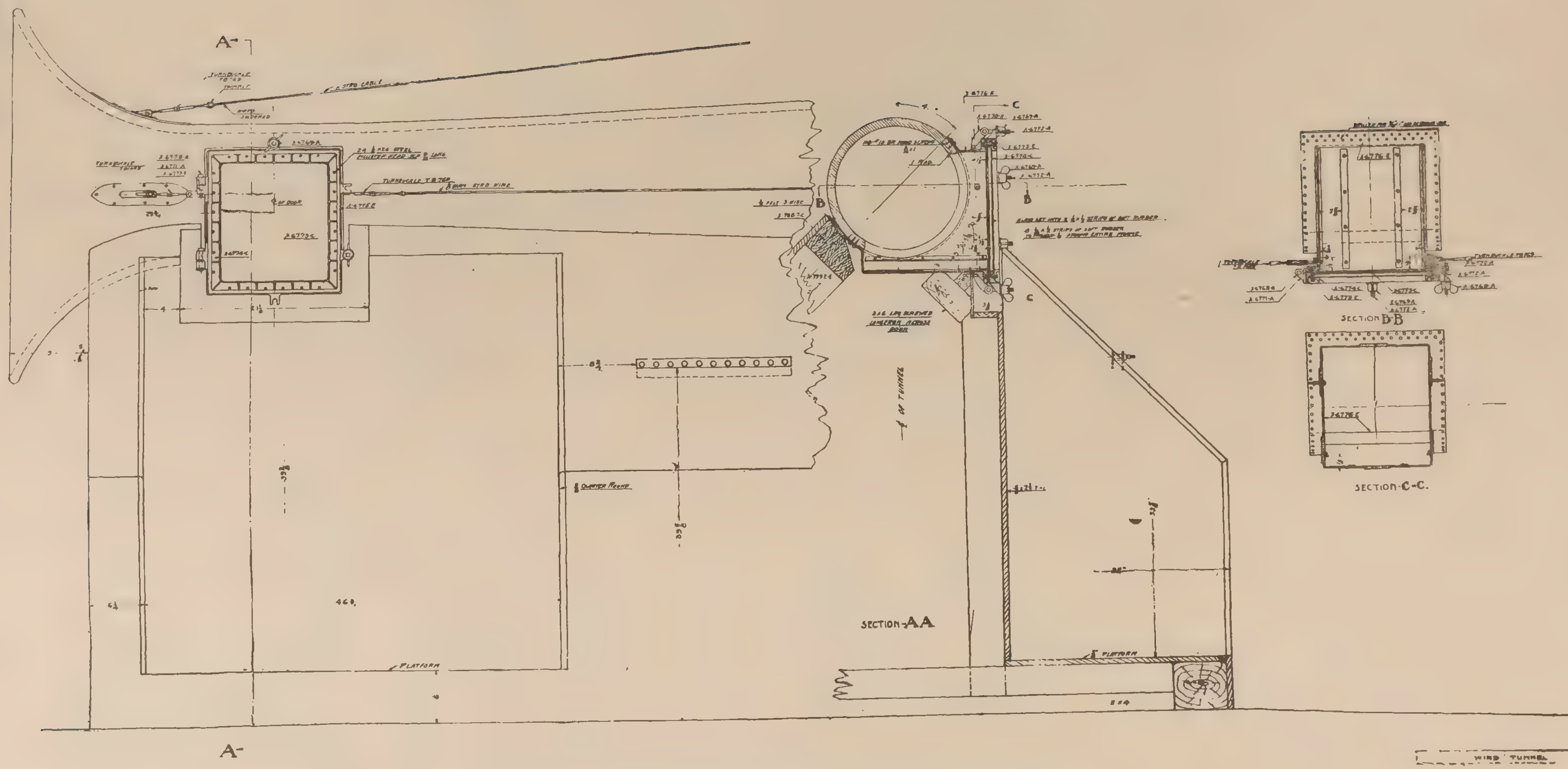


FIG. 10.—Entrance and throat of high-speed wind tunnel.

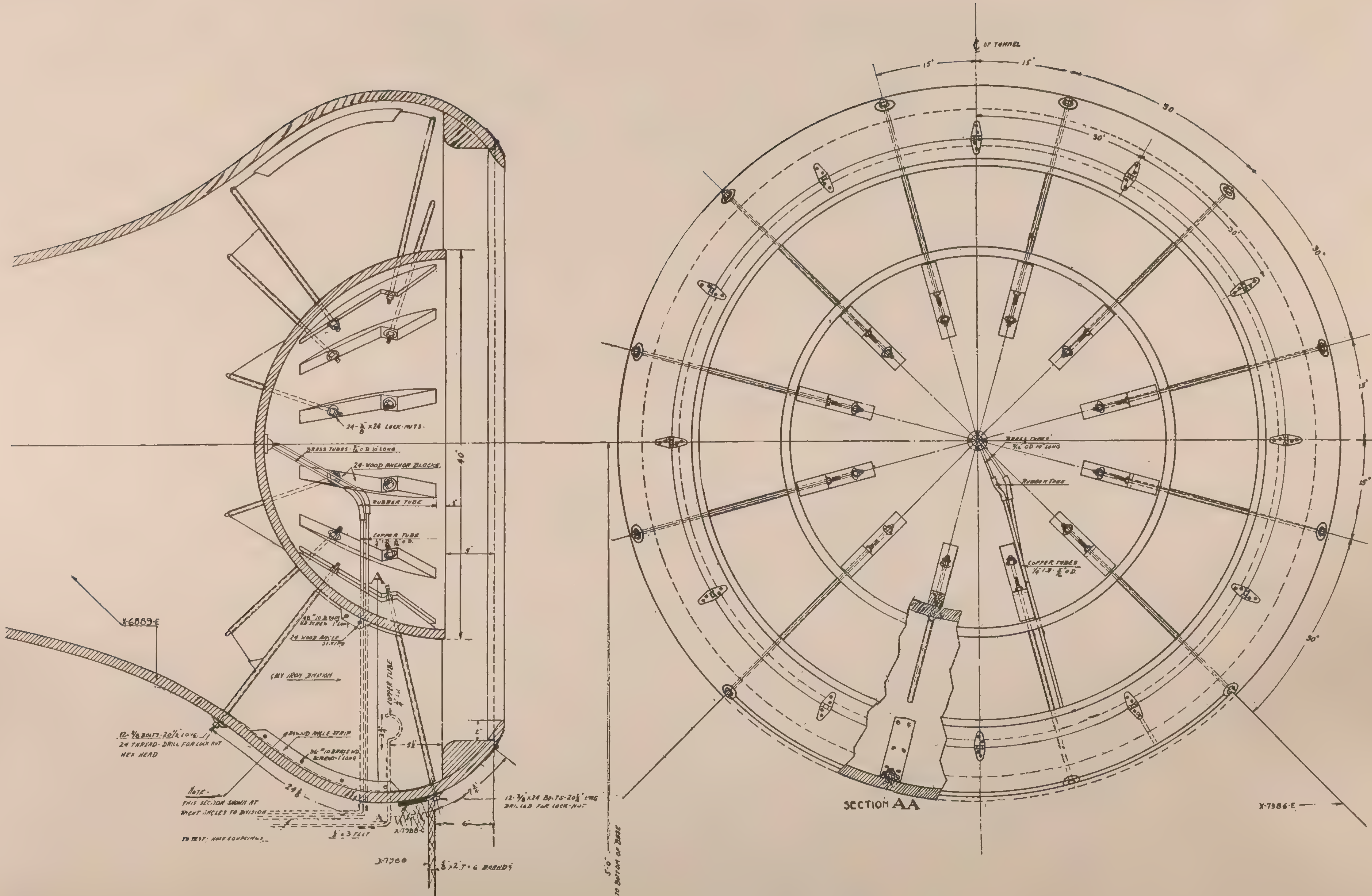


FIG. 11.—Arrangement of vanes in exit cone.

in section 1 of this report; there is no appreciable loss of energy at the intake, and the traverse of a diameter at the commencement of the throat shows no appreciable velocity variation except at the walls. The usual honeycomb is omitted, but a four-bladed "straightener" 48 inches long is inserted in the cone 4 feet downstream from the model, and there is a straightener outside the intake having 16 flat radial blades. The former "straightener" cuts the fluctuations of the velocity from 15 per cent down to 2 per cent. The cone is of 5° angle for the first 100 inches.

The power plant consists of a Sprague dynamometer, capable of delivering 200 horsepower for one-half hour at 250 volts and 1,770 revolutions per minute without overheating. The 5-foot fan is made with a solid center disk 40 inches in diameter, and has 24 blades 10 inches long. At the upstream side of the 40-inch disk, a bell of equal diameter is fixed in the tunnel so that the air is led up to the annular discharge opening with a minimum of eddies. (See fig. 11.)

The balances are of two types. The first one (fig. 9), designed by C. P. Grimes, measures lift and drift on two separate instantaneous-reading Toledo scales. It is mounted upon a portable carriage. The spindle for the model projects horizontally and axially from this carriage into the mouth of the wind tunnel, carrying the model at its free end. The spindle terminates in a thin, flat bar, the latter clamping a graduated disk which is rigid with the model at the center of the span. This type of balance possesses three advantages, as follows: (1) Instantaneous readings make it possible to synchronize balance and velocity observations and to practically eliminate the effect of velocity fluctuations; (2) the air forces can be qualitatively studied, as, for instance, in the case where a given set-up has two values of K_y, when the balance can be seen to change from one reading to another; (3) the method of support affords a highly accurate means of skin friction observation.

The second type of balance is of the "vector" type, invented by the Wright Bros., with improvements developed by the writers; the principle is indicated in the sketch figure 18. This balance reads L/D with an accuracy superior to the ordinary type; and it reads lift and drift in terms of static pressure. The advantage of the latter feature is that the reading is deadbeat.

OPERATION OF THE WIND TUNNEL.

The method of carrying out the test consists in setting the model in the tunnel at a known angle of attack and measuring the lift and drift forces by means of the indicating Toledo scales. As the precision of these instruments is better than one-tenth per cent, they are considered sufficiently accurate for work of this kind. The three readings of velocity head, lift, and drag are taken by three different observers, the readings being synchronized by means of signals. The tunnel is run at speeds varying for each angle, from about 30 miles per hour to about 450 miles per hour; lift, drag, and velocity-head readings being taken at each speed.

In order to check up the direction of the wind in the tunnel the model is turned upside down and the run repeated with the model set at the same angle. By this means it is found that there is a fairly uniform correction of about 0. 4°. This correction has not been applied to the small charts showing K_y for a given angle at various speeds, but has been applied to the larger chart in the center showing K_y plotted against angle.

An observer at the lift scale chooses a point about which the indicator hovers, and when the pointer is so hovering he makes a signal; a second observer on the drift scale, and a third observer at the manometer then make simultaneous observations, each observer having previously become accustomed to the respective lag between his instrument and that of the observer who gives the signal. In this way the fluctuations of velocity in the wind tunnel become less important for accurate results. An automatic recording device for doing the work of the observers at once suggests itself, assuming that the various instruments are properly synchronized. The development of such an instrument has been investigated but not completed.

The tests are made with an increasing velocity; that is, the motor is started at a low r. p. m., a set of readings taken, and then the motor speed increased by means of a rheostat. Occasional

check runs are made with decreasing speeds. Where speeds are approached at which the flow becomes unstable, the condition is easily observed upon the balances, which may be seen to hover successively at two distinct points, the speed remaining the same.

EFFICIENCY OF THE WIND TUNNEL.

The question of efficiency of the wind tunnel is one which was made the subject of much preliminary study. By efficiency is understood the ratio of kinetic energy of the air stream at the throat of the tunnel minus the energy absorbed by the fan, all divided by the energy of the air stream at the throat.

$$e=\frac{\frac{\rho}{2g}AV^3-E}{\frac{\rho}{2g}AV^3}$$

where $\frac{\rho}{g}$ is the density, A the cross sectional area of the throat, V the velocity at the throat, and E the rate of absorption of energy by the fan.

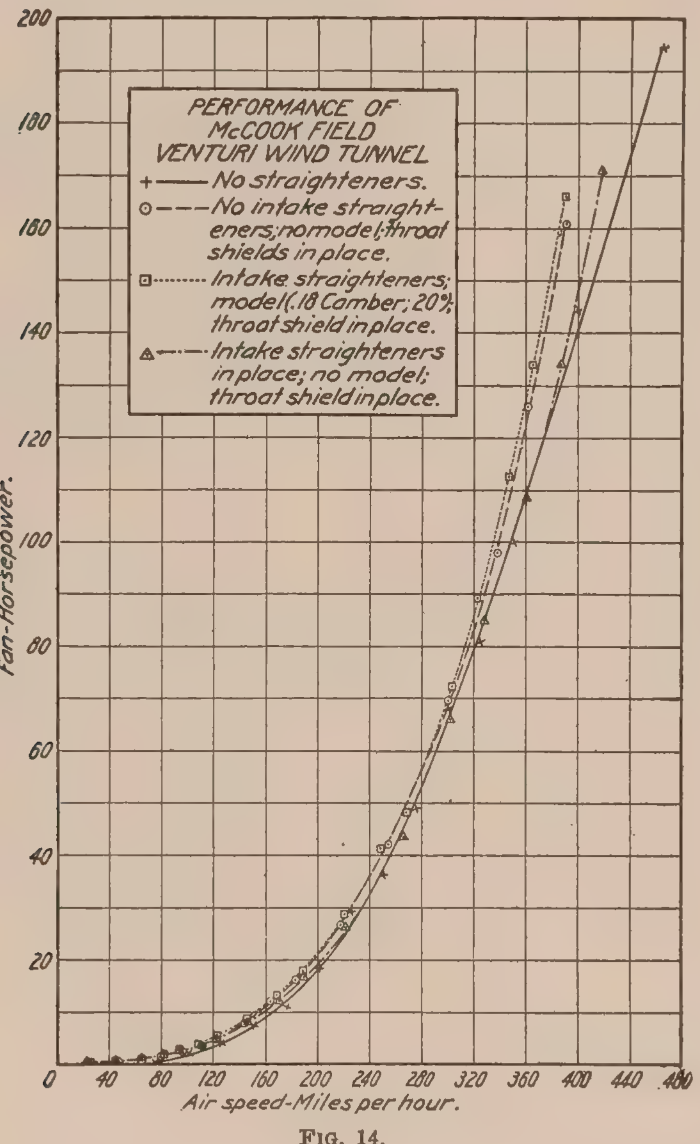


FIG. 14.

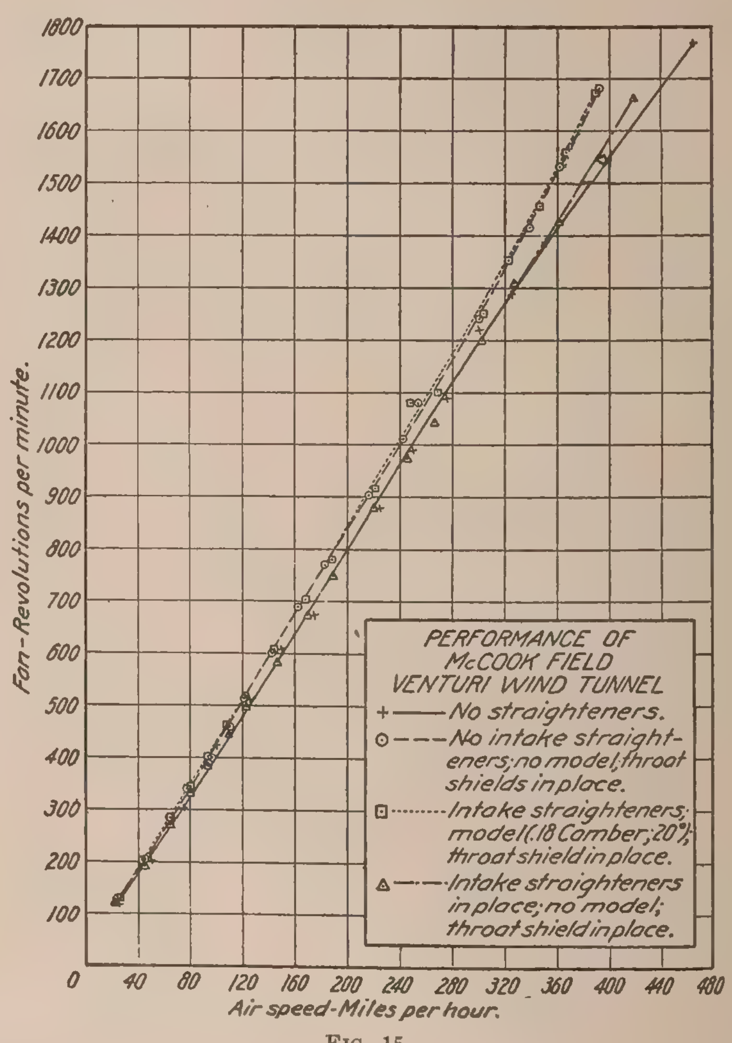
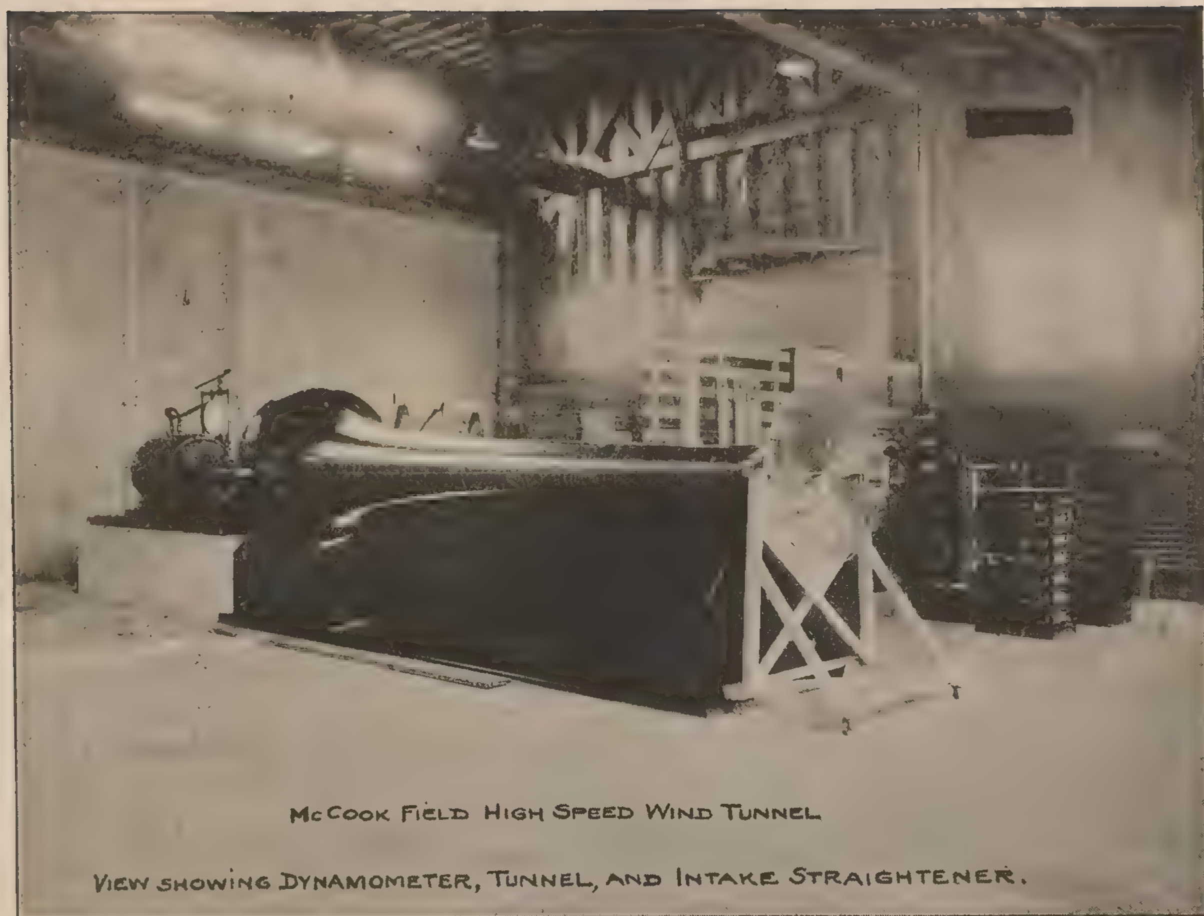


FIG. 15.

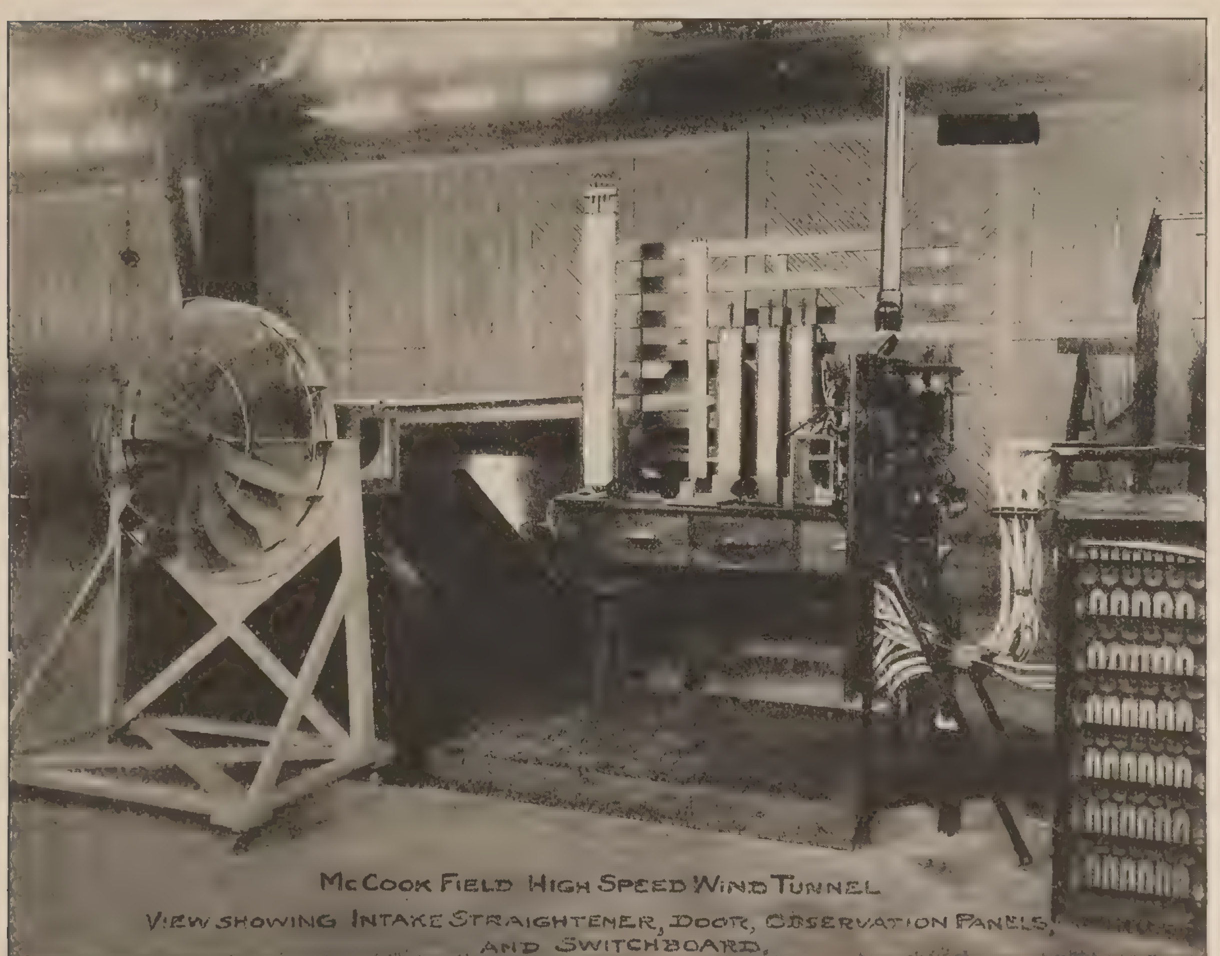
It is to be noted that this ratio differs from the conventional efficiency factor used in wind-tunnel discussions (kinetic energy at throat over fan energy). It is treated in note II of this report. A value of 76 per cent was reached, higher than has been usual in determining aerofoi coefficients in other wind tunnels. The five curves of figures 14 and 15 show the performance of the wind tunnel under different conditions.



McCook Field High Speed Wind Tunnel

View showing Dynamometer, Tunnel, and Intake Straightener.

FIG. 12.



McCook Field High Speed Wind Tunnel

View showing Intake Straightener, Door, Observation Panels, and Switchboard.

FIG. 13.

Performance of McCook field venturi wind tunnel; no straighteners.

(Barometer, 29.55 inches Hg. Room, 62° F. Density, 0.0755 lb./ft.)

Tunnel speed.	Fan speed.	Torque, 63-inch arm.	Horse-power.	Difference at choke.	Inches of water, center section.			Inches of water, exit section.			Ratio of static suction.	
					Static.	Impact.	Difference.	Static.	Impact.	Difference.	Choke to center.	Choke to exit.
Mi./hr.	*R. p. m.*											
25	120	....	....	0.306	0.07	0.00	0.07	0.07	0.13	....	4.28	....
50	205	....	....	1.22	.29	.03	.26	.20	.23	....	4.21	....
75	305	2.0	0.61	2.76	.65	.20	.45	.50	.50	0.00	4.245	....
100	425	5.2	2.21	4.91	1.20	.10	1.10	.90	.70	.20	4.09	....
125	510	8.0	4.08	7.67	1.70	.20	1.50	1.25	1.00	.25	4.53	6.14
150	610	13.0	7.93	11.03	2.50	.50	2.00	1.90	1.60	.30	4.42	5.80
175	675	16.4	11.07	15.02	3.15	.70	2.45	2.30	1.90	.40	4.77	6.53
200	800	23.5	18.80	19.63	4.50	.50	4.00	3.20	2.00	1.20	4.36	6.14
225	880	29.5	25.95	24.86	5.60	.50	5.10	3.90	2.70	1.20	4.44	6.37
250	990	37.0	36.63	30.70	6.60	.60	6.00	5.00	3.00	2.00	4.60	6.07
275	1,090	45.0	49.00	37.15	8.00	1.00	7.00	6.00	4.50	1.50	4.645	6.19
300	1,220	56.0	68.30	44.20	9.50	2.00	7.50	7.70	5.70	2.00	4.652	5.74
325	1,290	63.0	81.20	51.80	11.00	3.00	8.00	8.70	5.70	3.00	4.72	5.95
350	1,390	72.0	100.00	60.10	13.00	3.00	10.00	10.10	....	7.00	4.625	5.95
400	1,550	93.0	144.20	78.60	16.00	4.00	12.00	12.20	....	9.00	4.915	6.44
465	1.770	110.0	194.60	106.60	21.00	....	....	16.50	....	....	5.07	6.46
....	....	....	....	....	....	....	....	....	....	....	[1] 5.10	[1] 6.15

[1] Average.

	Diameter.	Area.		Distance from choke.
	Inches.	*Inches.*²	*Feet.*²	*Inches.*
Choke..........................	14.000	153.94	1.068	0
Center..........................	22.375	393.20	2.732	100
Exit section....................	47.250	1,753.50	12.180	190

The net blade discharge area is 8.38 square feet, which is 7.84 times the area of the choke. With a choke speed of 100 miles per hour, air was noted to leave the propeller at an angle of 45° with the plane of the fan and at an angle of 15° radially from a tangent. The component velocity was noted to be about 47 miles per hour. The average ratio of choke to exit suction throughout the above range from 25 to 465 mi./hr. is 6.5.

NOISE.

Careful study of fan and cone design results not only in reduced losses but also in reduced noise. In the past the noise has been a serious objection to speeds greater than 70 miles per hour in wind tunnels. It may be said that 60 per cent of the roar of any airplane is due to the propeller. For wind tunnel use, the combination of fan and cone adopted has brought about a considerable improvement, as indicated in the following tabulation:

From the operator's position:

The fan is noiseless at..	50
The fan starts to roar at..	60
Conversation is easy at..	125
Conversation is slightly forced at..	155
Conversation is possible 12 inches apart at..	240
Conversation is possible at 4 inches apart at..	300

PRECISION.

The precision of wind-tunnel work in general is dependent in the last analysis upon the velocity readings. By means of adopting the instantaneous reading method, however, the inaccuracies usually to be expected due to velocity fluctuation have been greatly decreased, as was shown at the start of the experiments by study of the velocity graphs. The method was found normally to be very satisfactory, but under abnormal conditions, as for instance when the doors were open and the wind was blowing outside, the tests became impracticable.

The precision of any one reading depends upon the skill of the observers and on the amount of time at their disposal for identifying the fluctuations of their respective instruments. The individual readings of the tests here reported have precision slightly less than could be obtained in the conventional wind tunnel where a honeycomb is used; hence the desirability of a plurality of observations at small velocity increments. The results as plotted show the same order of precision as is reached in the conventional wind tunnel. Moreover, scaling effect of experiments so far exceeds the degree to be predicted from past knowledge of the $\frac{LV}{\nu}$ ratio, that it is not found necessary to question the adequacy of the precision.

In general no difficulty was encountered in checking a given run with different crews and on different days.

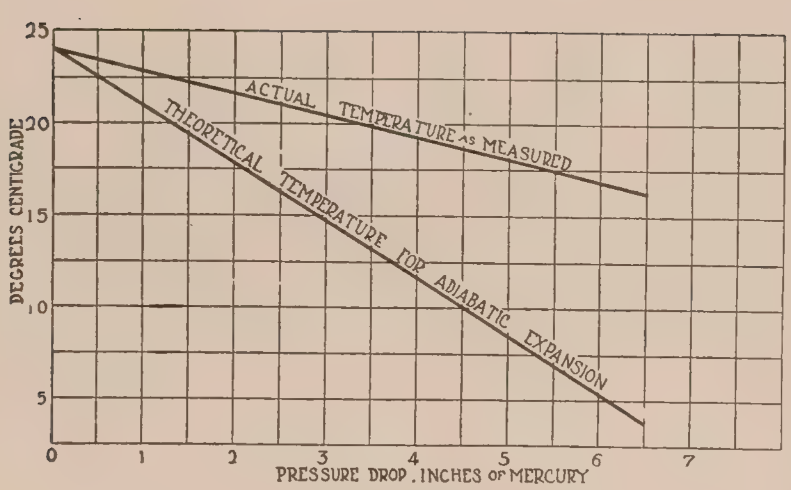


FIG. 17.

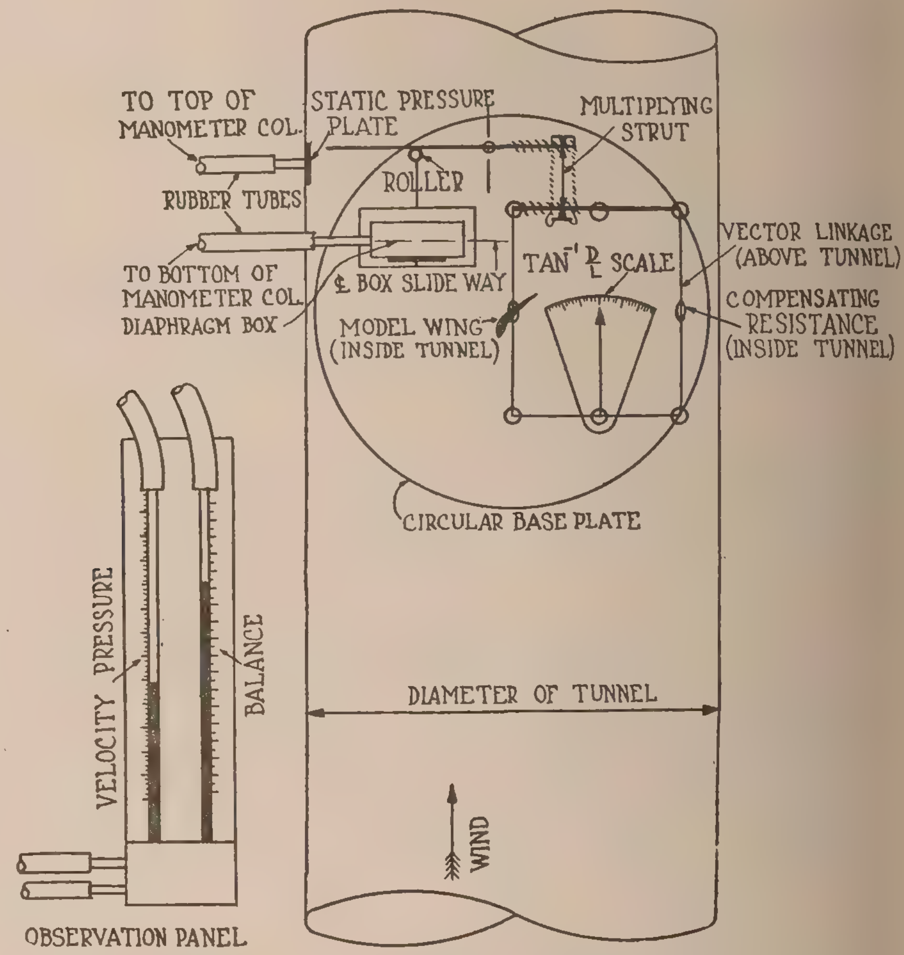


FIG. 18.

The determination of true velocity is dependent upon a knowledge of the temperature and density of the air flowing through the observation section. The temperature is calculated on the assumption that the expansion is adiabatic from the atmospheric pressure to the pressure corresponding to the dewpoint, and is polytropic below the latter pressure. A correct knowledge of throat temperature is, of course, essential; and it is necessary to develop a special method of thermometry for reading it. Present methods are inapplicable to its direct measurement, for a thermometer introduced into the air stream occasions more or less adiabatic compression of the air striking it, with consequent rise of temperature at the point of impact. (See chart, fig. 17.) The most advantageous position for the thermometer is with the bulb downstream, where it is subject chiefly to skin friction rather than impact. Further investigations are being made of the matter.

For the graphs of this report the usual wind-tunnel practice has been followed, wherein the density in the room rather than in the tunnel is used as a basis on which to figure velocity. This has been done for convenience in view of the complicated laws which govern the density of the air in the tunnel itself. The correction when applied does not change the value of the lift coefficient, but changes the corresponding value of the velocity.

FIG. 18½.—TRANSITION TYPE OF AIR FLOW ABOUT ÆROFOIL AT SECOND CRITICAL SPEED (LOOKING ALONG AXIS OF TIP VORTEX).

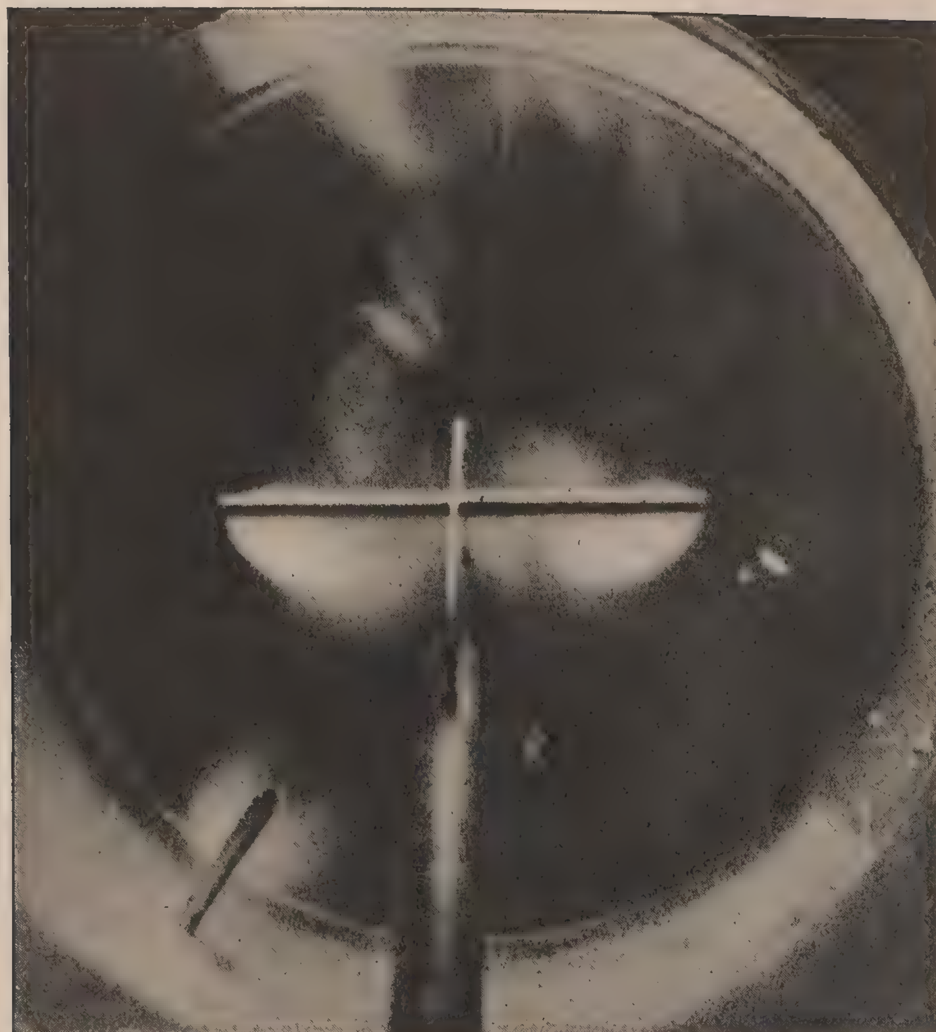


FIG. 19.—AIR FLOW IN TRANSITION STAGE BETWEEN HIGH AND LOW LIFT RÉGIME.

FIG. 20.—FLOW CORRESPONDING TO HIGH LIFT RÉGIME. STABLE UP TO THE CRITICAL SPEED.

It might be desirable to explain how the lift coefficient is calculated, so that it will be apparent that the density does not enter into the calculation. Suppose that h represents the height of the water column corresponding to the velocity head at a velocity V; then, if $\frac{\rho}{g}$ represents the density of the air in slugs, $\frac{\rho}{g} \times V^2 = K_1 h$

If P represents the lift on the model, and A its area, and K_y the absolute lift coefficient, then—

$$P = K_y \frac{\rho}{g} A V^2, \text{ and } K_y = \frac{P}{\frac{\rho}{g} A V^2}$$

Substituting,

$$K_y = \frac{P}{A K_1 \times h}$$

It is evident that this last expression is independent of density, and as this equation is used in calculating the value of the lift coefficients in all cases, the density of the air in the tunnel does not affect the value of the lift coefficient.

METHOD OF SUPPORTING THE AEROFOIL.

The effect of the center support on the lift coefficient is not considered serious. This conclusion is based on experiments run at other laboratories where the effect of the support has been carefully determined; also on a comparison of the present series of experiments with tests made elsewhere on a larger model supported at the end, for the same VL values. The effect of the center support on drag, however, is known to be very large, so that the results on drag have not been given except in a qualitative way.

The one-end spindle used at the National Physical Laboratory can not be utilized under the conditions of these tests. In order to definitely delimit the effect of the supporting member, further developments are proposed wherein this effect will virtually be eliminated from the tests.

VISUALIZATION OF FLIGHT VORTICES.

The method of visualizing air flow, discovered by the writers together with C. P. Grimes, offers a solution to one of the fundamental problems of aerodynamics. This problem is the quantitative empirical measurement of the phenomena of fluid dynamics appertaining to flight and air flow.

The accepted theory upon which flight has its physical basis is purely rational. It has not yet been directly applicable to engineering design, because empirical measurement of flight vortices has never been made. Therefore, the aeronautical engineer's use of aerodynamics is largely according to the cut and try method. He can not, on the drafting board, depart from known shapes, speeds, or sizes; should he wish to do so, he must first build a model and determine the coefficients applicable to his new design.

To illustrate this point, it is only necessary to refer to the simplest case, that of an airplane wing. We can measure the coefficient of force on a small model of this wing to an accuracy of 1 per cent. But we do not know definitely how the accuracy is changed by scaling up to full size, or to full speed. We can not, without tests, predict the change of coefficient to be expected when the wing shape is altered, or the angle of attack, or the position with reference to other surfaces.

Aerodynamical theory will serve practical use when supported by empirical data. In the past flight vortices have never been measured, nor even visualized to a usable extent. Analysis of air flow has been confined to the use of smoke or powder set loose in the air to indicate lines of flow; or of threads used as wind vanes. Or we have been driven to analogies derived from study of fluids of differing viscosity and density, such as water. Or, further, we have sought by measurement of static pressures in the air surrounding a body to deduce the lines of flow. But

these methods have given small encouragement toward the practical application of the vortex theory to engineering use.

The method described herewith depends upon the fact that the moisture in the air condenses out as fog when the temperature is reduced to the dew point, provided there is a solid or liquid nucleus to start the condensation. In the McCook Field wind tunnel the temperature drop is brought about through expansion of the air during acceleration due to 100 inches of water suction. Relative humidity of the atmosphere can be artificially raised if too low. The necessary nucleus for condensation is provided by the model itself.

Flight vortices become readily visible and can be photographed with the aid of searchlights. Several efforts were made to take the pictures with a plate camera, but these were not very successful. Finally a good moving picture was taken and some of the films enlarged. While these films showed up very well on the screen the detail was not very clear in enlargement, so that in addition to the searchlights, which were provided with nitrogen-filled incandescent lamps, two carbon arc lights were set up in order to give a greater amount of blue light. The results obtained from the motion-picture camera with the carbon arc lighting were fairly satisfactory, and a number of enlargements from the motion-picture films are reproduced in this paper. Figure 21 is an enlargement of a moving-picture film looking downstream. It is inferior to visual observation, the vortices showing as below the model, due to parallax. To the naked eye they are in line with the wing tips and are clean-cut, perfect circles. They extend downstream a distance of several dozen chord lengths from the rear corner of each wing tip, enlarging in diameter as the distance increases, and converging slightly in the horizontal plane. (See figs. 1 and 30.) In the vertical plane the tip vortex axis takes a decided downward angle, intermediate between the horizontal and the line of travel of the flat sheet of edge vortices.

NOTE.—For an excellent mathematical discussion of the shape and arrangement of the tip vortices and the trailing vortices see Report No. 28 of the National Advisory Committee for Aeronautics (United States). On page 44 the author, Dr. George DeBothezat, has given some sketches, figures 38 and 39, which show in an interesting way that the axis of the tip vortices is intermediate between the direction of movement and the sheet of trailing edge vortices.

For corroboration of the vortex phenomena at slow speeds where condensation is not visible, steam jets have been useful. When of proper saturation, such a jet introduced in the tunnel intake provides a good indication of the flow lines, and is superior to the conventional smoke jet.

Adequate analysis of the flight vortices will be made in future with special apparatus at present under consideration. The shape, size, and direction of the tip vortices can be easily noted and seem fairly susceptible of measurement. The periodic run of the edge vortices is too quick for recognition by the naked eye or even for identification by the moving-picture camera; it requires stroboscopic analysis.

The observed vortices differ for different aerofoil set-ups and different speeds. For instance the observed tip vortex at 18° has less than one-half the diameter manifest at 8°, while the line of edge vortices is less noticeable at 18°.

Again, the character of the general vortex phenomenon undergoes remarkable change at the critical speed. In the high-lift régime the general shape is like a trough whose floor (edge-vortex sheet) slopes downward from the trailing edge and whose walls (tip vortices) are increasingly high as the distance downstream increases. The cross section is roughly like a shallow U.

At higher speeds, however, in the low-lift régime the observed phenomenon is suddenly altered. Following out the above homely analogy, we may imagine that the "walls" of the trough remain substantially as before. The "floor," however, splits longitudinally, curls upward, and extends the two limbs, now free, to a point well above the level of the tip vortices. Figures 26 and 2 are enlargements of two motion-picture exposures which were intended to record the sequence of the change. These photographs are not altogether satisfactory, and are therefore supplemented by the two sketches of figure 34.

FIG. 21.—ON THE RIGHT SIDE THE FLOW HAS GONE OVER TO THE INEFFICIENT TYPE. THIS IS STABLE AT SPEEDS HIGHER THAN THE CRITICAL SPEED.

FIG. 22.—THE MODEL SUPPORT HAS BEEN MOVED OUT OF LINE WITH THE TUNNEL AXIS TO MAKE ROOM FOR THE CAMERA.

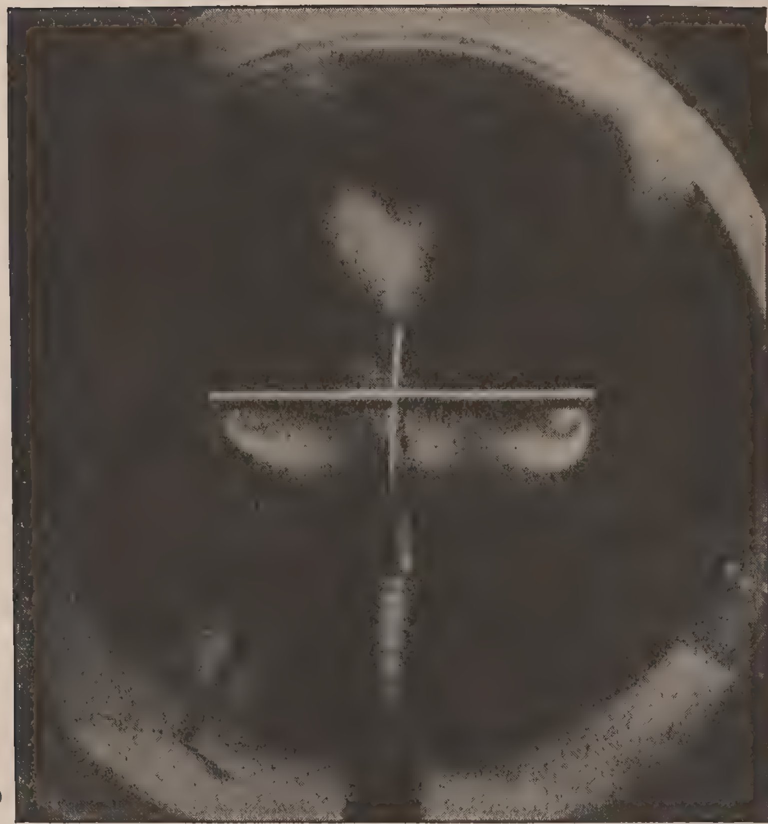


FIG. 23.—AIR FLOW AT SECOND CRITICAL SPEED. NOTE THE DISTURBANCE BEHIND CENTER OF ÆROFOIL CAUSED BY SUPPORTING ROD.

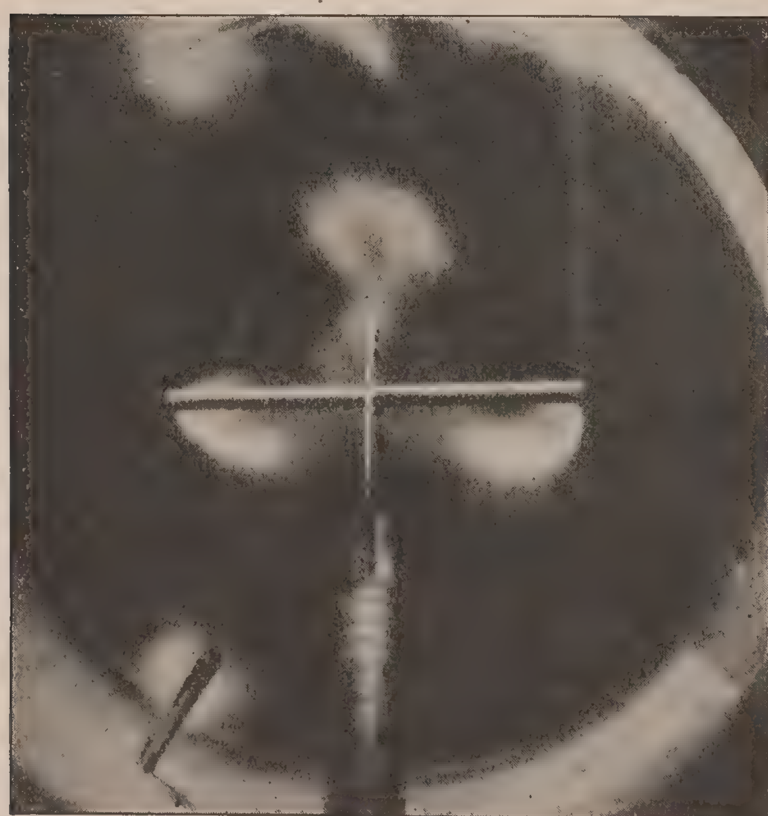


FIG. 24.—IN THIS PICTURE THE DISTURBED AIR FLOW BEHIND THE SUPPORT HAS TAKEN A DIFFERENT SHAPE. NOTE TENDENCY TO MERGE WITH DISTURBANCES AT PERIPHERY OF TUNNEL.

FIG. 25.—DARK CENTER OF RIGHT TIP VORTEX SHOWN IN PERSPECTIVE.

FIG. 26.—LOW LIFT RÉGIME ON LEFT HALF SPAN: HIGH LIFT ON RIGHT.

FIG. 27.—THIS EXPOSURE IS ADJACENT IN THE CINEMA FILM TO THAT OF FIG. 26.

FIG. 28.—THE DISTURBANCE BEHIND THE CENTER SUPPORT SHIFTS BACK AND FORTH VERY RAPIDLY AND AT THE SAME TIME CHANGES ITS SHAPE.

The shape and movement remind one in a striking manner of the lightning observed when electrical discharge takes place in the sky.

FIG. 29.—UNSYMMETRICAL CONDITIONS OF AIR FLOW ON THE TWO HALF SPANS.

FIG. 30.—VIEW FROM ABOVE THE MODEL; LINE OF SIGHT ABOUT 30° FROM LINE OF FLOW.

FIG. 31.—EFFICIENT TYPE OF FLOW STABLE AT LOW SPEEDS.

FIG. 32.—UNSTABLE FLOW OCCURRING ONLY AT THE CRITICAL SPEED.

Note the cusp halfway out the right wing.

FIG. 33.—INEFFICIENT TYPE OF FLOW, STABLE AT HIGH SPEEDS.

Note the stream extending vertically upward halfway out the right half span.

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Figure 34–a is a diagram of the end view of the high-lift phenomenon as it appears distinctly to the naked eye. The right-hand side of the aerofoil in figure 26 approximates this condition, the left-hand side having already gone over to low-lift flow. Figure 34–b is a diagram of the low-lift phenomenon; the left-hand side of aerofoil in figure 2 shows this fairly well, the right-hand side being in a transition stage. Figure 33 shows the low-lift flow better than figure 2, but is dim. Figure 1 is a three-fourths view of the low-lift flow, and also represents other features mentioned above.

FIGURE - 34-a

HIGH LIFT REGIME

An interesting variation of the flight vortices is furnished by replacing the aerofoil by a flat disk normal to the wind. Here the phenomenon can be seen as a "streamline" fog surface, converging toward a point half a dozen diameters downstream.

Figure 19 illustrates the distance above the aerofoil to which the flight vortex phenomena may extend and shows their tendency to merge with other whirls attributable to the wind tunnel walls. The extent of the phenomena may be four or more chord lengths above the aerofoil; this further develops investigation made by the writers in 1911, when it was shown experimentally that the air flow above a wing is disturbed to a distance of at least four chord lengths. (See "The Center of Pressure Travel on Airplane Surfaces and Birds' Wings," Mass. Inst. of Technology, 1911.)

FIGURE - 34-b

LOW LIFT REGIME

FIG. 34.

Reference is also made to the work of J. R. Pannell, dealing with experimental evidence as to the extent of circulation about an aerofoil.

VISUALIZATION OF UNOBSTRUCTED AIR FLOW.

When the model is removed the vortices and eddies of flow through the unobstructed throat may be observed by looking into the intake or through the transparent shield of the observation section. The condensation is more pronounced behind the impact tube and thermometer bulb than elsewhere, since these are obstacles to the flow and therefore constitute nuclei for condensation. A projecting cotter pin one-sixteenth inch high at the wall causes a perfect vortex, which shows up against the white foggy background as a black circle.

The general appearance of the air flow, which may be considered typical of all air flow, is as follows: A cross section at the throat shows a seething mass of fog specters, denser at the walls than at the center, though occasionally the entire disk fills up with fog to the point of opaqueness. The specters have in the cross-sectional plane a gentle movement like the flame of an alcohol stove, showing the constant readjustment of equilibrium. Vortices and S-shaped whirls continually form and, after moving about, lose themselves in the general confusion. In a diagonal view they take the appearance of long, foggy fibers, stretching down the tunnel like wooden moldings. The axes of whirl are, of course, longitudinal. Under proper humidity and lighting conditions the whole becomes a beautiful iridescent sight, violet and purple hues predominating.

REPORT No. 83.

WIND TUNNEL STUDIES IN AERODYNAMIC PHENOMENA AT HIGH SPEED.

PART III.

MODEL TESTS ON PROPELLER AEROFOILS.

By F. W. Caldwell and E. N. Fales.

The six aerofoils adopted for tests were of 6 inches length, 1 inch chord, and 0.1 to 0.2 camber. The cross-sectional shape, as shown in figure 35, is that upon which the engineering

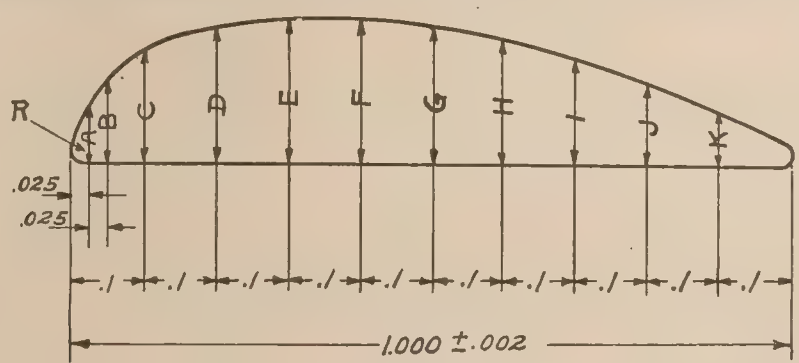


Fig. 35.

Max. thickness.	A	B	C	D	E	F	G	H	I	J	K	R
0.08	0.032	0.047	0.063	0.076	0.080	0.079	0.076	0.069	0.059	0.044	0.028	0.008
.10	.041	.059	.079	.095	.100	.099	.095	.087	.074	.056	.035	.010
.12	.049	.070	.094	.114	.120	.118	.114	.104	.088	.067	.042	.012
.14	.057	.082	.110	.133	.140	.138	.133	.121	.103	.078	.049	.014
.16	.065	.094	.126	.152	.160	.158	.152	.139	.118	.089	.056	.016
.18	.073	.106	.142	.171	.180	.178	.171	.156	.133	.100	.063	.018
.20	.082	.118	.158	.190	.200	.198	.190	.174	.148	.112	.070	.020

division has standardized for propeller use. As was expected, a broad interpretation of results was facilitated by the diversity of cambers, angles, and speeds available. Thanks to the discovery of the method of air-flow visualization it was possible to give a physical interpretation to the balance readings. In the discussion of results which follows it will therefore be noted that the phenomena will be referred to interchangeably in terms either of vortex formation or of lift-and-drift.

The outstanding conclusion to be drawn from the tests is that we have more than one régime of air flow to deal with in aerofoil study, and that these régimes are separated by conditions of discontinuity. The characteristics usually associated in aeronautical engineering with a practical aerofoil do not apply outside the small range of cambers, speeds, and angles utilized in flight. Beyond this range the flow about the aerofoil no longer produces the familiar results in terms of lift and drag, but becomes analagous to the flow about a body of irregular shape. Efficient lift of an aerofoil is only a single case of several distinct aerodynamic phenomena resulting from air flow past a solid body. When the speed of air flowing past an aerofoil increases there is first a régime of relatively low-lift effect, then at higher speeds an efficient lift effect such as applies in flight, then at still higher speeds a drop back to a second low-lift effect. As the angle or camber increases the high-lift régime becomes discontinuous and is succeeded by the low-lift régime; the transition point is spoken of in conventional graphs as the "critical, or stalling, angle," or the "burble point."

All of the sections show, at certain angles, two speeds at which the flow is unstable and discontinuous. At the point of discontinuity occurring at the lower speed increase of speed shows an increased lift coefficient and a decreased drag coefficient so that the lift-drag ratio is enormously increased. At the point of discontinuity occurring at the higher speed increase of speed shows a decreased lift coefficient and an increased drag coefficient, so that the lift-drag ratio is enormously decreased. Thus these sections have a definite speed range at each angle within which the flow is efficient and produces a high lift-drag ratio and a fairly constant lift coefficient. It may be called the régime of high L/D, and includes the phenomena appertaining to practical flight. This speed range has been definitely measured for the higher angles, but apparently it goes beyond the speed obtainable in the tunnel for the lower angles.

Before this series of experiments was started the attention of the writers was called by Mr. Orville Wright to the fact that he had obtained two distinct values of the lift coefficient and the drag coefficient in testing certain aerofoils at certain angles of attack and constant speed. Mr. Wright also called our attention to the fact that the high value of the lift coefficient always corresponds to the low value of the drag coefficient, and the low value of the lift coefficient always corresponds to the high value of the drag coefficient. The two values are now clearly proved by the "dew-point" method of observation to represent distinctly different types of flow.

It will therefore be noted that the K_y curves are drawn discontinuous to correspond with discontinuity in the type of air flow. In some cases the graphs show a third curve intermediate between the high-régime curve and the low-régime curve. This third intermediate line undoubtedly represents different types of flow on the two parts of the aerofoil. This is possible because of the center support which divides the aerofoil into two parts. At the point of discontinuity corresponding to the second critical speed the lift reading becomes unsteady; the flow phenomena become unstable and jump from one type to another, until the new form is established. Figures 2, 19, 23 show the unstable flow of the transition, figures 33 and the left side of 25 show the flow of the final low-lift régime.

The discovery of this second critical speed is one of the novel and significant features of the experiments. Simultaneous observation of the balance and of the flight vortices made the discovery possible, affording proof that the two types of flight vortices can be identified with the two values of the lift coefficient, one belonging to a high L/D régime, the other to a low L/D régime.

With the general remarks above it is now proper to make a detailed study of the characteristics of the various curves submitted in this report. The three variables are (1) angle of attack, which is defined as the angle between the flat undersurface of the aerofoil and the direction of the wind; (2) camber, that is, the maximum thickness of aerofoil divided by the chord; (3) velocity, which is given in miles per hour and refers to pressures and temperatures of standard air. It must be understood that all these three factors of angle, camber, and speed affect the régime of air flow, and it is insufficient to explain the air flow régimes in terms of either factor separately from the others. With this reservation certain analyses are given herewith as to the various effects of these factors. Refer to figures 36 to 41.

The effect of the three factors in determining whether the low or high lift régime obtains is shown by the following three examples:

1. As example of angle change, compare the 0.12 camber aerofoil at 17° and 20°; the high K_y régime does not appear for 20°.

2. As example of camber change compare the 0.10 camber aerofoil with 0.18 camber at 15°, the former case showing high-lift régime, the latter case showing low-lift régime. It must be kept in mind that the unit of "angle" is an arbitrary one, and for its proper definition must depend upon the camber; hence, it is to be expected that the effect of angle and camber changes will be analogous.

3. As example of velocity change, see 0.12 camber aerofoil at +17°; it shows high K_y only at intermediate velocities.

Effect of Angle.

The effect of change in angle may be studied in terms of the slope of the K_y-angle curve. For low velocity all cambers show flat curves at the larger negative angles; for high velocity flatness occurs beyond the burble point. As the speed is increased there is a general tendency for the slope of the K_y-angle curve to become less. This is more marked the thicker the section tested, until finally the section having a camber ratio of 0.2, when run at the speed of 450 miles per hour, shows practically no variation in lift coefficient between the angles of $-8°$ and $+10°$. The lift on this section is therefore nearly independent of the angle of attack between the angles of $-8°$ and $+10°$ at this speed.

The slope is greatest where the régime of high K_y values obtains, namely, in the angles of the flight region. The slope is small, and in general positive, at high negative angles and at positive angles beyond the burble point. Negative slopes are not general. It seems that the conventional practice of plotting K_y-angle curves beyond the burble point as continuous curves of negative slope is unjustified, in view of the discontinuity evidenced after the burble point has been reached.

Effect of Camber.

Within the limits observed, K_y increases with the camber up to the burble point, after which it decreases.

Effect of Speed.

Speed change has an effect which is more prominent for the larger cambers than for the small. The effect of speed change on the K_y-angle curves is prominent at small negative angles, where the slope is greater as the speed is less. As a corollary to this it may be noted in general that within the limits observed, the K_y-speed curves approach zero at great speeds.

Maximum value of K_y.

The maximum attainable value of K_y seems limited, no matter what combination of speed, angle, and camber we choose. Thus the highest K_y (0.65) is reached in the 0.14 camber aerofoil at 15° and 150 miles per hour. The lowest (-0.19) is reached in the 0.12 camber at $-10°$ and 50 miles per hour. The 0.10 camber was not tested at a greater negative angle than $-8°$. Had the tests on this section been carried to $-10°$ it would have shown a lift coefficient algebraically less than -0.19. Study of this lowest value can be completed only when a flat-plate experiment shall be added to the series. (The negative angles give the equivalent of an aerofoil having flat upper surface and convex under surface.)

Shift of the Angle of Zero Lift.

One of the interesting discoveries of these tests is the shift of the angle of no lift with speed. This is not so noticeable in the thinnest section, where the change is only about 3° between speeds of 50 and 450 miles per hour. The shift is progressively greater with the thicker aerofoils, however, until in the case of the aerofoil with 0.2 camber the angle of zero lift shows a change of about 18° between speeds of 50 and 400 miles per hour.

As the velocity approaches zero the angle at which the lift coefficient is zero appears to approach zero degrees as a limit.

PRACTICAL SIGNIFICANCE OF THE RESULTS.

This no-lift angle shift is significant, as regards experimental no-thrust pitch. A number of model propeller tests have shown this no-thrust pitch to increase with the speed. It is understood that the experimental no-thrust pitch of a propeller does not correspond to the pitch at which the sections are striking the air along the line of zero lift. The two, however, are rather close together as far as the angle of attack is concerned and the shift of the angle of zero lift explains the change in experimental pitch very well.

These tests have at the outset proved useful in clearing up certain problems in propeller design. A practical example concerns the design of the reversible propeller, where the aerofoil in reverse position has to work at a negative angle of attack. Reference to the lift coefficients for the 0.12 camber aerofoil show that at a velocity of 450 miles per hour this aerofoil has very little negative lift, K_y being about -0.1 at an angle of $-12°$ from the chord. It would obviously be impossible to get much thrust out of an aerofoil of this kind in a reverse position. In order to have a satisfactory thrust it is apparent that the aerofoil must be modified in such a way as to depart from the flat underface.

The discovery of the critical speed of these aerofoils has an interesting bearing on the possibilities of high-speed propellers.

We have found by practical experience that if we do not go below a value of V/ND of 0.65, we get a very fair propeller efficiency. As we have gradually increased the speed of our planes we have gone on increasing the revolutions per minute of the engine and the diameter of our propellers so that the value of V/ND has remained about the same for the great majority of propellers in actual service.

We have always assumed that there was no limit to this development aside from the characteristics of the plane and engine. That is, we have made the assumption that we could double our propeller speed just as soon as we were able to double our plane speed and strengthen our engine enough to stand the stresses involved.

It now appears, however, as though there is a limit to propeller speed aside from the value of V/ND, or, to use more familiar terms, aside from pitch ratio.

Unfortunately, even the speed obtainable in the McCook Field wind tunnel is not great enough to measure the limiting velocity for relatively thin sections when set at low angles. Consequently we are only able to infer that it exists from extrapolation of the curve of critical speeds.

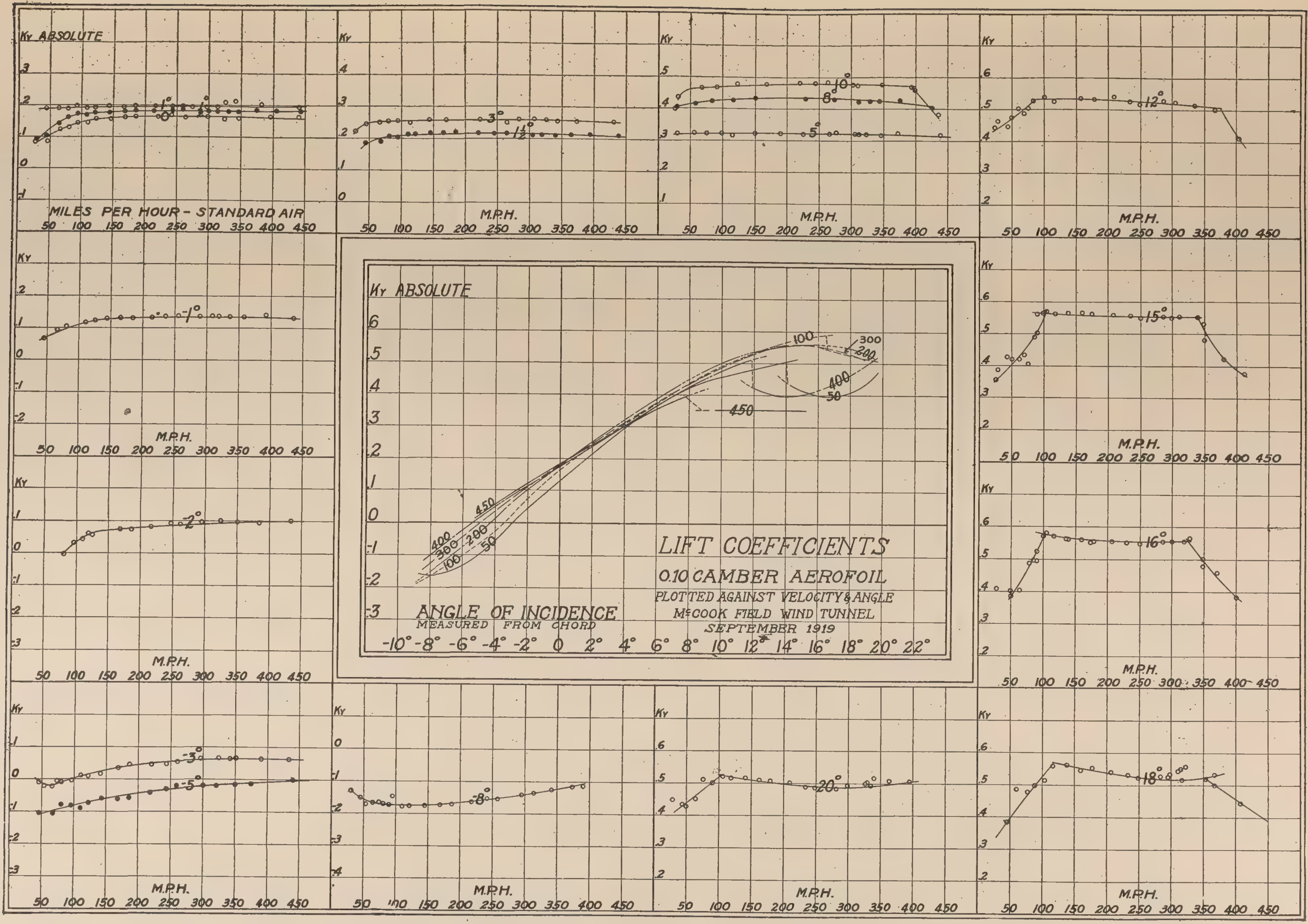

KY ABSOLUTE
MILES PER HOUR - STANDARD AIR
M.P.H.
LIFT COEFFICIENTS
0.10 CAMBER AEROFOIL
PLOTTED AGAINST VELOCITY & ANGLE
McCOOK FIELD WIND TUNNEL
SEPTEMBER 1919
ANGLE OF INCIDENCE
MEASURED FROM CHORD

Fig. 36.

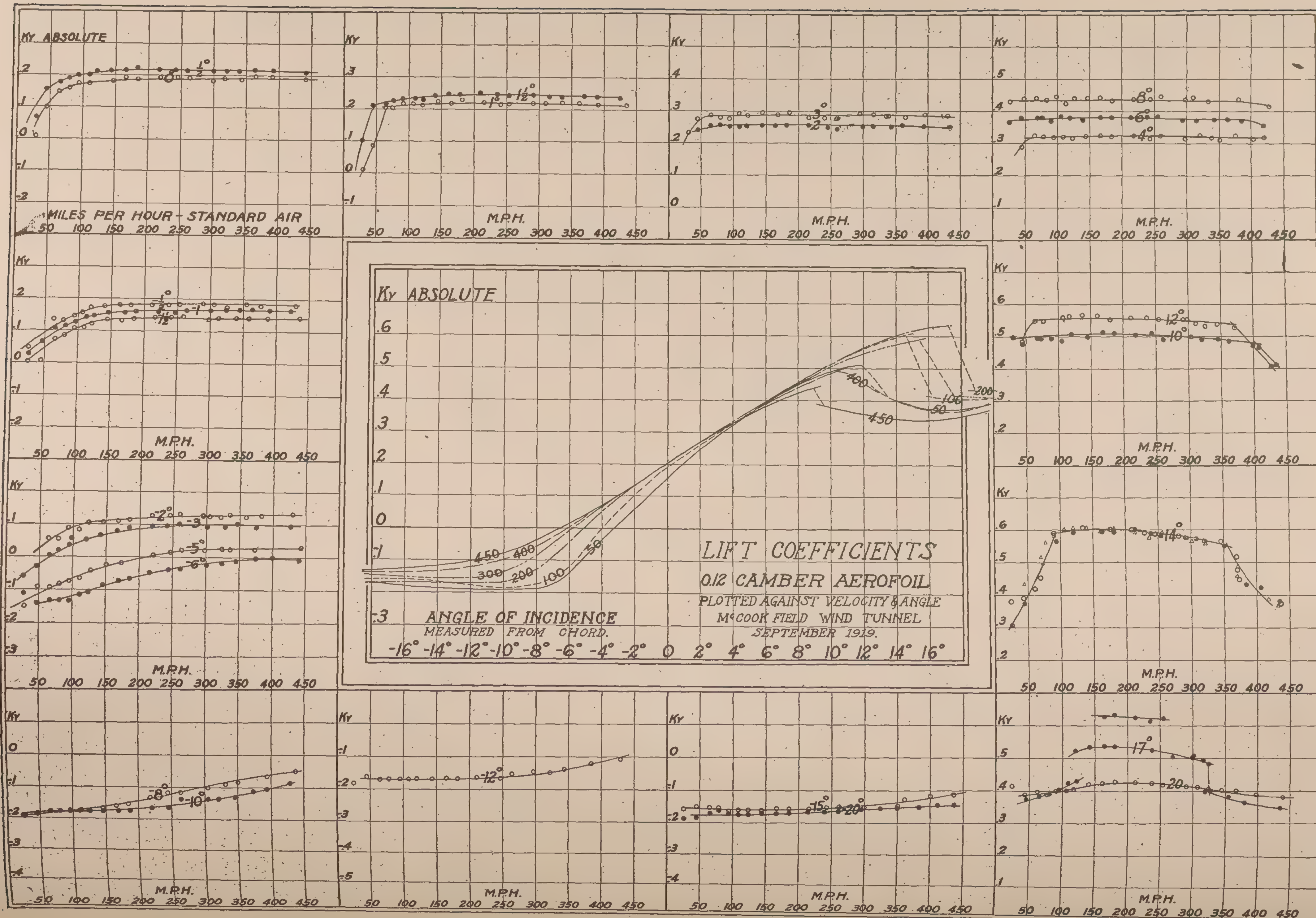

KY ABSOLUTE
MILES PER HOUR - STANDARD AIR
M.P.H.
50 100 150 200 250 300 350 400 450
LIFT COEFFICIENTS
0.12 CAMBER AEROFOIL
PLOTTED AGAINST VELOCITY & ANGLE
McCOOK FIELD WIND TUNNEL
SEPTEMBER 1919.
ANGLE OF INCIDENCE
MEASURED FROM CHORD.
-16° -14° -12° -10° -8° -6° -4° -2° 0 2° 4° 6° 8° 10° 12° 14° 16°

FIG. 37.

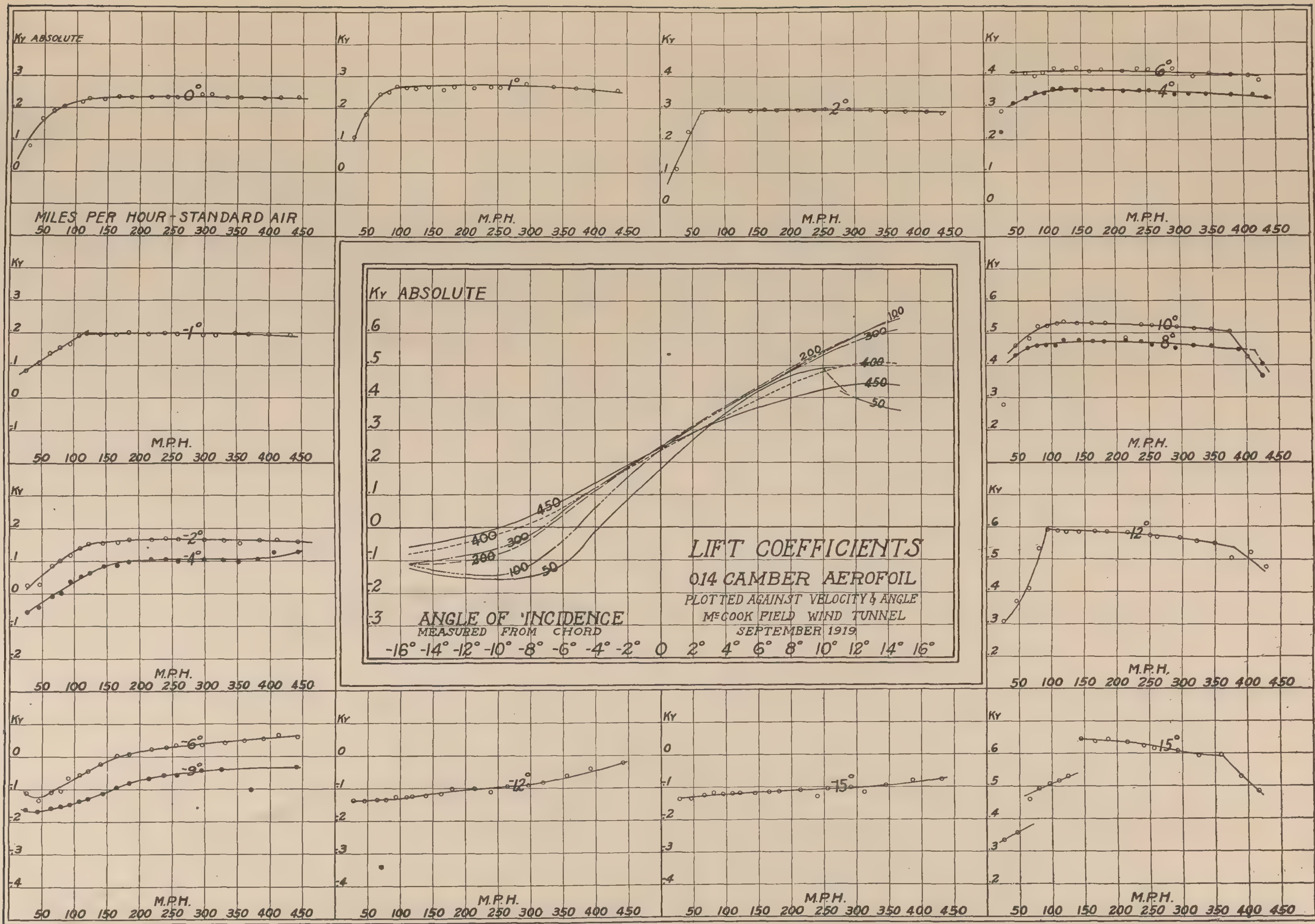

KY ABSOLUTE
MILES PER HOUR-STANDARD AIR
M.P.H.
50 100 150 200 250 300 350 400 450
0°
1°
2°
4°
6°
8°
10°
12°
15°
-1°
-2°
-4°
-6°
-9°
-12°
-15°
KY ABSOLUTE
LIFT COEFFICIENTS
0.14 CAMBER AEROFOIL
PLOTTED AGAINST VELOCITY & ANGLE
McCOOK FIELD WIND TUNNEL
SEPTEMBER 1919
ANGLE OF INCIDENCE
MEASURED FROM CHORD
-16° -14° -12° -10° -8° -6° -4° -2° 0 2° 4° 6° 8° 10° 12° 14° 16°

FIG. 38.

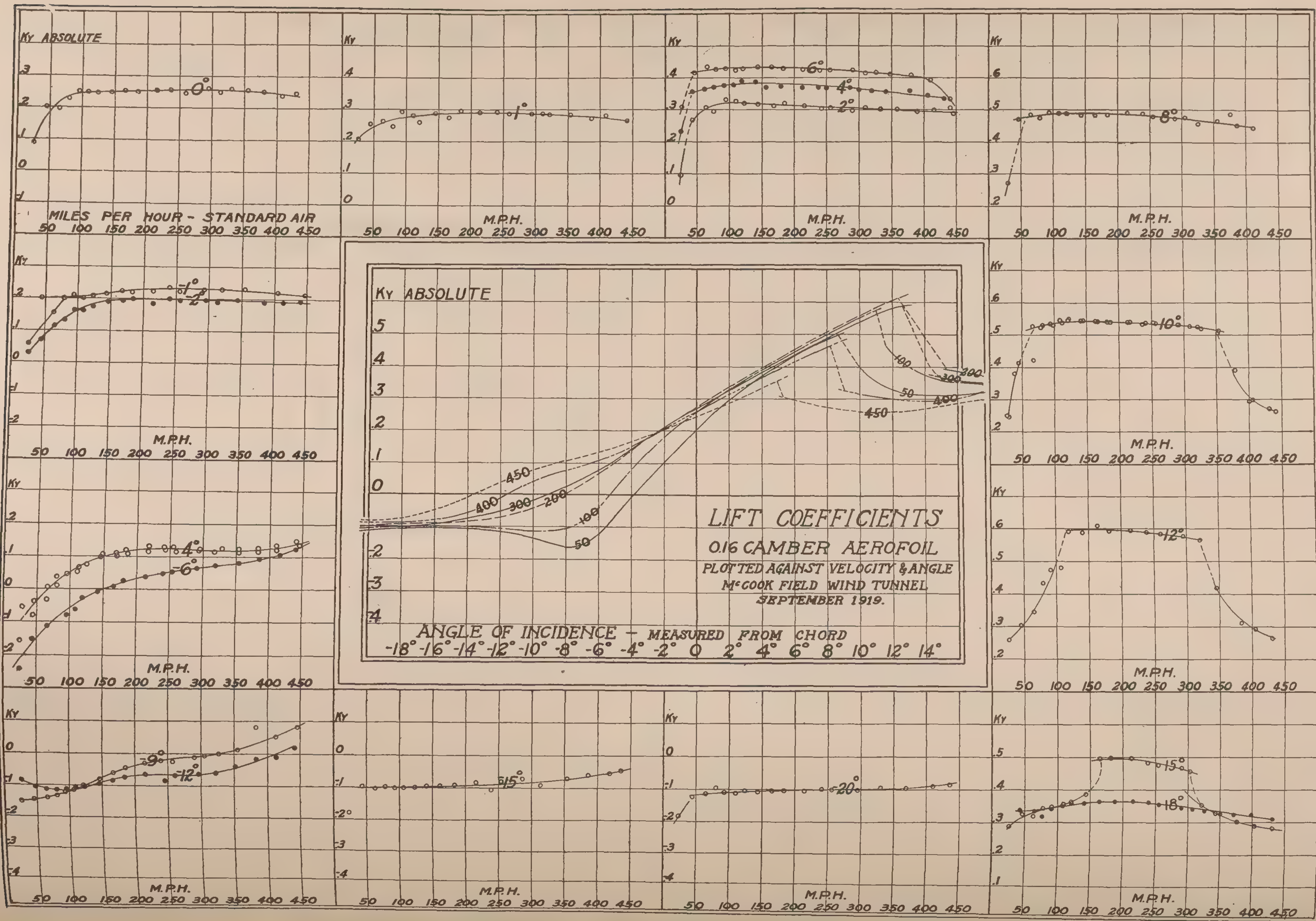
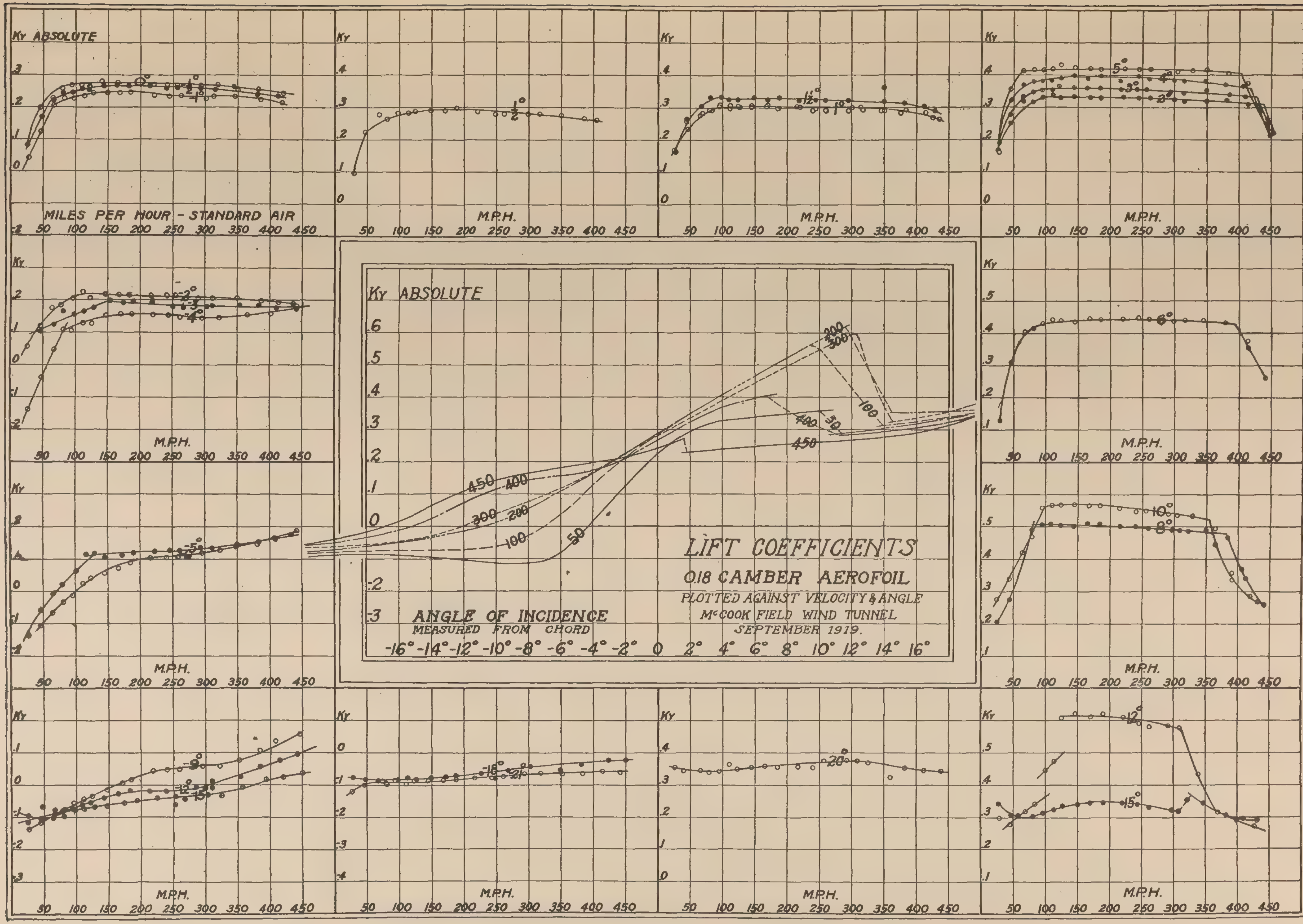

LIFT COEFFICIENTS
0.16 CAMBER AEROFOIL
PLOTTED AGAINST VELOCITY & ANGLE
McCOOK FIELD WIND TUNNEL
SEPTEMBER 1919.
KY ABSOLUTE
ANGLE OF INCIDENCE – MEASURED FROM CHORD
-18° -16° -14° -12° -10° -8° -6° -4° -2° 0 2° 4° 6° 8° 10° 12° 14°
MILES PER HOUR – STANDARD AIR
M.P.H.
50 100 150 200 250 300 350 400 450

FIG. 39.



LIFT COEFFICIENTS
0.18 CAMBER AEROFOIL
PLOTTED AGAINST VELOCITY & ANGLE
McCOOK FIELD WIND TUNNEL
SEPTEMBER 1919.
Ky ABSOLUTE
ANGLE OF INCIDENCE
MEASURED FROM CHORD
-16° -14° -12° -10° -8° -6° -4° -2° 0 2° 4° 6° 8° 10° 12° 14° 16°
MILES PER HOUR - STANDARD AIR
M.P.H.
50 100 150 200 250 300 350 400 450
Ky

FIG. 40.

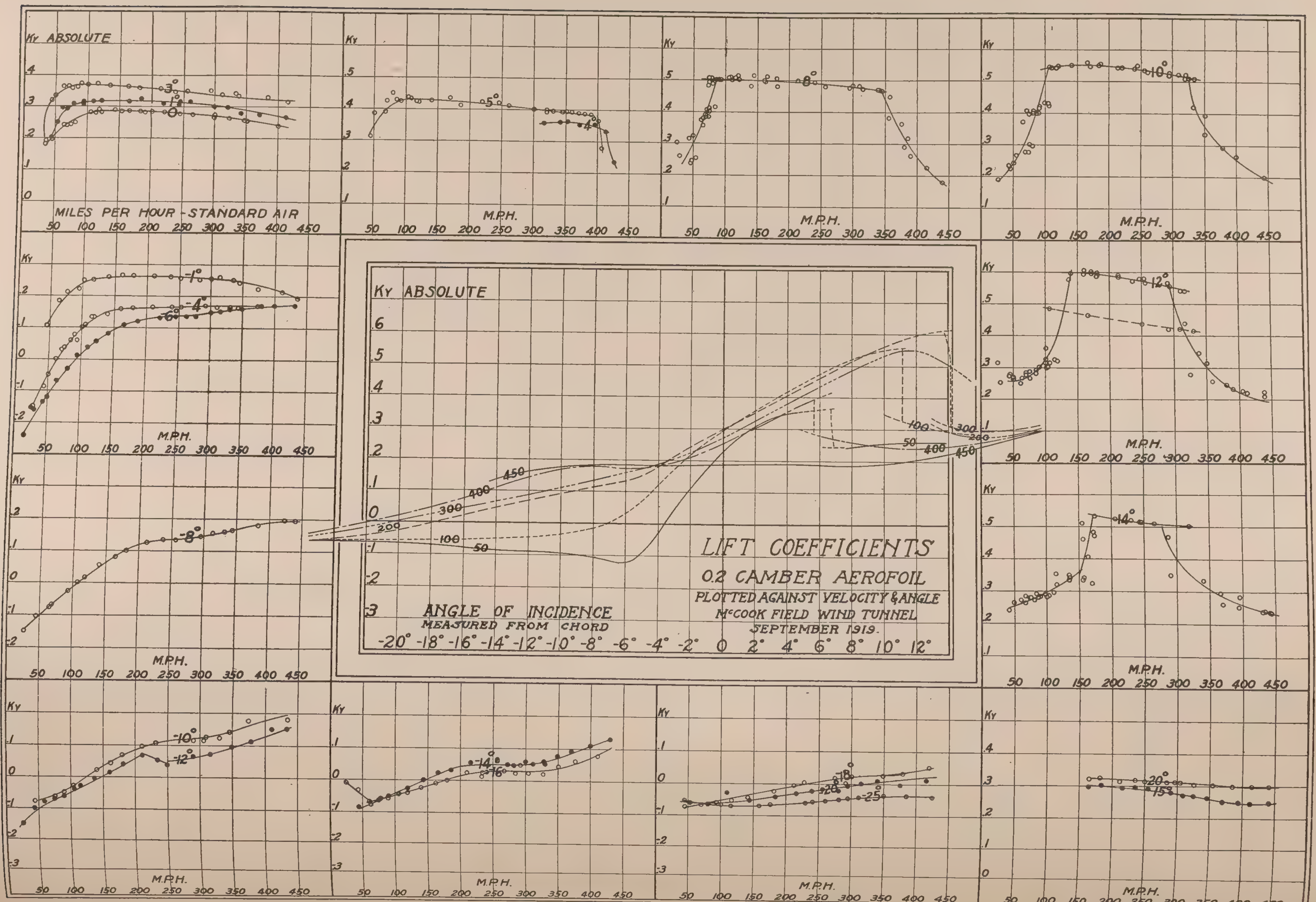
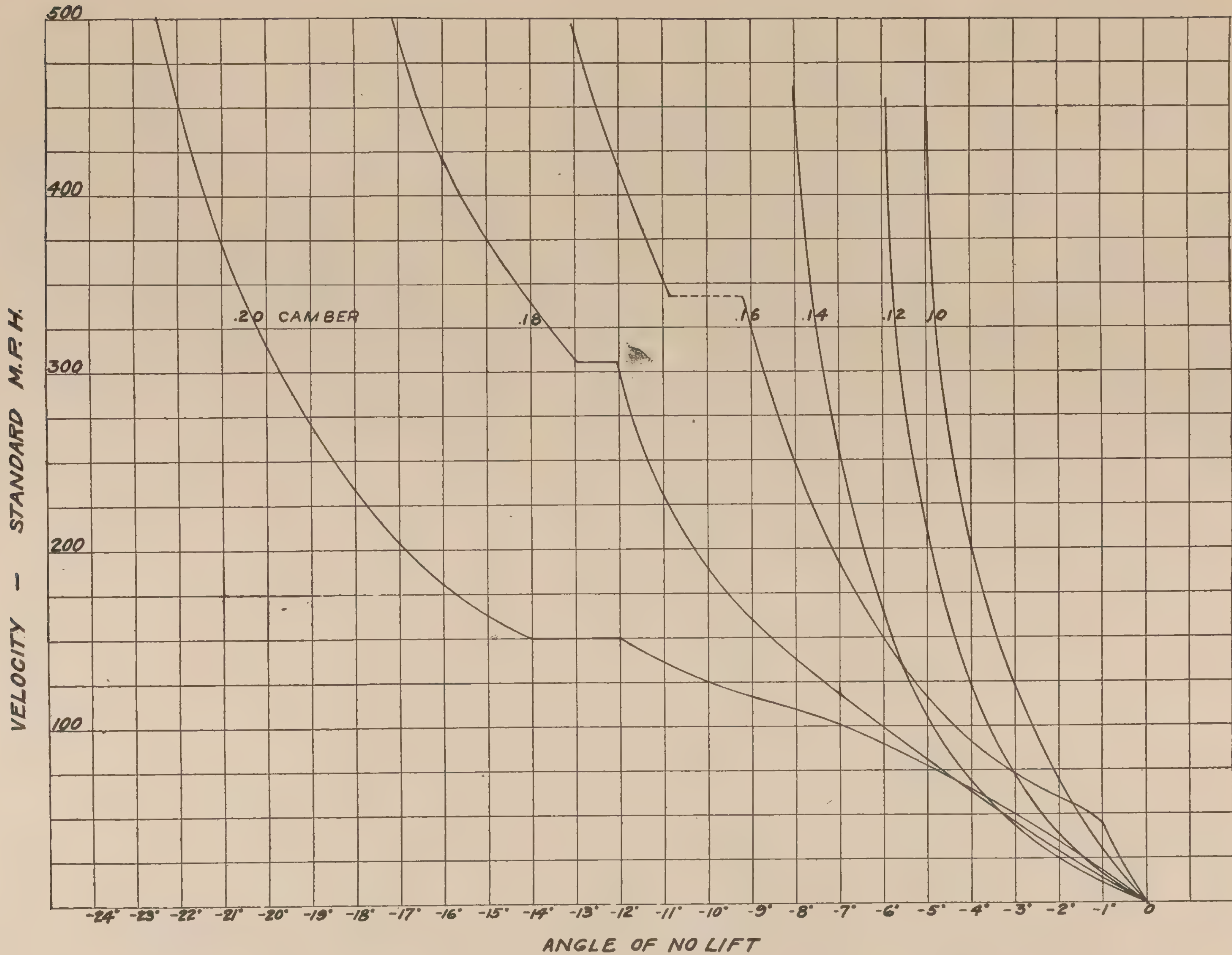

LIFT COEFFICIENTS
0.2 CAMBER AEROFOIL
PLOTTED AGAINST VELOCITY & ANGLE
McCOOK FIELD WIND TUNNEL
SEPTEMBER 1919.
KY ABSOLUTE
ANGLE OF INCIDENCE
MEASURED FROM CHORD
-20° -18° -16° -14° -12° -10° -8° -6° -4° -2° 0 2° 4° 6° 8° 10° 12°
MILES PER HOUR - STANDARD AIR
M.P.H.
50 100 150 200 250 300 350 400 450

FIG. 41.



500
400
300
200
100
.20 CAMBER
.18
.16
.14
.12
.10
VELOCITY — STANDARD M.P.H.
-24° -23° -22° -21° -20° -19° -18° -17° -16° -15° -14° -13° -12° -11° -10° -9° -8° -7° -6° -5° -4° -3° -2° -1° 0
ANGLE OF NO LIFT

FIG. 42

REPORT No. 83.

NOTE 1.

A FUNDAMENTAL PROPERTY OF VISCOUS FLUIDS.

By George de Bothezat.

One of the most fundamental requirements of modern hydrodynamics is an exact statement of the conditions that determine a given state of flow. It is well known that under some conditions several different types of flow are compatible with the equations of hydrodynamics and the boundary conditions, but that in reality only certain types of flow establish themselves. In the preamble to my paper, An Introduction to the Study of the Laws of Air Resistance of Aerofoils,[1] I have already mentioned briefly this state of things. The valuable experimental investigations of Mr. Caldwell and Mr. Fales on flow phenomena give me the opportunity to explain myself more fully on this subject.

My conception of this question is the following: The continuity of the fluid flow is determined in modern hydrodynamics only by the *equation of continuity.* This condition constitutes only a *necessary,* but not *sufficient,* condition for the continuity of flow. Helmholtz called attention to the fact that the discontinuity of the tangential components of the fluid velocity along certain curfaces is compatible with the equation of continuity—that is, this equation does not exclude the possibility of the gliding of one fluid layer over the other. But if such gliding can start, when and where is it to start? We are here brought directly to the question, When will the fluid remain as a continuous fluid mass, and when can a disintegration of the fluid mass become possible? Or, in other words, What are the general mechanical conditions under which a continuous fluid mass will remain as such? It is easy to conceive that it is quite impossible to assume that a continuous fluid mass will always remain continuous under all conditions; and the equation of continuity alone is far from being sufficient to determine the flow-continuity. It seems to me that in modern hydrodynamics this last important fact has been somewhat overlooked.

Many well known facts and experiments show us that the disintegration or partition of a continuous fluid mass into separate parts or particles is a common phenomenon; for instance, the forming of foam at the top of waves; or simply the dividing of a fluid mass into several masses. For example, we pour a part of the water contained in a glass into another glass and so get two volumns of water out of one. Now in all cases when we observe the division of a fluid mass, this phenomenon occurs, so to say, by virtue of speed. That is, only when the speed gradient inside the fluid mass exceeds a certain value can we have fluid separation, or, more generally, can the formation of surfaces of discontinuity inside the fluid take place. The analogy to elastic phenomena in solid bodies is easily perceived. In elastic bodies it is the displacements that are the criterions of continuity. In fluid bodies it is the speed gradients. In a solid body if two of its particles are caused to separate a certain distance from each other the body breaks at the point considered. In a fluid body two particles can be separated to any distance inside the fluid mass without destroying the fluid continuity if only the displacement is made without exceeding in the fluid mass certain values of the velocity gradient. But, if in the course of such a displacement too high values of the velocity gradient are reached, the separation of the fluid particles will take place. In other words, in the same way as critical values of displacement bring the breakage of solid bodies, in a similar way critical values of velocity gradient bring fluid breakage. We do not have to do here with fluid breakage in the literal sense of the word, because as soon as the velocity gradient drops below its critical value the fluid particles, if in contact, will rebuild a continuous mass; so that we have to do merely with fluid separation. We are thus brought to an understanding of the phenomenon of fluid separation, it being a question of the velocity gradient.

[1] Washington, D. C., 1919, National Advisory Committee for Aeronautics, Report No. 28.

As soon as the foregoing conception has been reached and we wish to include these phenomena in the domain of dynamics; that is, to apply the concept of *force* to the analysis of these facts, we are immediately brought to the necessity of considering the stress distribution inside the fluid, which stress must depend only on the distribution of the velocity gradient.

The whole question appears as follows: A continuous mass of fluid must be considered as remaining continuous only if, in addition to the equation of continuity being satisfied, the stresses inside the fluid mass do not exceed certain critical values. These critical stresses depend on the distribution of the velocity gradient inside the fluid. When the critical stress is once reached, the fluid mass must disintegrate—that is, surfaces of discontinuity must be formed. If it is the tension stress that first reaches its critical value, simple division of the fluid will take place. If it is the shearing stress that first reaches its critical value, we can have either a simple gliding of a fluid layer over another or the formation of foam; the last on account of the reciprocal property of shearing stresses, which requires that the critical shearing stresses shall appear simultaneously in two orthogonal directions.

The type of flow that establishes itself in each case is such that the stresses inside the fluid are everywhere lower than their critical values. If the critical value is reached somewhere, the flow type changes in such a way that the stresses drop below the critical values. The stress distribution inside a fluid thus appears, in addition to stability requirements, as a fundamental criterion of the flow type.

Let us now formulate the above conception in a more precise form. I will here give only a general sketch of the problem and consider only the most simple case. It will be assumed that the reader is acquainted with the methods of stress analysis and the equation of motion of viscous fluids. According to Stokes for the case of the motion of a viscous fluid parallel to a plane the stresses developed in an element of a fluid are

$$T_{xy}=\mu\left(\frac{\partial u}{\partial y}+\frac{\partial v}{\partial x}\right)$$

$$N_x=-p+2\,\mu\frac{\partial u}{\partial x}$$

$$N_y=-p-2\,\mu\frac{\partial v}{\partial x}$$

Where N_x and N_y are the normal components of the resultant stress in the surface elements normal to the X and Y axes, and T_{xy} is the tangential component of the resultant stress in the same elements, equal for both elements as result of the reciprocal property of the shearing stresses; p is the pressure at the point considered and μ the viscosity coefficient. We will consider the fluid as incompressible; that is,

$$\frac{\partial u}{\partial x}+\frac{\partial v}{\partial y}=O$$

and the motion as steady; that is,

$$\frac{du}{dt}=u\frac{\partial u}{\partial x}+v\frac{\partial u}{\partial y}$$

$$\frac{dv}{dt}=u\frac{\partial v}{\partial x}+v\frac{\partial v}{\partial y}$$

The vortex intensity ω at the point considered is given by

$$2\omega=\frac{\partial v}{\partial x}-\frac{\partial u}{\partial y}$$

Let us refer the motion of the fluid at the point considered to what I call the natural fluid coordinates τ and ν whose axes are directed along the tangent to the streamline through the point considered and the principal normal to the same streamline at the same point.

If we designate by V the value of the speed of the fluid particle at the point considered and by V_τ and V_ν the components of V along the τ and ν axes, we find

$$u = V_\tau = V$$

$$v = V_\nu = 0$$

and

$$2\omega = \frac{\partial V_\nu}{\partial \tau} - \frac{\partial V_\tau}{\partial \nu}$$

It is easy to prove that [2]

$$\frac{\partial V_\nu}{\partial \tau} = \frac{V}{\rho}$$

and thus

$$\frac{\partial V_\tau}{\partial \nu} = \frac{V}{\rho} - 2\omega$$

Accordingly,

$$\frac{\partial u}{\partial y} + \frac{\partial v}{\partial x} = \frac{\partial V_\tau}{\partial \nu} + \frac{\partial V_\nu}{\partial \tau} = 2\left(\frac{V}{\rho} - \omega\right)$$

And consequently for the tangential stress we find

$$T_{\tau\nu} = 2\mu\left(\frac{V}{\rho} - \omega\right)$$

On the other hand,

$$\frac{du}{dt} = u\frac{\partial u}{\partial x} = V\frac{\partial V}{\partial \tau} = \frac{dV}{dt}$$

on account of $v = V_\nu = 0$; and thus

$$\frac{\partial u}{\partial x} = -\frac{\partial v}{\partial y} = \frac{1}{V}\frac{dV}{dt}$$

and consequently for the normal stresses we find:

$$N_\tau = -p + \frac{2\mu}{V}\frac{dV}{dt}$$

$$N_\nu = -p - \frac{2\mu}{V}\frac{dV}{dt}$$

We thus see that the principal stress axes are making with the τ, ν axes an angle α given by

$$\tan 2\alpha = \frac{V\left(\frac{V}{\rho} - \omega\right)}{\frac{dV}{dt}}$$

The maximum shearing stress is equal to

$$T = 2\mu\sqrt{\left(\frac{V}{\rho} - \omega\right)^2 + \frac{1}{V^2}\left(\frac{dV}{dt}\right)^2}$$

And the principal normal stresses are equal to

$$N_1 = -p + T$$
$$N_2 = -p - T$$

[2] See the papers of George de Bothezat, edited by the National Advisory Committee for Aeronautics, General Theory of Blade Screws, p. 86, or Introduction into the Study of Laws of Air-resistance of Aerofoils, p. 61.

The maximum shearing stress T thus appears to depend on the actual speed V of the fluid element considered, the radius ρ of the curvature of its path, its actual tangential acceleration dV/dt and the vortex intensity ω at the point considered. When $(V/\rho-\omega)=O$ the principal stress axes coincide with the (τ, ν) axes; when $dV/dt=O$ the principal stress axes bisect the (τ, ν) axes. In the general case the principal stress axes can have any position relative to the (τ, ν) axes.

Discontinuity will appear in the fluid motion if T or N_1 exceed certain critical values. For N_1 the critical value is very close to $N_1=O$ or $T=p$. If it is T that first reaches a critical value T_c this will give rise to a shearing discontinuity. We will have the formation of surfaces or volumes of discontinuity. But the last as generally admitted are unstable, and will pass over into certain systems of vortices, which can, as seen from the foregoing formulae, release the stresses.

When the expressions for stresses are used at their critical values the viscosity coefficient can not be considered as constant and must take a special critical value at the moment the stresses reach their critical values.

If we designate by σ the cross section of a flow hole, it will be easy to see that

$$\frac{\partial V}{\partial \tau}=-V\frac{d\sigma}{\sigma d\tau}$$

and as

$$\frac{dV}{dt}=V\frac{\partial V}{d\tau}$$

we find

$$\frac{dV}{dt}=-V^2\frac{d\sigma}{\sigma d\tau}$$

and introducing the last value in the expression of the maximum shearing stress we find

$$T=\frac{2\mu V}{\rho}\sqrt{\left(1-\frac{\rho\omega}{V}\right)^2+\rho^2\left(\frac{d\sigma}{\sigma d\tau}\right)^2}$$

The last relation gives the expression of the maximum shearing stress in function of V, ω and the geometrical configuration of the flow.

Karman, at the end of his investigations on the mechanics of fluid resistance,[3] mentions the great importance that the determination by theory of all elements of the vortex systems he studies would have. It is only by the study of the stress distribution inside fluids that the answer to those questions can be found. In the study of lubricating oils, the conception of the critical shearing stress is also of first importance and constitutes the most important property of the oil as lubricating agent.

The determination of the values of the critical stress inside fluids constitutes an experimenta problem of first importance.

In the flow experiments of Fales and Caldwell, when we observe the change of the type of flow, which can be seen in a very clear way by the ''dew-point'' method, as I myself have several times witnessed, this means that the critical stresses have been reached and the flow goes to another type, with another vortex distribution, which releases the stresses.

All the foregoing is only a very short sketch containing merely the statement of the question of a very broad and quite new domain of hydrodynamics. I hope to have some day the opportunity of publishing all the results to which I have been brought in the study of these questions.

GEORGE DE BOTHEZAT.

DAYTON, OHIO, *October, 1919.*

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[3] The paper of Karman's here referred to is given translated as Note IV at the end of Dr. de Bothezat's Introduction to the Study of Laws of Air Resistance of Aerofoils. (See p. 75.)

REPORT No. 83.

NOTE II.

THE EFFICIENCY OF A WIND TUNNEL.

By GEORGE DE BOTHEZAT.

So far as I know no definition of the efficiency of a wind tunnel has been given. The ratios used by different authors for the comparative appreciation of different wind tunnels were purely conventional estimates of the degree of perfection. The whole difficulty resides here in the fact that for a wind tunnel it is not the *engine efficiency* that is involved but the *efficiency of an irreversible cycle.* I will first illustrate by an example exactly what I mean.

Let us consider a rubber ball that undergoes the following cycle: The ball is dropped from a height H_0 with an initial speed equal to zero. At the level H the ball reaches the ground with a speed v, rebounds and rises again. On account of different resistances the ball will not generally reach the original height H_0. To enable the ball to do so, let us suppose that at the height H_1, when the ball has the speed v_1, an impulse is given to it that brings it to such a height H_2 with a speed v_2, that the ball can afterwards reach its original height H_0 and from there start falling over again. In this example the ball undergoes an irreversible cycle which consists in dropping from a height H_0, getting a certain kinetic energy equal to $\frac{1}{2}mv^2$, and reaching again the height H_0 by aid of the work done by the impulse.

When the ball drops, its potential energy measured by the height H_0 goes over into kinetic energy. When the ball reaches the level H, its kinetic energy $\frac{1}{2}mv^2$ will generally be less than the work $(H_0-H)\ mg$ done by gravity, on account of possible resistances. When the ball rises its kinetic energy goes into potential energy. The work Π equal to

$$\Pi=\tfrac{1}{2}m\ (v_2^2-v_1^2)+(H_2-H_1)\ mg$$

represents the work absorbed by the resistances, which has to be communicated to the ball to allow it to reach its original height H_0.

Thus, in the irreversible cycle considered, the energy put into play is equal to $L=\frac{1}{2}mv^2$. The losses are

$$\Pi=\tfrac{1}{2}m\ (v_2^2-v_1^2)+(H_2-H_1)\ mg.$$

The losses expressed in per cent of the energy put into play are equal to

$$\frac{\Pi}{L}.$$

And the efficiency η of our irreversible cycle comes out equal to

$$\eta=1-\frac{\Pi}{L}=1-\frac{\frac{1}{2}m\ (v_2^2-v_1^2)+(H_2-H_1)\ mg}{\frac{1}{2}mv^2}$$

If in the fall we had no losses, we would have

$$\tfrac{1}{2}mv^2=(H_0-H)\ mg.$$

If in addition we assume $v_2=v_1$ and introduce the notations

$$H_0-H=h_0\ ;\ \ H_2-H_1=h_0-h$$

we find

$$\eta=\frac{h}{h_0}.$$

The foregoing examples presents a complete analogy to the phenomena that take place in a wind tunnel. Let us consider a wind tunnel having an open circuit and fitted with a cone, at

the end of which is disposed a suction fan. The outside air has a pressure p_0—analogous to the height H_0 of our ball—and a speed zero. The air is sucked into the wind tunnel and in the throat we have a pressure p—analogous to the ground level height H—and an airspeed v. In the cone the air stream expands. The pressure increases and the speed drops. And just in front of the fan we have a pressure p_1—analogous to the height H_1—and a speed v_1. The fan sucks the air and brings it, immediately behind the fan, to a pressure $p_2>p_1$ and a speed v_2 generally very close to v_1. Finally by diffusion of the air stream, behind the fan, into the free atmosphere, the airspeed v_2 is lost and the pressure p_0 is reached again. We thus see that the air in the wind tunnel undergoes an irreversible cycle which consists in taking the air at the pressure p_0, bringing it to the pressure p and speed v and then back again to the pressure p_0, the losses being compensated by the fan. The energy put into play in this case is equal to

$$L=\Sigma\tfrac{1}{2}\delta\Delta Sv^3.$$

Where ΔS is an annulus of the throat cross section, δ is the density of the air and the sum Σ is taken over the whole throat cross section. If we designate by ΔB the drop in the value of the Bernouilli constant just up to the front of the fan and by ΔB^1 the drop behind, the losses in the wind tunnel flow phenomena come out equal to

$$\Pi=\Sigma\Delta S_1v_1\,(\Delta B+\Delta B^1)+\Sigma\tfrac{1}{2}\delta_1\Delta S_1v_1^3.$$

And the *efficiency of the wind tunnel* considered, comes out equal to

$$\eta=\frac{L-\Pi}{L}=1-\frac{\Sigma\Delta S_1v_1\,(\Delta B+\Delta B^1)+\Sigma\tfrac{1}{2}\delta_1\Delta S_1v_1^3}{\Sigma\tfrac{1}{2}\delta\Delta Sv^3}.$$

Let us take the ideal case of a wind tunnel for which $\Delta B+\Delta B^1=O$. For such a case we will find

$$\eta=1-\frac{\tfrac{1}{2}\Sigma\delta_1\Delta S_1v_1^3}{\tfrac{1}{2}\Sigma\delta\Delta Sv^3}$$

If in addition we assume v and v_1 constant in the corresponding cross-sections, then, on account of flow continuity, expressed by $\delta_1\Delta S_1v_1=\delta\Delta Sv$, we find

$$\eta=1-\frac{v_1^2}{v^2}=1-\frac{\delta^2S^2}{\delta_1^2S_1^2}$$

The values of δ and δ_1 can be calculated to a first approximation by assuming the whole process to be adiabatic. The last expression represents thus the efficiency of an ideal wind tunnel, that is, the limit that a real wind tunnel can not reach.

In some wind tunnels the losses Π are so great that $\Pi>L$, then the efficiency comes out negative; this only means that the wind tunnel considered is so poor that the losses exceed the energy put into play.

Let us now designate by ρ the efficiency of the fan and by L_m the power output of the motor that drives the fan. We will then have

$$\Pi=\rho L_m$$

The efficiency of the wind tunnel then becomes

$$\eta=1-\frac{\rho L_m}{L}$$

The *total efficiency of the wind-tunnel-fan system* is equal to

$$\eta^1=\frac{L-L_m}{L}=1-\frac{\Pi}{\rho L}$$

If we designate by Δp the pressure difference that the fan maintains on its two sides, the useful work done by the fan comes out equal to

$$\Sigma \Delta p \Delta S_1 v_1 = \Sigma \Delta S_1 v_1 (\Delta B + \Delta B^1) + \Sigma \tfrac{1}{2} \delta \Delta S_1 v_1^3 = \Pi$$

thus

$$\Delta p = (\Delta B + \Delta B^1) + \tfrac{1}{2} \delta v_1^2$$

considering

$$v_1 \cong v_2$$

I will remark here that the main part of the losses in a wind tunnel are made up of the losses in the cone and the kinetic energy of the air leaving the wind tunnel. For the whole wind-tunnel-fan system the fan losses have to be added.

One of the most delicate parts of the wind tunnel problem is the phenomenon of the flow expansion in the cone. The author of this note is of the opinion that if the fluid were ideal, no expansion could take place, and that the expansion obtained in wind tunnel cones is exclusively due to fluid viscosity. This brings us to think that improvement of wind tunnels could be reached by providing some special arrangements in the cone, which would oblige the stream to expand.

The foregoing can easily be extended to wind tunnels with closed circuit.

This short note has to be considered only as a short sketch of the question of the wind-tunnel efficiency, which gives merely the conceptional part of the problem. The author hopes to have some day the opportunity to treat more completely the wind tunnel problem.

GEORGE DE BOTHEZAT.

DAYTON, OHIO, *October, 1919.*

REPORT No. 84

DATA ON THE DESIGN OF PLYWOOD FOR AIRCRAFT

BY ARMIN ELMENDORF
Engineer in Forest Products Laboratory

REPORT No. 84.

DATA ON THE DESIGN OF PLYWOOD FOR AIRCRAFT.

By Armin Elmendorf.

Forest Products Laboratory.

PURPOSE OF THE STUDY.

This report makes available data which will aid the designer in determining the plywood that is best adapted to various aircraft parts. It gives the results of investigations made by the Forest Products Laboratory of the United States Forest Service at Madison, Wis., for the Army and Navy Departments, and is one of a series of reports on the use of wood in aircraft prepared by the Forest Products Laboratory for publication by the National Advisory Committee for Aeronautics.

The object of the study was to determine, through comprehensive tests, the mechanical and physical properties of plywood and how these properties vary with the density, number, thickness, arrangement of the plies and direction of grain of the plies. While the data were sought primarily and immediately with a view to obtaining information needed by aircraft designers, the results have a broader field of application.

USE OF PLYWOOD IN AIRPLANES.

Plywood is being used extensively in airplanes for fuselage sides, bulkheads, engine bearers, wing rib webs, gusset and thrust plates, flooring, diaphragms, and at times for partially covering wings, in particular at the leading edges. In some machines stabilizer, elevator, and rudder surfaces are covered with thin plywood. Its use as a substitute for linen in covering wings has, however, not yet found favor, chiefly on account of the excess weight over linen.

From the standpoint of general engineering design the selection of veneer species and thickness introduces elements quite distinct from those involved in the design of an ordinary structural member of wood. More variables are involved, for in addition to the properties of the various species there are added unique properties due to number of plies and thickness and direction of the grain of the various plies. For the designer of aircraft certain further and special considerations enter into the problem. In the first place, strength with a minimum weight is required, while in the design of most stationary structural members weight is a minor consideration. Again, the forces acting on the different parts of an airplane are usually very complex, and both their magnitude and direction can in many cases only be approximated. The position and magnitude of the loads for which stationary structural members must be designed are, on the other hand, usually known with greater precision.

The complexity of the forces acting on airplane parts usually makes the designer's problem one of determining relatively superior constructions rather than of exact computation of required dimensions. Nevertheless, the actual size of some plywood parts of an airplane may be worked out with a reasonable degree of accuracy by using the strength data included in this bulletin. An example of comparatively exact design is afforded in the construction of large trussed wing ribs in which it is desired to know the dimensions of the tension members of wood. The table of tensile strength of veneer will serve for this purpose, although the details of fastening also require consideration.

DEFINITION OF PLYWOOD.

Much confusion has been caused by the indiscriminate use of the terms "veneer" and "plywood." The former term should be restricted to the relatively thin sheets of wood cut with special veneer machinery from the surface of a log revolving in a massive lathe or by slicing or sawing from the face of a log, known, respectively, as rotary, sliced, and sawed veneer. "Plywood," on the other hand, refers to the combination of several plies or sheets of veneer glued together, usually so that the grain of any one ply is at right angles to the grain of the adjacent ply or plies.

PROPERTIES OF ORDINARY WOOD COMPARED WITH PLYWOOD.

Wood, as is well known, is a nonhomogeneous material, with widely different properties in the various directions relative to the grain. This difference must be recognized in all wood construction, and the size and form of parts and placement of wood should be such as to utilize to the best advantage the difference in properties along and across the grain. It is the strength of the fibers in the direction of the grain that gives wood its relatively high modulus of rupture and tensile and compressive strength parallel to the grain. Were it a homogeneous material such as cast iron, having the same strength properties in all directions that it has parallel to the grain, it would be unexcelled for all structural parts where strength with small weight is desired. As it is, the tensile strength of wood may be 20 times as high parallel to the grain as perpendicular to the grain, and its modulus of elasticity from 15 to 20 times as high. In the case of shear the strength is reversed, the shearing strength perpendicular to the grain being much greater than parallel to the grain. The low parallel-to-the-grain shearing strength makes the utilization of the tensile strength of wood along the grain difficult, since failure will usually occur through shear at the fastening before the maximum tensile strength of the member is reached.

The large shrinkage of wood across the grain with changing moisture content may introduce distortions in a board that decrease its uses where a broad, flat surface is desired. The shrinkage from the green to the oven-dry condition across the grain for a flat-sawed board is about 8 per cent and for quarter-sawed board about 4½ per cent, while the shrinkage parallel to the grain is practically negligible for most species.

It is not always possible to proportion a solid plank so as to develop the necessary strength in every direction and at the same time utilize the full strength of the wood in all directions of the grain. In such cases it is the purpose of plywood to meet this deficiency by cross banding, which results in a redistribution of the material.

In building up plywood a step is made in obtaining equality of properties in two directions, parallel and perpendicular to the edge of a board. The greater the number of plies used for a given panel thickness, the more nearly homogeneous in properties is the finished panel. Thus, in an airplane engine mounting made of 15-ply veneer, the mechanical properties of the panel parallel and perpendicular to the grain of the faces are almost the same. Broadly speaking, what is gained in one direction is lost in the other. For a very large number of plies we may assume that the tensile strength in the two directions is the same and that it is equal to the average of the parallel-to-the-grain and perpendicular-to-the-grain values of an ordinary board. This is not always exactly true, since the maximum stress of the plies in both directions may not be reached at the same time. Internal stresses due to change of moisture content may also tend to unbalance the strength ratio.

SCOPE AND METHOD OF TESTS.

The results and conclusions which follow are based on tests of about 34 species. In general, 8 thicknesses of plywood were tested, as follows: 3/30, 3/24, 3/20, 3/16, 3/12, 3/10, 3/8, and 3/6 inch.

Most of the tests were on panels composed of three plies of equal thickness of the same species, with the grain of successive plies at right angles. In addition tests were made on plywood of various numbers of plies; having various ratios between the core and the total panel

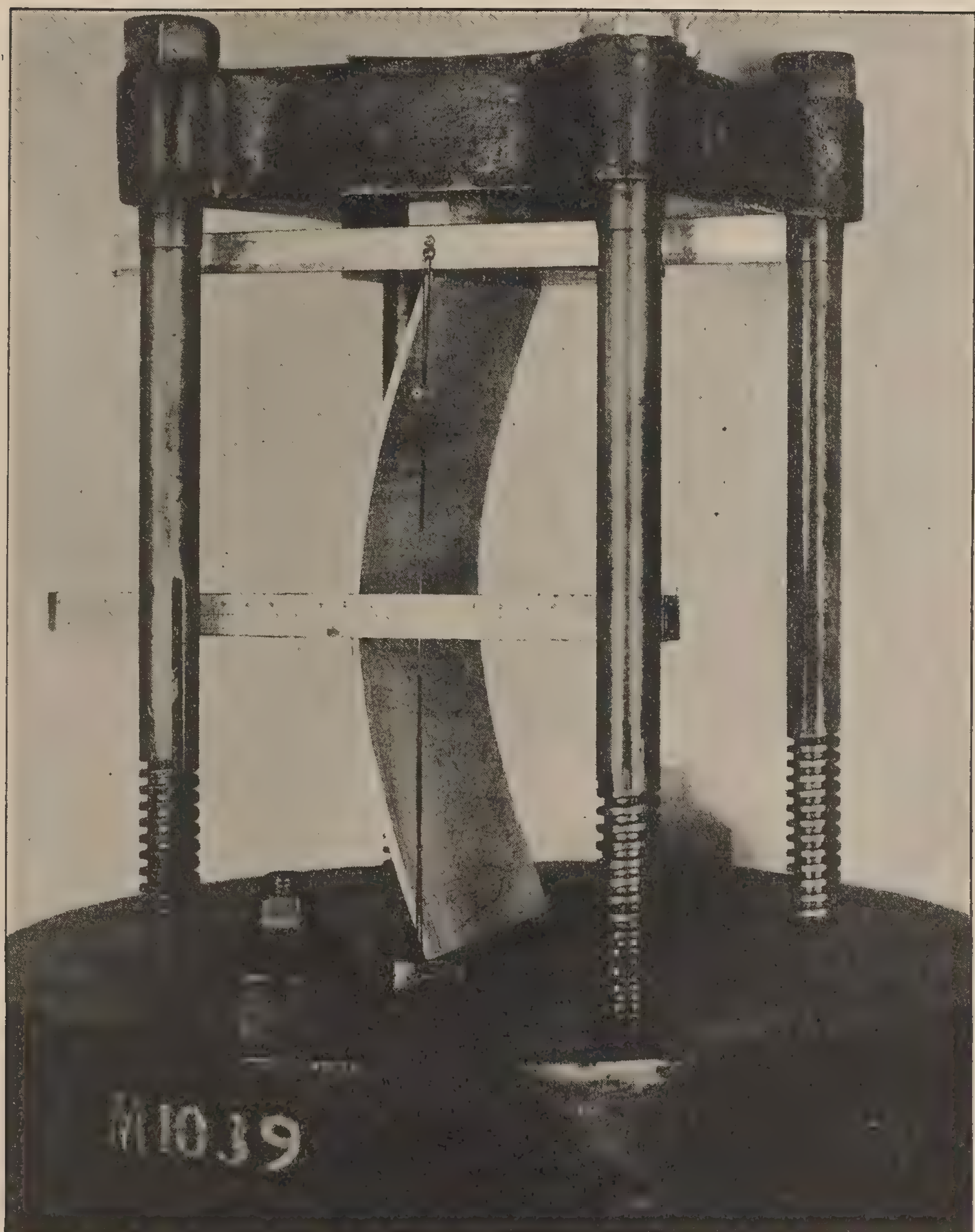

FIG. 1.—COLUMN-BENDING TEST.

FIG. 2.—SPLITTING TEST.

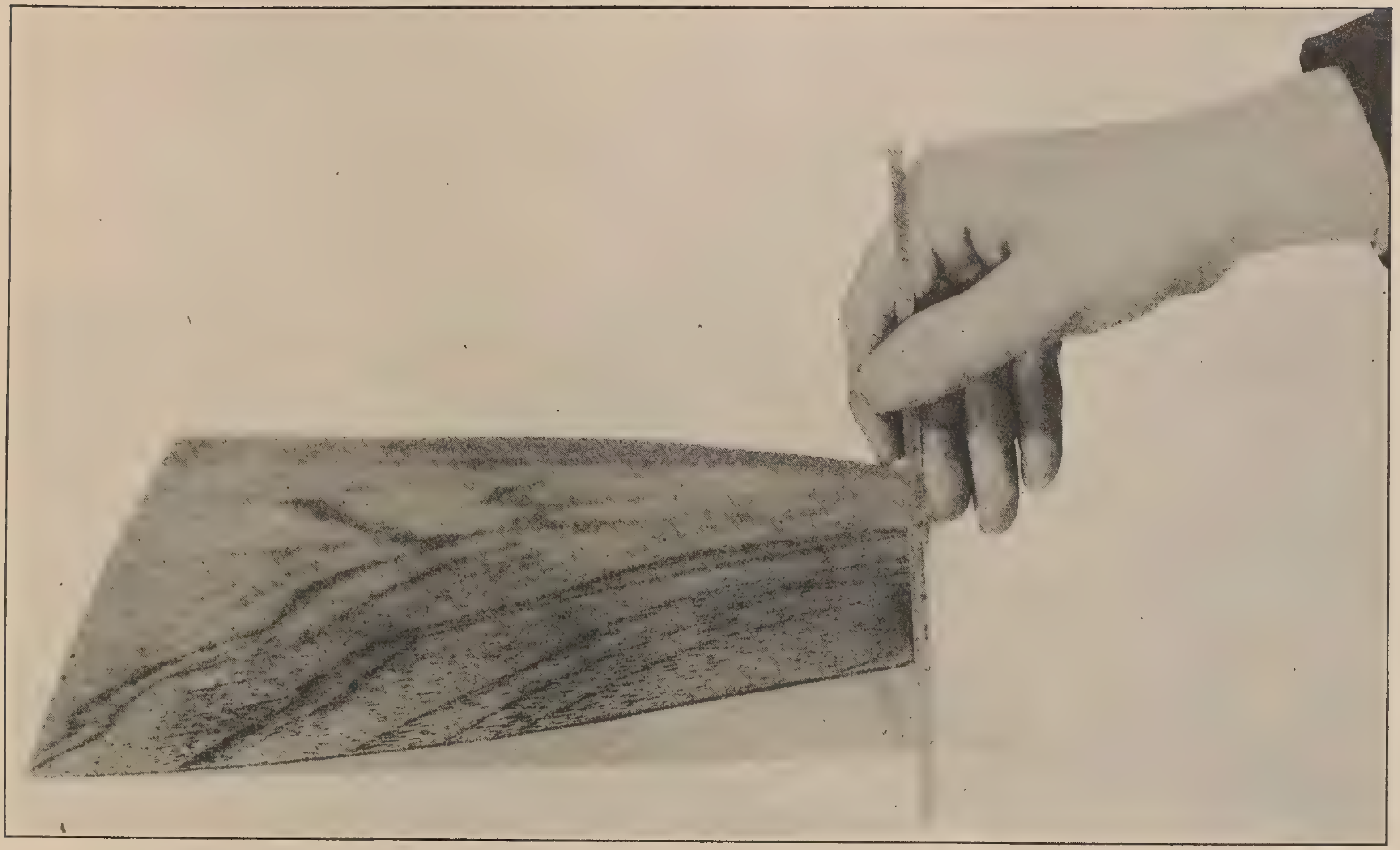

FIG. 3.—METHOD OF MEASURING CUPPING.

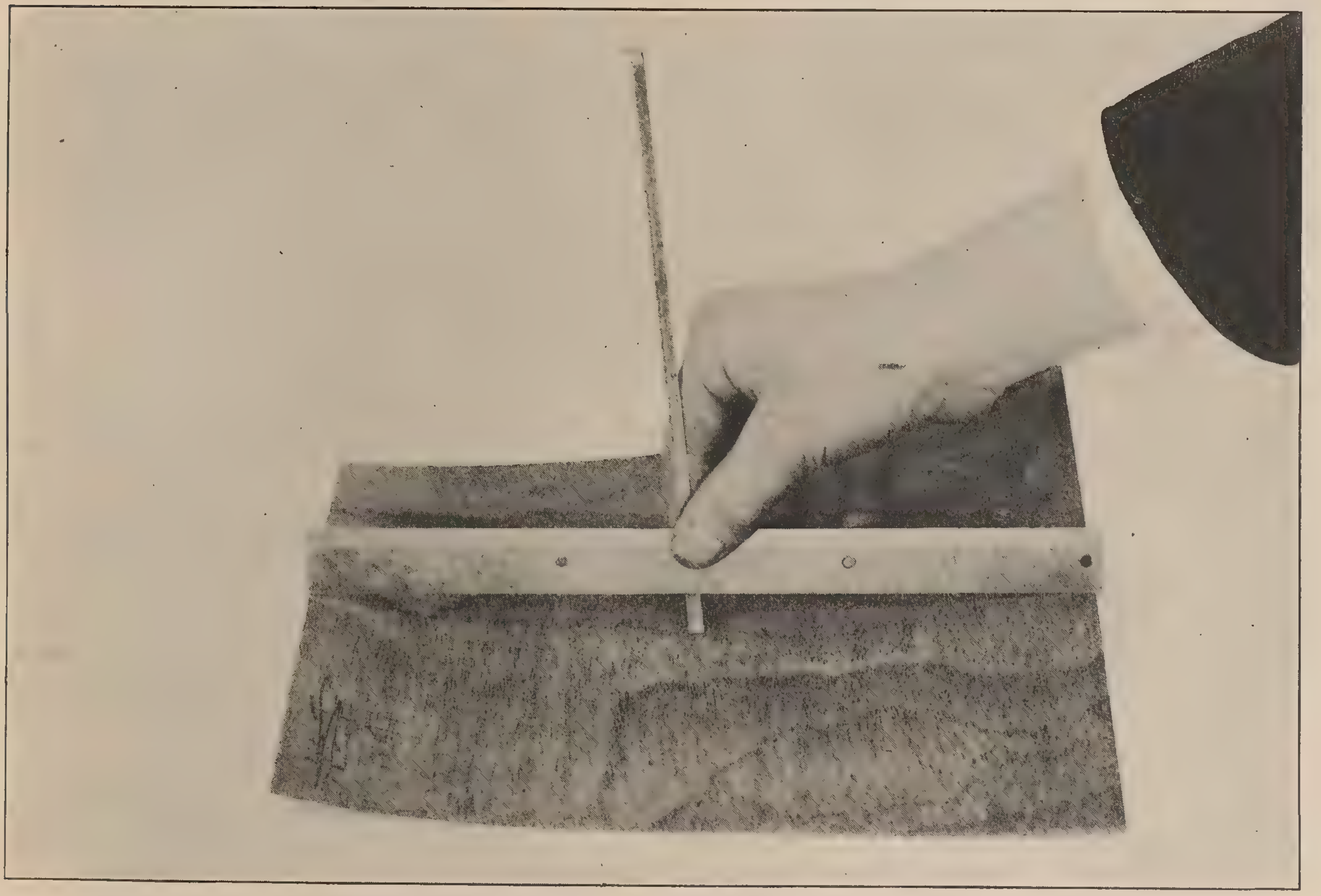

FIG. 4.—METHOD OF MEASURING TWISTING.

thickness; having the plies glued at angles other than 90° with each other; and on plywood in which the core and the faces were not of the same species.

Bending tests.—As a rule bending tests were made on specimens measuring 5 by 12 inches, although some of the thinner specimens were cut to a length of 6 inches. In half of the tests the grain of the faces was parallel to the direction of application of the load, and in half perpendicular.

Figure 1 shows the method of conducting the column-bending test. The ends of the test piece were rounded to approximately a semicircle. Deflections were measured at the center of the specimen as shown in the photograph. The product of the load and the corresponding deflection was recorded as the bending moment. For some of the thicker specimens it was not satisfactory to test in column bending on account of separation of the plies. These specimens were tested as a beam in ordinary cross bending.

The formula for computing the column-bending modulus is given at end of report. The results of the tests are included in Table 1.

In most cases in column bending the direct compressive stress at the maximum moment is only a small fraction of the bending or flexural stress, so that the column-bending modulus may be used with little error in all computations in a capacity similar to the bending strength or modulus of rupture of plain timber tested in cross bending. Like the modulus of rupture, it is not an actual stress but a measure of the strength in bending.

Tension tests.—Tests were made to determine the tensile strength of plywood both parallel and perpendicular to the grain of the faces. Specimens 3 by 12 inches in size were used, the center portion being trimmed down to approximately an inch wide. They were held by ordinary flat grips, and tested in direct tension to rupture.

Plywood tension members, while not very common, are in use and the data may be applied in computations. The tensile strength is the average stress over the section at failure.

The results of the tensile tests are included in Table 1 and 4.

Splitting test.—For splitting tests square pieces 3¼ by 3¼ inches were used. Upon the center of the test piece a conical spear (shown in fig. 2) was first dropped from a height of one-half inch. The spear was 8 inches long and 2 inches in diameter at the upper end and with the rod weighed 11.22 pounds. Carrying the test piece upon its point it was then dropped from increasing heights with an increment of one-half inch until failure due to splitting occurred. The resistance of the material to splitting is represented by its "splitting energy;" the formula for its computation is given near end of report.

The splitting energy is a measure of resistance to splitting at the screw or bolt fastenings of veneer panels. It is merely a factor for comparing different panels, and as a numerical quantity can not be used in design.

A comparison of the relative resistance to splitting of various three-ply panels will be found in Table 1.

Warping tests.—Warping may consist of cupping or twisting, or a combination of these two actions. Pieces of plywood 12 inches square were used for warping tests.

To determine cupping, a straightedge was placed over a median line drawn on the specimen perpendicular to the grain of the faces (see fig. 3), and the recession of the point deflected farthest from the straightedge was measured. This recession was recorded as the cupping of the panel.

To determine twisting, the panel was placed upon a flat surface so that three corners were resting upon the surface. The distance from the surface to the fourth corner was measured as shown in figure 4 and recorded as the "twist in 12 inches."

Information of the kind obtained in this test is of value in selecting a panel for structural parts where flat, undistorted surfaces are important. The results indicate roughly the comparative resistance to external conditions that tend to warp or distort a panel. The smaller the cupping and twisting under test, the more desirable the panel for flat work.

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Tests to determine the modulus of elasticity of plywood.—Moduli of elasticity were determined either from the column-bending test or from the cross-bending tests on plywood. Formulas used in the computations are given near end of report.

Shrinkage tests.—A limited number of shrinkage measurements were made upon 4¼-inch squares of plywood glued with water-resistant glue after they had been soaked in water for 10 days and then brought to oven-dry condition. Changes in thickness and dimensions parallel and perpendicular to the face grain were measured.

WARPING OF PLYWOOD.

Symmetrical construction.—On account of the great difference in shrinkage of wood in the direction parallel to the grain and perpendicular to it, a change in moisture content of plywood will inevitably either introduce or relieve internal stresses. Suppose, for example, the moisture content of a three-ply construction having the grain of the core at right angles to the grain of the faces is lowered. The core will tend to shrink a great deal more than the faces in the direction of the grain of the faces. This subjects the faces to compression stresses and the core to tensile stresses. If the faces are of exactly the same thickness and of like density the stresses are symmetrically distributed and no cupping should ensue.

On the other hand, suppose the grain of one face runs in the *same* direction as the core. It is obvious that the internal stresses are no longer symmetrically distributed, inasmuch as the compressive stress in one face has been removed. This face now shrinks a great deal more than the other face in the direction of the grain of the latter. The result is cupping.

The effect of drying on a three-ply unsymmetrical construction in which the grain of two adjacent plies was parallel is shown in (*b*), figure 5. The panel has curled up into a cylindrical surface with the parallel plies on the inner side. By adding another ply at right angles to the core we see that symmetry could again be established and that while we would have a four-ply panel it virtually gives a three-ply construction with a core of double the face thickness and would be regarded as such.

The necessity for exercising care in sanding the faces of a panel is obvious, inasmuch as with different thicknesses on the faces a changing of moisture content would introduce unequal forces.

In order to obtain symmetry it is also necessary that both faces or symmetrical plies be of the same species.

Summarizing briefly, a veener panel to retain its form with changes of moisture must be symmetrically constructed. Symmetry is obtained by using an odd number of plies. The plies should be so arranged that for any ply of a particular thickness there is a parallel ply of the same thickness and of the same species on the opposite side of the core and equally removed from the core.

Direction of the grain of the plies.—In careless construction the successive plies may not always be glued with the grain either exactly parallel or exactly at right angles to the core. An extreme case of this kind is shown in (*a*), figure 5, in which the plies were glued so that the grain of each face of the panel was at 45° with the grain of the core and the two faces were at 90° with respect to each other. A construction involving angles other than 0 and 90° introduces twisting. Tests have shown that deviations as small as 5° from the standard 90° construction may introduce considerable twisting. Figure 6 shows examples of faulty constructions of plywood, including several panels in which the grain of one face is not parallel to that of the other face nor at right angles to the grain of the core.

Moisture control.—Since a change in moisture content may introduce cupping and twisting if the panel is not carefully constructed, the moisture content of the veneer should be so controlled as to give as far as practicable plies of the same moisture content before gluing and finished panels should have about the same moisture content when they leave the conditioning room as they will average when in use. For service in the open air, a moisture content of frcm 10 to 15 per cent in the finished panels will usually give satisfactory results.

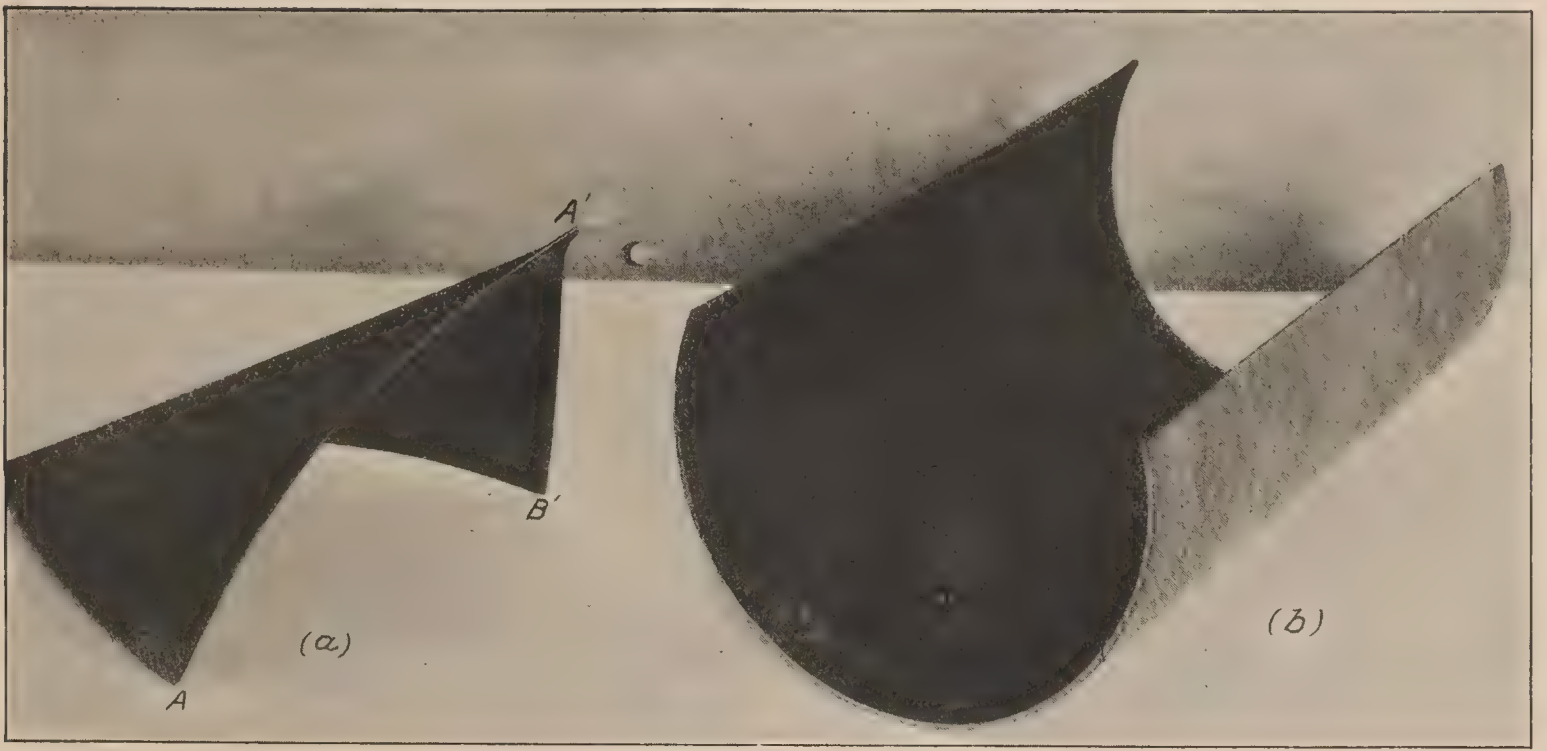

FIG. 5.—TWISTING AND CUPPING OF PLYWOOD.

(*a*) Twisting resulting from a construction with grain of faces at 45° witn grain of core. (*b*) Cupping which results from unsymmetrical construction in plywood.

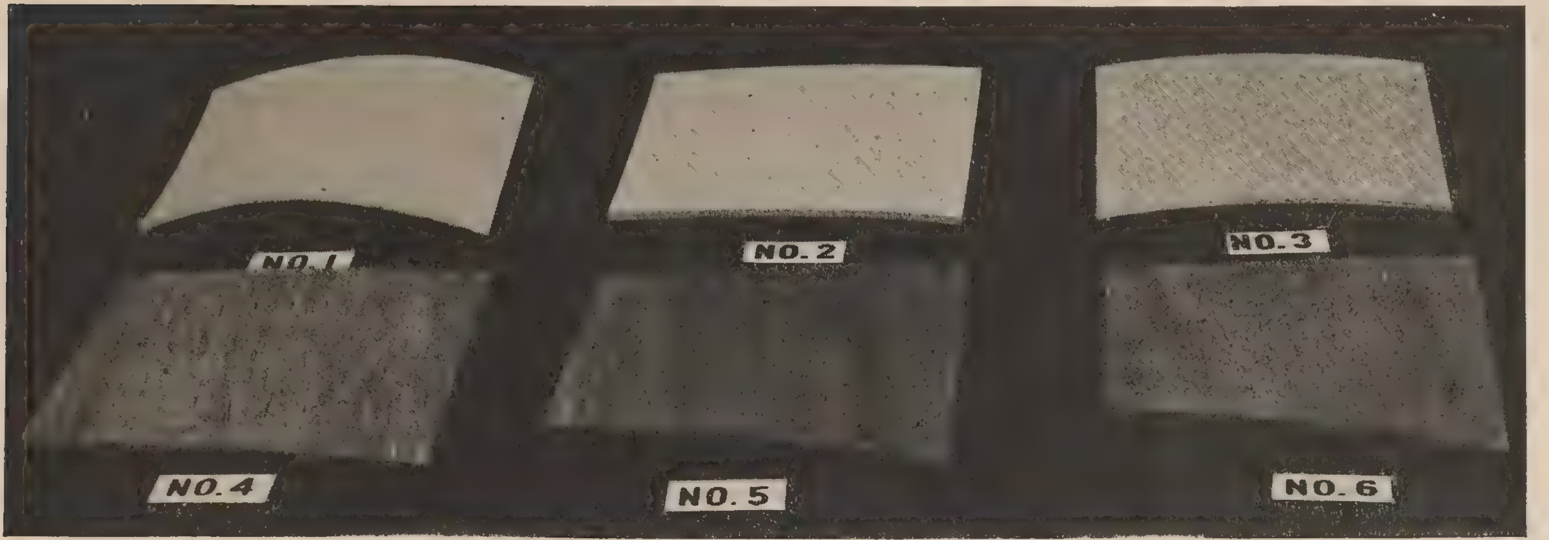

FIG. 6.—FAULTY PLYWOOD CONSTRUCTION CAUSING WARPING.

Panel No. 1.—Two-ply, $\frac{1}{16}$ maple veneer, grain of one ply at 90° to grain of other.
Panel No. 2.—Four-ply, $\frac{1}{16}$ maple veneer, grain of successive plies at 90°.
Panel No. 3.—Three-ply, $\frac{1}{16}$ maple veneer on one face and $\frac{1}{16}$ basswood core, and $\frac{1}{16}$ basswood on other face, grain of successive plies at 90°.
Panel No. 4.—Three-ply, $\frac{1}{16}$ red-gum veneer, angle between grain of faces 10°, between core and faces 85°.
Panel No. 5.—Three-ply, $\frac{1}{16}$ red-gum veneer, angle between grain of faces 20°, between core and faces 80°.
Panel No. 6.—Three-ply, $\frac{1}{16}$ red-gum veneer, angle between grain of faces 30°, between core and faces 75°.

Relation of density of veneer to warping.—Numerous tests have shown that the warping of plywood panels when subjected to varying moisture contents is least for the panels made of low density veneer, and that in general warping increases with increasing density.

Relation between warping and the ratio of the core of three-ply wood to the total plywood thickness.—A high ratio of core to total plywood thickness contributes to maintaining a flat, unwarped surface. In general, a ratio of from 0.5 to 0.7 for three-ply construction will give satisfactory results where flatness is an important consideration. Of three-ply panels having cores of the same weight the panels having cores of low density will, in general, show less warping than those having high density.

SHRINKAGE OF PLYWOOD.

The shrinkage of plywood will vary with the species, the ratio of ply thickness, the number of plies, and the combination of species. The average shrinkage obtained from several hundred tests on a variety of combinations of species and thicknesses in bringing three-ply wood from the soaked to the oven-dry condition was about 0.45 per cent parallel to the face grain and 0.67 per cent perpendicular to the face grain, with the ranges of from 0.2 to 1 per cent and 0.3 to 1.2 per cent, respectively. Individual cases of some species may give wider ranges than these. The species included in the tests were mahogany, birch, poplar, basswood, red gum, chestnut, cotton gum, elm, sugar maple, black walnut, Spanish cedar, and spruce. From this it is seen that the shrinkage of plywood is only about one-tenth as great as that across the grain of an ordinary board.

EFFECT OF INCREASING THE NUMBER OF PLIES.

The question frequently arises, Should three plies or more than three be used for a panel of a given thickness? The particular use to which the panel is to be put must answer this question. Commercial considerations are a factor also.

An increase in the number of plies results in a decrease in the tensile and bending strength parallel to the grain of the faces and an increase in the corresponding strength at right angles to the grain of the faces.

If the same bending or tensile strength is desired in two directions, parallel and perpendicular to the grain of the faces, the greater the number of plies the more nearly the desired result is obtained. It must be borne in mind, however, that a plywood with a large number of plies, while stronger at right angles to the grain of the faces, can not be so strong parallel to the grain of the faces as three-ply wood, and hence a three-ply panel is preferable where greater strength is desired in one direction than in the other.

Where great resistance to splitting is desired, as in plywood that is fastened along the edges with screws and bolts and is subject to forces through the fastenings, a large number of plies affords a better fastening.

It is common experience that a glued joint is more likely to fail when thick laminations are glued with the grain crossed than when thin laminations are glued. The same weakness exists in plywood when thick plies are glued together. When plywood is subject to moisture changes, stresses in the glued joint due to shrinkage are greater for the thick plies than for the thin plies. Hence in plywood constructed with many thin plies the glued joints will not be as likely to fail as in plywood constructed with a smaller number of thick plies.

RATIO OF CORE TO TOTAL PLYWOOD THICKNESS.

At first thought it may seem that the proper selection of the ratio of core to total plywood thickness in three-ply construction may enable the designer to get the same strength in both directions, as is possible with many ply panels. While this is partially true, it is not true that the same ratio will serve for both tension and bending. Taking birch, for example, a ratio of core to total plywood thickness of 5 to 10 gives the same strength in tension in both directions, but a ratio of about 7 to 10 gives the same strength in bending in both directions. For either ratio the plywood is not nearly so resistant to splitting as plywood of a greater number of plies totaling the same thickness.

VENEER SPECIES FOR CORES.

Where high column strength for minimum weight and a flat panel are desired, full advantage of a strong species such as birch in the faces is best obtained by using a thick core of a species such as basswood or yellow poplar rather than a thinner core of the same weight but of a species of greater density.

The greater separation of the faces gives a marked increase in the resistance to forces that tend to bend the panel. Since the maximum load a column can carry varies as the cube of the thickness, the superiority of a low-density core panel over a high-density core panel of the same weight when the load is applied parallel to the grain is obvious. A core of the same weight but only half the specific gravity of another core will be twice as thick, and the panel faces will consequently be spaced twice as far apart.

The following low-density species are satisfactory for core stock in plywood: Basswood, fir (grand, noble, and silver), redwood, Spanish cedar, white pine, spruce (red, white, or Sitka), yellow poplar, western hemlock, sugar pine, and cotton gum.

VENEER SPECIES FOR FACES.

Face plies serve different functions in various parts of an airplane. Any species in any one of the three groups shown under "Uses and properties of various species" may be used for face stock. In order to obtain the same strength as species in the first group it is necessary to use thicker veneer for other species.

Thickness factor K_s.—By multiplying the thickness of a piece of birch veneer by the constant K_s (Table 2) for any particular species a veneer of approximately the same bending strength as birch is obtained. If it is desired to substitute one species for another, therefore, this strength ratio should be considered. The values of K_s are obtained from data on the strength of three-ply wood in which each ply is of the same thickness and species, and its application to cases widely different from this will involve some error. K_s is derived as follows:

The strength in bending is measured by the bending moment a piece of plywood can sustain. If we denote the maximum bending moment of a strip of three-ply wood 1 inch wide and of thickness d_1, by M_1 and the stress at failure by S_1 (column-bending modulus), then

$$M_1=\frac{S_1\,d_1^2}{6}.$$

Similarly the maximum moment of another strip of a different species will be denoted by M_2, its stress at failure S_2, and thickness d_2. By a proper selection of thickness d_2 the second strip may be made to withstand the same maximum bending moment, so that $M_2=M_1$ or $S_2d_2^2=S_1d_1^2$.

Then the desired thickness

$$d_2=d_1\sqrt{\frac{S_1}{S_2}}$$

Taking d_1 as the unit thickness of a birch plywood strip and expressing the maximum stresses in percentage of birch, we have

$$d_2=\sqrt{\frac{100}{S_2}}$$

or, in general,

$$K_s=\sqrt{\frac{100}{S}},$$

where K_s is the thickness of the plywood whose column-bending modulus corresponds to S, and whose total bending strength, given by the bending moment, is the same as that of birch plywood of thickness unity.

It is necessary to use considerable care in the application of this factor to plywood of mixed species, as the constants do not apply under such conditions.

Thickness factor K_w.—This factor serves to obtain the thickness of a ply of any species equal in weight to a ply of yellow birch of unit thickness. It is obtained by dividing the density[1] of birch by the density of the species for which the thickness is desired. For yellow poplar, for example, the thickness of a ply equal in weight to a 1/16-inch ply of birch is $1.54 \times 1/16 = 0.096$ inches.

Uses and properties of various species.—On the basis of their mechanical properties, the veneer species commonly used for the face stock of plywood for airplanes may be grouped as follows:

Group 1.—Beech, birch (sweet or yellow), hard maple, black walnut.

Group 2.—White elm,[2] red gum, soft maple, mahogany (African or true), sycamore.

Group 3.—Basswood, Spanish cedar, fir (grand, noble, or silver), cotton gum, western hemlock, sugar pine, white pine, yellow poplar, redwood, spruce (red, white, or Sitka).

Where a flat panel, high bending strength, or high column strength with minimum weight are desired, species of group 3 should be used as face stock. Some of these species, such as spruce, can not be finished properly without a considerable amount of sanding, and all but light sanding is undesirable because it may unbalance the construction. In fuselage bulkheads or other hidden parts where finish is secondary, any of the species of group 3 should be satisfactory for face veneer as well as for core stock.

Hardness, resistance to abrasion, and strength of fastening increase considerably with increasing density of wood, so that where any one or all of these factors are of importance the heavier woods beginning with group 1 should be used.

Where finish is desired species of group 1 or group 2 should be used.

Where the plywood must be steamed, or soaked and bent into a form in which it is to remain, species of group 1 or group 2 should be used.

Where failure of an airplane part is likely to occur from buckling, as in plywood fuselages in which the shell carries considerable stress, it is recommended that the plywood be made entirely of low-density species, such as those in group 3. Numerous tests on plywood columns have shown that three-ply columns of low-density species, such as are included in group 3, carry from 2 to 2.5 times the load of three-ply columns of the same weight of species included in group 1. Buckling is a form of column failure, and for that reason greater resistance to buckling per unit weight would be expected from the use of low-density veneer.

SIZE, WEIGHT, AND THICKNESS OF COMMERCIAL VENEER.

The average length of sawed veneer sheets is about 14 feet, and the maximum 24 feet; the average length of sliced veneer is about 10 feet, and the maximum 18 feet; rotary-cut veneer averages about 6 feet, with a maximum of 16 feet. Sawed veneer is seldom cut less than 1/28 nch thick. Sliced veneer of some species may be cut as thin as 1/100 inch, but is seldom cut thicker than 1/16 inch. Rotary-cut veneer of some species may be cut from 1/100 inch to almost 1/2 inch in thickness. Sawed and sliced veneer sheets are limited in width by the diameter of the log, whereas rotary cut veneer may be any width consistent with easy handling.

Except for the 1/100 inch veneer, all the thicknesses listed in Table 3 are commercial. Table 3 may be used in computing the weight of veneer sheets of any size and thickness, and of plywood made of any combination of the species listed. A sample computation is given near end of report.

JOINTS IN PLYWOOD PANELS.

There are three types of joints commonly used for joining plywood panels: (*a*) Riveted joints, (*b*) glued joints in individual plies, and (*c*) glued joints extending through the entire thickness of plywood. These will be considered in detail.

Riveted joints.—The most satisfactory joints of the riveted type are made with tubular rivets. Tension tests have shown quite conclusively that it is very difficult to obtain more than

[1] The density data for the domestic species used in computing K_w are those given in United States Department of Agriculture Bulletin 556, "Mechanical Properties of Woods Grown in the United States," and do not include the weight of the glue.

[2] White elm should not be used where a high finish is desired. However, it has exceptional bending qualities.

50 per cent efficiency with a single row of rivets. Efficiencies somewhat higher than 50 per cent may probably be obtained if two or more rows of rivets are used. In such cases the rivets should be staggered. The size of the rivet seems to have little effect upon the strength of the joint, providing the proper spacing is used. The distance between centers of rivets should be about equal to twice the outside diameter of the rivets. It is obvious that for such spacing very many rivets are required, and that the labor in making the joint is very great.

Joints in individual plies.—Two pieces of plywood may be fastened together by means of glued joints in individual plies. Joints in individual plies take a variety of forms. (See fig. 7.) Strength, ease of manufacture, and efficiency considered, the simple scarf joint appears to be the most desirable of the group. The simple butt joint should not be used where strength is important. The edge joint is satisfactory if carefully made. The slope of the scarf in the simple scarf joint should be within the range of from 1 in 20 to 1 in 30.

The use of joints in individual plies has an advantage over the other types, in that the joints in the plies may be staggered, so that a single defective joint only partially weakens the entire panel. The preparation of a joint of this type requires less time and labor than a riveted joint, but more than a scarf joint extending through the entire thickness of the panel.

Joints extending through the entire thickness of plywood.—Two types of scarf joints extending through the entire plywood thickness are shown in figure 8. These are known as the straight scarf joint and the Albatross scarf joint. It will be seen that in the Albatross joint the face ply of the one panel does not meet the face ply of the second panel, or only partially meets it.

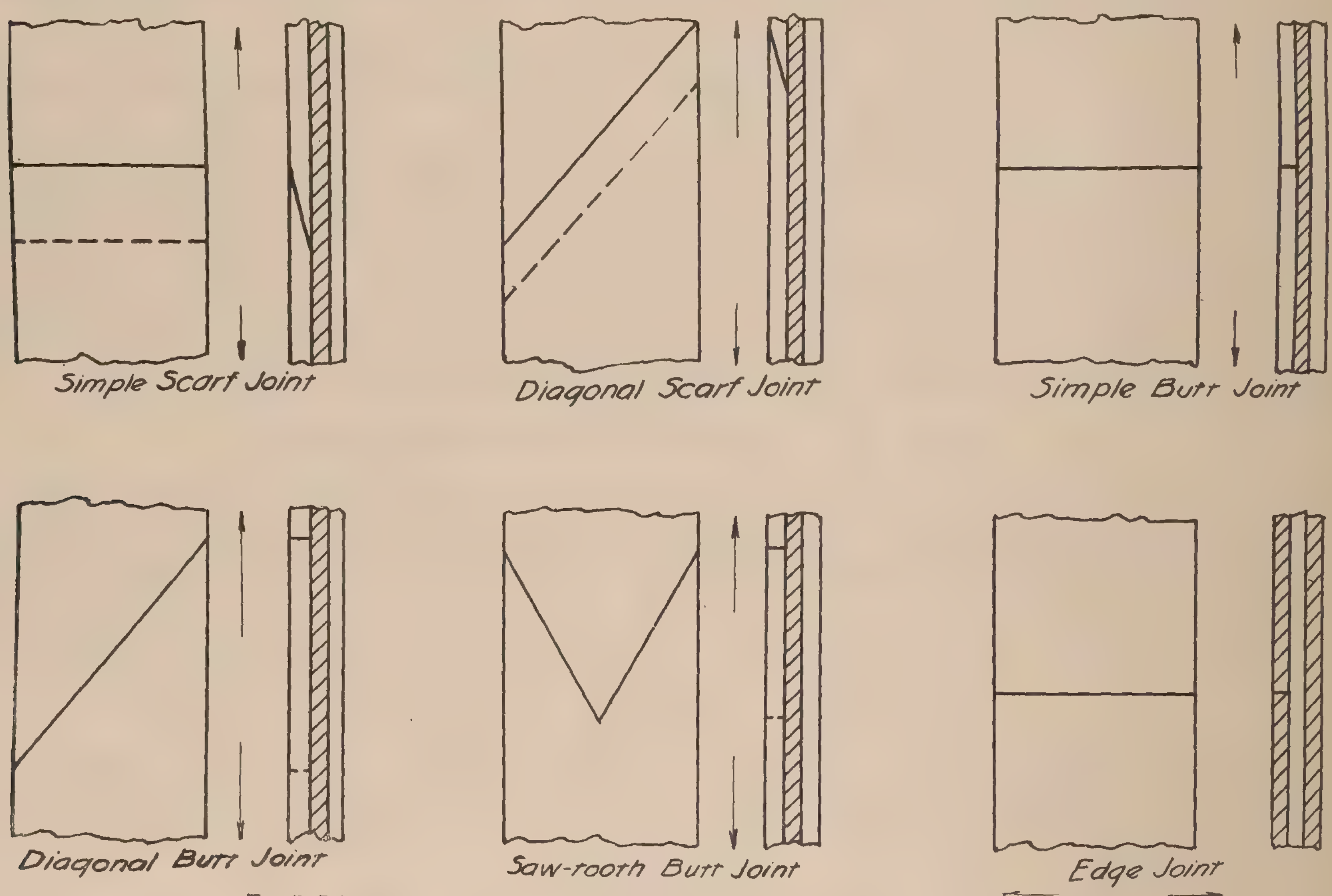

FIG. 7. Joints in the face veneer of 3-ply panels. Arrows indicate direction of face grain.

In place of being glued to wood that has the grain running in the same direction, the face ply of one panel is glued to the core of the other panel, the grain of the core being at right angles to the grain in the face. Joints in which the grain of the two pieces joined is at right angles are not so strong as joints in which the direction of grain in the two pieces is the same.

Tests indicate quite conclusively that in tension the straight scarf joint is superior. An efficiency of over 90 per cent may be obtained with this type of joint for a slope of scarf as low as 1 in 10. On account of the variations in the effectiveness of the gluing by different manufacturers it is recommended that a slope of scarf greater than this be used. A slope between 1 in 20 and 1 in 30 is recommended.

Severe weakening of scarf joints is often caused by sanding the face plies at the joint. Observations on joints thus sanded showed that in some cases more than half of the face ply was ground away. Inasmuch as the strength of a three-ply panel when bent parallel to the direction of the grain of the faces lies almost entirely in the face plies, it is obvious that a reduction in the thickness of the face plies will materially affect the strength of a panel. Consequently, it is recommended that the scarf joint be lightly sanded by hand if at all, so as not to decrease the thickness of the face veneer.

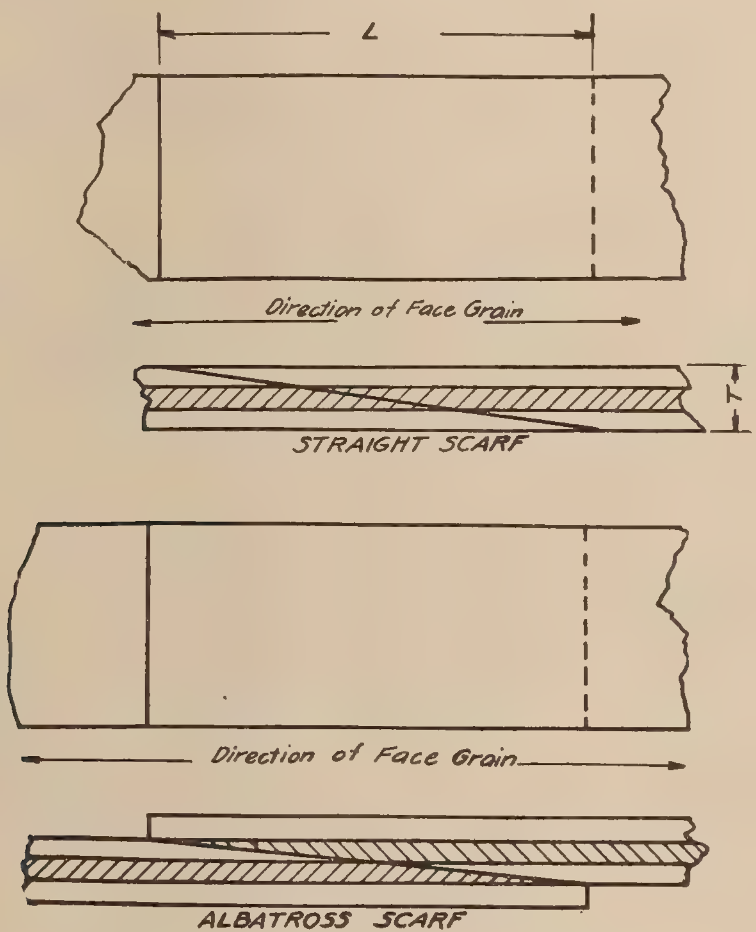

FIG. 8. Joints extending through the panel. Slope of scarf$=\frac{L}{T}$.

For scarfing plywood a jointer, sanding machine, or hand plane is ordinarily used.

Figure 9 shows the method used at the Forest Products Laboratory for pressing glued joints in plywood. The board above the panel should be relatively massive and flat, so as to distribute the pressure from the screws. Two or three layers of blotting paper furnish sufficient padding to accommodate irregularities in the surface.

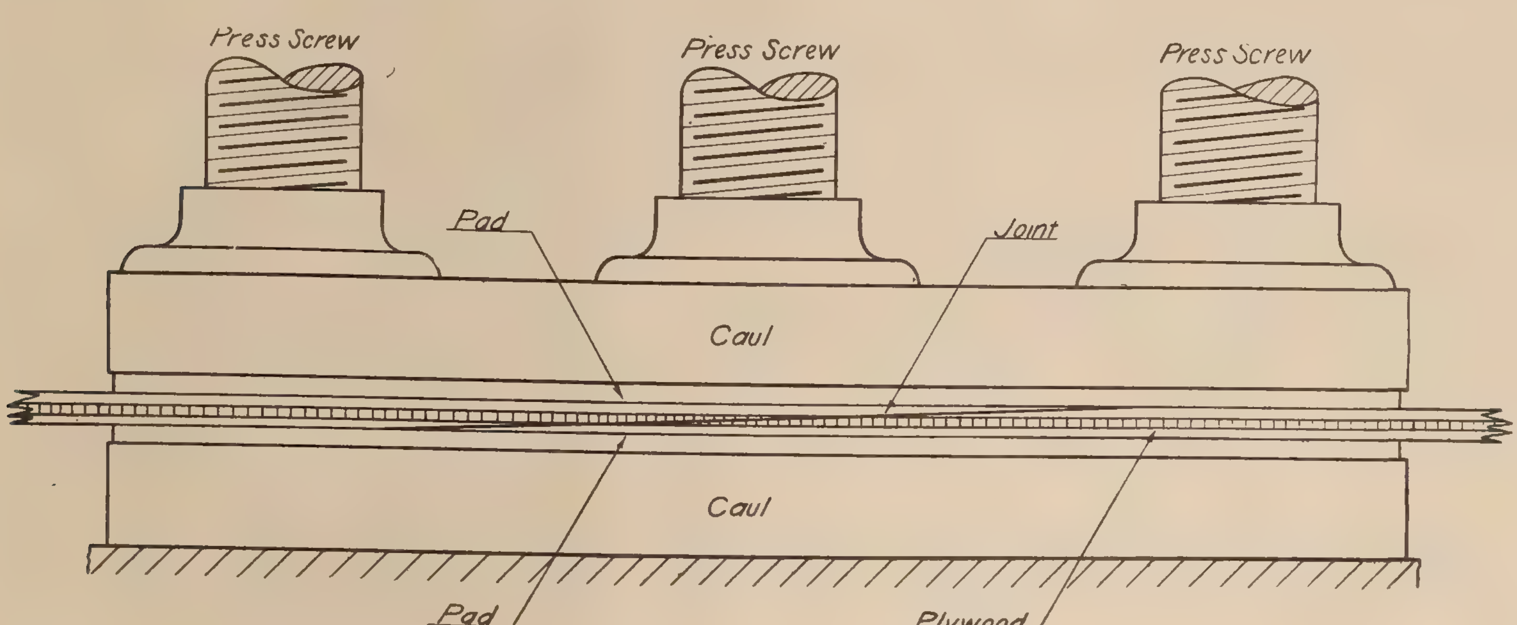

FIG. 9. Method of pressing glued joint.

FORMULAS USED IN COMPUTATION OF STRENGTH DATA.

Column-bending modulus.—The term "column-bending modulus" applies to the stress obtained by adding the direct compression stress at the maximum moment to the flexural stress at the maximum moment. The following formula applies:

$$S=\frac{P}{A}+\frac{6\,M}{bd^2}\text{ where}$$

S = Column-bending modulus.
A = Area of cross section of test piece.
P = Load at maximum moment.
M = Maximum bending moment.
b = Width of test piece.
d = Thickness of test piece.

Modulus of rupture.—The modulus of rupture is the computed stress in the outermost fibers of the plywood, when tested in crossbending as a simple beam. The following formula applies for center loading:

$$MR=\frac{1.5P\times L}{bd^2}\text{ where}$$

MR = Modulus of rupture.
P = Maximum load at center.
L = Span of test piece.
b = Width of test piece.
d = Thickness of test piece.

Tensile strength.—The term "tensile strength" applies to the stress obtained by dividing the load at rupture by the minimum cross-section area of the specimen.

$$S=\frac{P}{A}\text{ where}$$

S = Tensile strength.
P = Maximum load in tension.
A = Total cross-sectional area at minimum section.

Modulus of elasticity.—The moduli of elasticity of all column-bending and cross-bending specimens were computed by substitution in the following formulas:

For column bending,

$$E=\frac{PL^2}{\pi^2 I}$$

For cross bending (center loading),

$$E=\frac{P^1L^3}{48\,fI}\text{ where}$$

E = Modulus of elasticity of the plywood.
P = Maximum load sustained in column bending.
P^1 = Any load within the elastic limit of the plywood.
L = Length of the plywood column or the length of the span in cross bending.
I = Moment of inertia of the cross section of the specimen.
f = Deflection corresponding to P^1.

Splitting energy.—The total work done or the splitting energy W was computed by adding together the distances through which the spear fell, h_1, h_2, etc., and multiplying by the weight (11.2 pounds) of spear and rod.

$$M=11.2\ (h_1+h_2+h_3+\cdots.)$$

Explanation of Table 1.

The data of this table may be used to compute the thickness of three-ply wood members of various species when the forces acting on these members are known. The strength in bending is given by the column-bending modulus, which may be used in computations in a capacity similar to the modulus of rupture of ordinary wood. The direction in which the external forces act on the member relative to the direction of the face grain of the plywood must be taken into consideration in using the data. The strength values correspond to the moisture contents listed.

Table 1.—*Strength of various species of three-ply panels.*

All plies in any one panel were of the same thickness and of the same species—grain of successive plies at right angles. In most cases eight thicknesses of plywood, ranging from 3/30 inch to 3/6 inch were tested.

Species.	Average specific gravity of plywood based on oven-dry weight and volume at test.	Average moisture (per cent).	Column bending.						Tensile strength.				Splitting resistance.	
			Column-bending modulus.				Modulus of elasticity.							
			Parallel.[1]		Perpendicular.[1]		Parallel.[1]	Perpendicular.[1]	Parallel.[1]		Perpendicular.[1]			
			No. of tests.	Lbs. per sq. in.	No. of tests.	Lbs. per sq. in.	1,000 lbs. per sq. in.	1,000 lbs. per sq. in.	No. of tests.	Lbs. per sq. in.	No. of tests.	Lbs. per sq. in.	No. of tests.	Per cent of birch.[2]
Ash, black	0.49	9.1	120	7,760	120	1,770	1,070	96	120	6,180	120	3,940	240	3
Ash, commercial white	.60	10.2	200	9,930	200	2,620	1,420	143	200	6,510	200	4,350	400	71
Basswood	.42	9.2	200	7,120	200	1,670	1,210	85	200	6,880	200	4,300	400	63
Beech	.67	8.6	120	15,390	120	2,950	2,150	167	120	13,000	120	7,290	240	94
Birch, yellow	.67	8.5	195	16,000	200	3,200	2,260	197	200	13,210	200	7,700	400	100
Cedar, Spanish	.41	13.3	115	6,460	115	1,480	1,030	84	115	5,200	115	3,340	230	60
Cherry[3]	.56	9.1	115	12,260	115	2,620	1,630	152	115	8,460	115	5,920	230	80
Chestnut	.43	11.7	40	5,160	40	1,110	740	75	40	4,430	40	2,600	80	74
Cottonwood[4]	.46	8.8	120	8,460	120	1,870	1,440	109	120	7,280	120	4,240	240	85
Cypress, bald	.45	8.0	113	8,890	113	1,850	1,220	95	113	6,160	113	3,980	148	49
Douglas fir[5]	.48	8.6	176	9,340	200	1,940	1,530	126	200	6,188	200	3,910	374	63
Elm, cork	.62	9.4	65	12,710	65	2,500	1,980	136	65	8,440	65	5,500	130	99
Elm, white	.52	8.9	160	8,680	160	1,970	1,220	109	160	5,860	160	3,990	320	75
Fir, true[6]	.40	8.5	24	9,200	24	1,811	1,580	100	24	5,670	24	3,770	48	60
Gum[7]	.54	10.6	40	8,090	40	1,920	1,280	113	35	6,960	35	4,320	70	55
Gum, cotton	.50	10.3	80	7,760	80	1,580	1,300	111	80	6,260	80	3,760	160	60
Gum, red	.54	8.7	182	9,970	182	2,070	1,590	120	182	7,850	182	4,930	364	80
Hackberry	.54	10.2	80	8,100	80	1,880	1,150	99	80	6,920	80	4,020	160	84
Hemlock, western	.47	9.7	119	9,250	119	1,960	1,580	112	119	6,800	119	4,580	238	63
Magnolia[8]	.58	8.8	80	10,830	80	2,600	1,700	138	80	9,220	80	5,730	120	85
Mahogany, African[9]	.52	12.7	20	8,070	20	2,000	1.260	144	20	5,370	20	3,770	......	......
Mahogany, Philippine[10]	.53	10.7	25	10,160	25	2,310	1,820	169	25	10,670	25	5,990	50	90
Mahogany, true	.48	11.4	35	8,500	35	1,940	1,250	117	35	6,390	35	3,780	......	......
Maple, soft[11]	.57	8.9	120	11,540	120	2,420	1,750	145	120	8,180	120	5,380	240	106
Maple, hard[12]	.68	8.0	202	15,600	202	3,340	2,110	189	192	10,190	202	6,530	404	114
Oak, commercial red	.59	9.3	115	8,500	115	2,070	1,290	120	115	5,480	115	3,610	230	70
Oak, commercial white	.64	9.5	195	10,490	195	2,310	1,340	118	195	6,730	195	4,200	390	85
Pine, sugar	.42	9.4	65	8,050	70	1,670	1,310	90	70	5,430	70	3,690	140	47
Pine, white	.42	5.4	40	10,130	40	2,050	1,570	111	40	5,720	40	3,340	80	31
Poplar, yellow	.50	9.4	165	8,860	165	1,920	1,540	115	155	7,390	165	4,720	330	51
Redwood	.42	9.7	105	8,230	105	1,550	1,180	108	105	4,770	105	2,960	210	48
Spruce, Sitka	.42	8.3	121	7,710	121	1,690	1,370	105	121	5,650	121	3,410	224	78
Sycamore	.56	9.2	163	11,040	163	2,340	1,630	130	163	8,030	163	5,220	326	77
Walnut, black	.59	9.1	110	12,660	110	2,770	1,740	141	110	8,250	110	5,260	220	77
Yucca species	.49	7.3	33	2,960	33	900	560	44	33	2,210	33	1,700	66	14

[1] Parallel and perpendicular refer to the direction of the grain of the faces relative to the direction of the application of the force.
[2] The relative splitting resistance of the various panels tested depends largely on the holding strength of glue.
[3] Probably black cherry.
[4] Probably (common) cottonwood.
[5] Coast type.
[6] Probably white fir.
[7] Probably black gum.
[8] Probably (evergreen) magnolia.
[9] Probably Khaya sp.
[10] Probably tanguile.
[11] Probably silver maple.
[12] Sugar or black maple.

Note.—In some of the species listed above the tests are rather limited in number. Since there is considerable variation in the strength of wood, further tests on additional material would be expected to modify the values appreciably in some cases.

EXPLANATION OF TABLE 2.

When substituting one species for another in airplane plywood it is desirable to know the thickness of veneer which will give either the same bending strength or the same weight as the original material. The thickness factors K_s and K_w given in Table 2 will be found useful for this purpose. For instance, the thickness of basswood veneer required to afford approximately the same bending strength as one-tenth inch yellow poplar, may be obtained by multiplying the thickness of the yellow poplar by the ratio of the thickness factor (K_s) of basswood to that of yellow poplar. The factor K_w may be used in a similar computation to obtain the thickness of one species required to equal the weight of another.

TABLE 2.—*Thickness factors for veneer.*

Giving: (1) Veneer thickness for the same total bending strength as birch (K_s); (2) Veneer thickness for the same weight as birch (K_w).

Species.	D Average specific gravity of species [11] based on oven-dry weight and air-dry volume.	Specific gravity of glued plywood as tested based on oven-dry weight and volume at test.	Moisture content of plywood as tested.	S Unit bending strength compared with birch.[1]	K_s Thickness factor for the same total bending strength as birch. $\sqrt{\frac{100}{S}}$	K_w Thickness factor for the same weight as birch. $\frac{.63}{D}$
			Per cent.	*Per cent.*		
Ash, black	0.50	0.49	9.1	52	1.39	1.26
Ash, commercial white	.58	.60	10.2	72	1.18	1.09
Basswood	.38	.42	9.2	48	1.44	1.66
Beech	.63	.67	8.6	94	1.03	1.00
Birch, yellow	.63	.67	8.5	100	1.00	1.00
Cedar, Spanish	.34	.41	13.3	43	1.52	1.85
Cherry [2]	.51	.56	9.1	80	1.12	1.24
Chestnut	.44	.43	11.7	34	1.72	1.43
Cottonwood	.43	.46	8.8	56	1.34	1.47
Cypress, bald	.44	.45	8.0	57	1.32	1.43
Douglas fir [3]	.51	.48	8.6	60	1.29	1.24
Elm, cork	.66	.62	9.4	78	1.13	.95
Elm, white	.51	.52	8.9	58	1.31	1.24
Fir, true [4]	.38	.40	8.5	57	1.32	1.66
Gum [5]	.52	.54	10.6	55	1.35	1.21
Gum, cotton	.52	.50	10.3	49	1.43	1.21
Gum, red	.49	.54	8.7	64	1.25	1.29
Hackberry	.54	.54	10.2	55	1.35	1.17
Hemlock, western	.42	.47	9.7	60	1.29	1.50
Magnolia [6]	.51	.58	8.8	74	1.16	1.24
Mahogany, African [7]	.46	.52	12.7	56	1.34	1.37
Mahogany, Philippine [8]	.57	.53	10.7	68	1.21	1.10
Mahogany, true	.49	.48	11.4	57	1.32	1.29
Maple, soft [9]	.48	.57	8.9	74	1.16	1.31
Maple, hard [10]	.62	.68	8.0	100	1.00	1.02
Oak, commercial red	.63	.59	9.3	59	1.30	1.00
Oak, commercial white	.69	.64	9.5	69	1.20	.91
Pine, sugar	.37	.42	9.4	51	1.40	1.70
Pine, white	.39	.42	5.4	64	1.25	1.61
Poplar, yellow	.41	.50	9.4	58	1.31	1.54
Redwood	[12] .36	.42	9.7	50	1.41	1.75
Sycamore	.50	.56	9.2	71	1.19	1.26
Spruce, Sitka	.38	.42	8.3	50	1.41	1.66
Walnut, black	.57	.59	9.1	83	1.10	1.10
Yucca species	....	.49	7.3	23	2.09	....

[1] Average of the column bending moduli parallel and perpendicular to grain compared to birch, based on tests of 3-ply wood, each ply one-third of the total panel thickness.
[2] Probably black cherry.
[3] Coast type.
[4] Probably white fir.
[5] Probably black gum.
[6] Probably (evergreen) magnolia.
[7] Probably Khaya species.
[8] Probably tanguile.
[9] Probably silver maple.
[10] Probably sugar or black maple.
[11] Values of domestic species taken from U. S. Department of Agriculture Bulletin 556, Mechanical Properties of Woods Grown in the United States.
[12] Based on tests not included in Bulletin 556.

Explanation of Table 3.

This table gives the approximate weight of individual sheets of veneer in ounces per square foot, making possible the computation of the weight of plywood built up of any combination of thicknesses and veneer species listed and of any number of plies. The approximate weights of two common water-resistant plywood glues in ounces per square foot of glued surface are also given.

It should be remembered that the weight of wood is quite variable, and that large differences from the figures are to be expected, particularly with small quantities of material.

Example: To get the weight of a square foot of 5-ply wood consisting of 1-ply of 1/12-inch basswood, 2 plies of 1/16-inch basswood, and 2 plies of 1/20-inch yellow birch for faces, at 12 per cent moisture, glued with casein glue.

$$\text{Weight} = [(1 \times 2.64) + (2 \times 1.98) + (2 \times 2.62)]\ 1.12 + (4 \times 0.4) \overline{=} 14.9 \text{ ounces.}$$

The example above is slightly in error through neglecting the change in volume between the moisture content at 12 per cent and the moisture listed in the table.

Table 3.—*Oven dry weights of veneer of veneer of various species and thicknesses.*

[In ounces per square foot of 1-ply; veneer thickness in inches.]

Species.	Specific gravity based on oven-dry weight and air-dry volume.	Air-dry moisture content (per cent).	1/100	1/80	1/64	1/60	1/55	1/48	1/40	1/32	1/28	1/24	1/20	1/16	1/12	1/10	1/8	1/6	3/16	1/4
Ash, black	0.50	10.4	0.42	0.52	0.65	0.69	0.76	0.87	1.04	1.30	1.49	1.74	2.08	2.60	3.47	4.16	5.20	6.94	7.81	10.41
Ash, commercial white	.58	8.9	.48	.60	.75	.80	.88	1.00	1.21	1.51	1.72	2.01	2.41	3.02	4.02	4.82	6.04	8.05	9.05	12.06
Basswood	.38	8.4	.32	.40	.49	.53	.58	.66	.79	.99	1.13	1.32	1.58	1.98	2.64	3.16	3.96	5.28	5.94	7.92
Beech	.63	11.2	.52	.66	.82	.87	.95	1.09	1.31	1.64	1.87	2.19	2.62	3.28	4.37	5.24	6.56	8.74	9.84	13.12
Birch, yellow	.63	9.6	.52	.66	.82	.87	.95	1.09	1.31	1.64	1.87	2.19	2.62	3.28	4.37	5.24	6.56	8.74	9.84	13.12
Butternut	.39	7.6	.32	.41	.51	.54	.59	.68	.81	1.02	1.16	1.35	1.62	2.03	2.71	3.25	4.06	5.42	6.09	8.12
Cedar, Spanish	.37	7.3	.31	.38	.48	.51	.56	.64	.77	.96	1.10	1.28	1.54	1.92	2.56	3.08	3.85	5.13	5.77	7.70
Cherry, black	.51	9.2	.42	.53	.66	.71	.77	.88	1.06	1.33	1.52	1.77	2.12	2.65	3.54	4.25	5.31	7.08	7.97	10.62
Chestnut	.44	8.6	.37	.46	.57	.61	.67	.76	.92	1.14	1.31	1.52	1.83	2.29	3.05	3.67	4.58	6.10	6.87	9.16
Cottonwood (common)	.43	4.7	.36	.45	.56	.60	.65	.75	.90	1.12	1.28	1.49	1.79	2.24	2.98	3.58	4.47	5.97	6.71	8.96
Cypress bald	.44	9.0	.37	.46	.57	.61	.67	.76	.92	1.14	1.31	1.52	1.83	2.29	3.05	3.67	4.58	6.10	6.86	9.16
Douglas fir (Washington and Oregon)	.51	6.2	.42	.53	.66	.71	.77	.88	1.06	1.33	1.51	1.77	2.12	2.65	3.53	4.24	5.30	7.08	7.96	10.6
Douglas fir (Montana and Wyoming)	.44	9.4	.37	.46	.57	.61	.67	.76	.92	1.15	1.31	1.53	1.83	2.29	3.05	3.66	4.58	6.10	6.87	9.16
Elm, white	.51	8.8	.42	.53	.66	.71	.77	.88	1.06	1.33	1.52	1.77	2.12	2.65	3.54	4.25	5.31	7.08	7.97	10.62
Gum, black	.52	7.2	.43	.54	.68	.72	.79	.90	1.08	1.35	1.55	1.80	2.17	2.71	3.61	4.33	5.42	7.32	8.12	10.82
Gum, cotton	.52	6.1	.43	.54	.68	.72	.79	.90	1.08	1.35	1.55	1.80	2.17	2.71	3.61	4.33	5.42	7.32	8.12	10.82
Gum, red	.49	11.3	.41	.51	.64	.68	.74	.85	1.02	1.28	1.46	1.70	2.04	2.55	3.40	4.08	5.10	6.80	7.66	10.20
Hackberry	.54	9.2	.45	.56	.70	.75	.82	.94	1.12	1.40	1.61	1.87	2.25	2.81	3.75	4.49	5.63	7.50	8.44	11.24
Hemlock, western	.42	8.6	.35	.44	.55	.58	.64	.73	.87	1.09	1.25	1.46	1.75	2.18	2.91	3.50	4.37	5.83	6.56	8.74
Magnolia (evergreen)	.51	8.8	.42	.53	.66	.71	.77	.88	1.06	1.33	1.51	1.77	2.12	2.65	3.53	4.24	5.30	7.08	7.96	10.6
Mahogany, Central American	.49	7.9	.41	.51	.65	.68	.75	.85	1.02	1.28	1.46	1.70	2.04	2.55	3.50	4.08	5.10	6.80	7.66	10.20
Mahogany, African	.46	8.0	.38	.48	.60	.64	.70	.80	.96	1.19	1.37	1.59	1.91	2.39	3,19	3.83	4.78	6.38	7.17	9.57
Maple, silver	.48	8.2	.40	.50	.62	.67	.73	.83	1.00	1.25	1.43	1.67	2.00	2.50	3.33	4.00	5.00	6.66	7.50	7.00
Maple, sugar	.62	10.5	.52	.65	.81	.86	.94	1.08	1.29	1.61	1.85	2.15	2.58	3.23	4.30	5.16	6.46	8.60	9.69	12.91
Oak, commerical red	.64	10.7	.53	.67	.83	.89	.97	1.11	1.33	1.66	1.90	2.22	2.66	3.33	4.44	5.32	6.66	8.88	9.99	13.3
Oak, commerical white	.68	11.0	.57	.71	.88	.94	1.03	1.18	1.41	1.77	2.02	2.36	2.83	3.54	4.72	5.66	7.08	9.43	10.61	14.1
Pine, longleaf	.66	9.2	.55	.69	.86	.92	1.00	1.15	1.37	1.72	1.96	2.29	2.75	3.44	4.58	5.50	6.88	9.16	10.32	13.75
Pine, sugar	.37	11.4	.31	.39	.48	.51	.56	.64	.77	.96	1.10	1.28	1.54	1.93	2.57	3.08	3.85	5.14	5.78	7.70
Pine, shortleaf	.54	11.0	.45	.56	.70	.75	.82	.94	1.12	1.40	1.60	1.87	2.25	2.81	3.74	4.49	5.62	7.49	8.43	11.2
Pine, western yellow	.41	10.8	.34	.43	.53	.57	.62	.71	.85	1.07	1.22	1.42	1.71	2.13	2.84	3.41	4.27	5.69	6.40	8.54
Pine, white	.39	9.9	.33	.41	.51	.54	.59	.68	.81	1.02	1.16	1.35	1.62	2.03	2.71	3.25	4.06	5.42	6.09	8.12
Poplar, yellow	.41	6.1	.34	.43	.53	.57	.62	.71	.85	1.07	1.22	1.42	1.71	2.13	2.84	3.41	4.27	5.69	6.40	8.54
Spruce, Sitka	.38	8.9	.32	.40	.49	.53	.58	.66	.79	.99	1.13	1.32	1.58	1.98	2.64	3.16	3.96	5.28	5.94	7.94
Sycamore	.50	9.2	.42	.52	.65	.69	.76	.87	1.04	1.30	1.49	1.73	2.08	2.60	3.47	4.16	5.20	6.94	7.82	10.41
Tanguile (Philippine mahogany)	.54	11.8	.45	.56	.70	.75	.82	.94	1.12	1.40	1.60	1.87	2.25	2.81	3.74	4.49	5.62	7.49	8.42	11.20
Walnut, black	.57	4.8	.47	.59	.74	.79	.86	.99	1.19	1.48	1.70	1.98	2.37	2.97	3.96	4.75	5.94	7.92	8.91	11.87

Weight of glue per square foot of single glue line, blood albumen about 0.3 ounce; casein about 0.4 ounce.

Explanation of Table 4.

This table lists the tensile strength of three-ply wood of various common veneer species and the approximate strength of single-ply wood. The strength figures, given in pounds per square inch, correspond to the moisture contents listed.

Sample computation: To obtain the tensile strength of three-ply wood consisting of two 1/20-inch birch faces and a 1/16-inch basswood core.

Tensile strength parallel to face grain $=2\times 1/20\times 19{,}820=1{,}982$ pounds per inch of width.

Tensile strength perpendicular to face grain $=1\times 1/16\times 10{,}320=645$ pounds per inch of width.

This computation neglects the tensile strength of the ply or plies perpendicular to the grain, which is comparatively small, and the results are therefore slightly in error.

The mechanical properties of wood are quite variable, and the strength of individual pieces may be expected to differ considerably from the average values given.

Table 4.—*Tensile strength of plywood and veneer.*

Species.	Number of tests.	Moisture content at test.	Specific gravity [1] of plywood.	Tensile strength [2] of 3-ply wood parallel to grain of faces.	Tensile strength [3] of single-ply veneer 1½ (d).
		Per cent.		*Pounds per square inch.*	*Pounds per square inch.*
	(a)	(b)	(c)	(d)	(e)
Ash, black	120	9.1	0.49	6,180	9,270
Ash, commercial white	200	10.2	.60	6,510	9,760
Basswood	200	9.2	.42	6,880	10,320
Beech	120	8.6	.67	13,000	19,500
Birch, yellow	200	8.5	.67	13,210	19,820
Cedar, Spanish	115	13.3	.41	5,200	7,800
Cherry [4]	115	9.1	.56	8,460	12,690
Chestnut	40	11.7	.43	4,430	6,640
Cottonwood [5]	120	8.8	.46	7,280	10,920
Cypress, bald	113	8.0	.45	6,160	9,240
Douglas fir [6]	200	8.6	.48	6,180	9,270
Elm, cork	65	9.4	.62	8,440	12,660
Elm, white	160	8.9	.52	5,860	8,790
Fir, true [7]	24	8.5	.40	5,670	8,510
Gum [8]	35	10.6	.54	6,960	10,440
Gum, cotton	80	10.3	.50	6,260	9,390
Gum, red	182	8.7	.54	7,850	11,780
Hackberry	80	10.2	.54	6,920	10,380
Hemlock, western	119	9.7	.47	6,800	10,200
Magnolia [9]	80	8.8	.58	9,220	13,830
Mahogany, African [10]	20	12.7	.52	5,370	8,060
Mahogany, Philippine [11]	25	10.7	.53	10,670	16,000
Mahogany, true	35	11.4	.48	6,390	9,580
Maple, soft [12]	120	8.9	.57	8,180	12,270
Maple, hard [13]	192	8.0	.68	10,190	15,290
Oak, commercial red	115	9.3	.59	5,480	8,220
Oak, commercial white	195	9.5	.64	6,730	10,100
Pine, sugar	110	8.0	.42	5,530	8,300
Pine, white	40	5.4	.42	5,720	8,580
Poplar, yellow	155	9.4	.50	7,390	11,080
Redwood	105	9.7	.42	4,770	7,160
Spruce, Sitka	121	8.3	.42	5,650	8,480
Sycamore	163	9.2	.56	8,030	12,040
Walnut, black	110	9.1	.59	8,250	12,380
Yucca species	33	7.3	.49	2,210	3,320

[1] Specific gravity based on oven-dry weight and volume at test.
[2] Based on total cross-sectional area.
[3] Based on assumption that center ply carries no load.
[4] Probably black cherry.
[5] Probably (common) cottonwood.
[6] Coast type.
[7] Probably white fir.
[8] Probably black gum.
[9] Probably (evergreen) magnolia.
[10] Probably Khaya species.
[11] Probably tanguile.
[12] Probably silver maple.
[13] Sugar or black.
Data based on tests of 3-ply panels with all plies in any one panel same thickness and species.

REPORT No. 85

MOISTURE RESISTANT FINISHES FOR AIRPLANE WOODS

BY

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Architectural Assistant in Forest Products

FOREWORD.

This publication is one of a series of eight monographs prepared by the Forest Products Laboratory of the Forest Service, United States Department of Agriculture, for publication by the National Advisory Committee on Aeronautics.

The purpose of the series of monographs is to discuss in detail the various requirements of wood for use in aircraft and to make public some of the results obtained in the experimental and testing work of the Forest Products Laboratory undertaken for the Army and Navy during the war.

The subjects discussed will include: (1) Kiln-drying of airplane woods, (2) the effect of kiln-drying on strength, (3) the care of airplane stock, (4) the composition and use of glues, (5) the manufacture and testing of plywood, (6) wood in airplane construction, (7) moisture-resistant finishes, and (8) wood airplane parts.

REPORT No. 85.

MOISTURE-RESISTANT FINISHES FOR AIRPLANE WOODS.

By M. E. Dunlap.

Forest Products Laboratory.

In an investigation made at the Forest Products Laboratory to find an airplane-propeller coating which would prevent moisture changes in the propeller, a great many varnishes, enamels, and other coatings were studied. The study was confined to the ability of the coatings to prevent the passage of moisture either into or out of the wood. Other problems, such as the durability of the finish and how to apply stain, were not studied.

This report deals with the various coatings which were tested and their effectiveness as moisture-proofing agents. The terms "moisture proof" and "moisture resistant" indicate resistance to the passage of moisture through the coating. They do not refer to resistance to discoloration in the presence of moisture.

METHOD OF TESTING.

The tests were made by applying the coatings to panels of yellow birch ⅝ by 4 by 8 inches in size. The surfaces of the panels were carefully smoothed and the corners were rounded. In general, a coat of filler was first applied, followed by three coats of the varnish or other material which was being studied. Some materials which were studied required special methods of application, depending upon their character.

After the panels had dried thoroughly they were subjected for 17 days to an atmosphere with a humidity of 95 to 100 per cent. The absorption of moisture during this period, calculated in grams per square foot of surface exposed, was taken as a measure of the effectiveness of the coating.

MATERIALS USED.

In general, materials of the following types were tested: Oil, wax, oil varnish, enamel, spirit varnish, cellulose varnish, condensation varnish, rubber coating, and metal coating.

Linseed oil and wax treatment.—The first specification issued by the Air Service for coating propellers required five coats of linseed oil applied hot and two coats of floor wax. This coating reduced the absorption of moisture only very slightly.

Impregnation treatments.—Test panels were treated by forcing various material, including varnish, china-wood oil, and cellulose varnishes, into the wood. These treatments were made by immersing the specimens in the liquid in a treating cylinder and drawing a vacuum until a considerable amount of the air had been removed from the wood. The vacuum was then released and atmospheric pressure was allowed to force the liquid into the wood. Specimens were usually allowed to stand in the liquid overnight, after which they were removed, wiped off, and dried. The moisture resistance of the wood was only slightly increased by these treatments.

Condensation varnish.—Considerable time was spent in experimenting with this material. It is a patented product made from formaldehyde and phenol and is usually thinned with alcohol. It is extremely hard to apply as a coating because of the rapid evaporation of the thinner and because it must be baked for several hours at a temperature of about 180° F.

The resulting coating was not very effective on account of holes, was extremely hard and brittle, and if it became broken it was subsequently further cracked by the swelling of the wood through absorption of moisture. This type of coating was given up as entirely unsuitable. Crazing was also a fault of this material when used as a varnish.

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Oil varnish.—In panels made by applying three coats of varnish over a filler, the reduction in moisture absorption of the natural wood averaged about 66 per cent for 43 spar varnishes and about 85 per cent for 3 rubbing varnishes.

Enamels.—These are varnishes to which pigment has been added. Some interesting points were brought out in this test. It was found that the addition of a pigment to a spar varnish added materially to the moisture resistance of the coating. This was true of all pigments which were tried, including white lead, red lead, orange mineral, and barytes. The pigment barytes showed great superiority over the others tested. It has poor covering properties, and where equal parts of barytes and varnish by weight are mixed and ground together the transparency of the coating is little affected. This coating prevented the absorption of about 92 per cent of the moisture which would be absorbed by natural wood under the same conditions.

In this connection it might be mentioned that if a varnish be allowed to dry almost dust free, and then be rubbed with aluminum powder, a slight improvement in the moisture resistance of the coating will be obtained. Aluminum powder added to the varnish itself will also produce beneficial results.

Cellulose varnishes (pyroxylin).—A number of tests were made on these materials, although they are not commonly used for wood. In ability to prevent moisture absorption they fell in the same class as spar varnishes. The most successful way to apply them is by means of dipping machines or air brushes.

Rubber coatings.—Some tests were made of hard-rubber coatings which were applied to wood and vulcanized. The coating was found to be moisture proof. It did not adhere well enough to the wood, however, to pass whirling tests applied to propellers.

Sprayed metal.—The Schoop process of coating wood with molten metal was studied, using aluminum and copper. It was found that the coating produced was not perfect and did not adhere to the wood well enough for the purpose intended. The coating is also quite heavy. This method is used ordinarily in ornamental work.

Electroplated coatings.—In this case a moisture-proof coating was obtained. It was found, however, that the coating showed little strength, did not adhere sufficiently to the wood, and was quite heavy. For these reasons it appeared to be entirely unsuitable for use on propellers. No information is available concerning the method used in applying this coating. It appears that a coating of finely powdered copper was applied over the surface of the wood, which was then placed in a bath and the coating of copper applied electrolytically.

Metal-leaf coatings.—The most successful practical coating found consisted of a varnish or enamel coating in which aluminum leaf was incorporated.

Two types of aluminum-leaf finishes were used, which are about equally resistant to moisture transmission. One makes use of spirit varnish and the other of oil varnish, the successive coats being as follows:

> *Spirit-varnish type.*—Filler, plus 1 or 2 coats orange shellac, plus 1 coat spar varnish used as a size, plus 1 coat aluminum leaf, plus 2 coats of orange shellac with desired color, plus 1 coat spar varnish.
>
> *Oil-varnish type.*—Filler, plus 1 or 2 coats spar or rubbing varnish, plus 1 coat spar varnish used as a size, plus 1 coat aluminum leaf, plus 2 coats spar varnish or enamel.

Coating wood with metal leaf is not nearly so slow a process as laying leaf in sign making. As soon as the size reaches the right condition, the leaf can be applied by unskilled workmen directly from the book without the aid of gilders' tips. The time required to apply leaf to a propeller should not be more than 40 or 50 minutes, and this could be reduced as the finisher becomes more experienced.

It is important to allow the size to reach the proper condition before attempting to lay the leaf; the right point is just before the varnish sets dust free. The time required after application to reach this condition varies with the type of varnish used, but for spar varnishes it

PLATE I.—METHOD OF APPLYING ALUMINUM LEAF TO PROPELLER.

is usually from 1½ to 2 hours. The workman will soon learn how to judge the condition of the size by touching it lightly with his fingers. The size will dry much more quickly if it is thinned about one-fourth with turpentine. It should be applied as sparingly as possible.

To apply the complete finish of the spirit-varnish type requires in the neighborhood of 10 hours and to dry the various coats about 90 hours, making the total time about 100 hours. The oil-varnish finish takes longer to dry and would probably total about 240 hours. The latter finish is possibly the more durable.

Different numbers of coats of varnish.—An interesting series of experiments was carried out by applying different numbers of coats of varnish (from 2 to 12) to standard test specimens. These tests were made in such a way that all panels received their final coat on the same day. Two spar varnishes of medium moisture resistance and one of good moisture resistance were used. The two varnishes of medium grade gained in resistance as the number of coats increased up to about six coats; after this point there was practically nothing gained by the addition of subsequent coats. In the case of the more resistant varnish an increase in moisture resistance was observed with the increased number of coats up to 12, although the gain became less as the number of coats increased.

Dipping tests.—A dipping machine was used in carrying out part of the tests, and it was found that a very smooth and uniform coating could be applied. The panel was completely immersed in the varnish and drawn out very slowly. The thickness of the coating can be easily regulated by using a suitable speed of withdrawal. Slightly greater moisture resistance is obtained by this method than by the brush method. This is due probably to the greater uniformity of the coating.

Brush coating.—Practically all of the varnish was applied by brush except as already noted.

Forced drying.—A series of panels was tested by drying in an oven at 110° F. The process used is as follows: Apply the varnish and allow it to set over night, place the specimens in the oven and dry them for four hours, remove, cool, and recoat. This method was found better than putting the panels in the oven immediately after coating, since there was less tendency to blister.

Effectiveness of the same coating applied to different woods.—Three coats of spar varnish were applied to a number of specimens of about a dozen different species, and it was found that there was little difference in the results obtained.

Figure 1 shows the relative effectiveness of different materials and methods of treatment in preventing the absorption of moisture by wood.

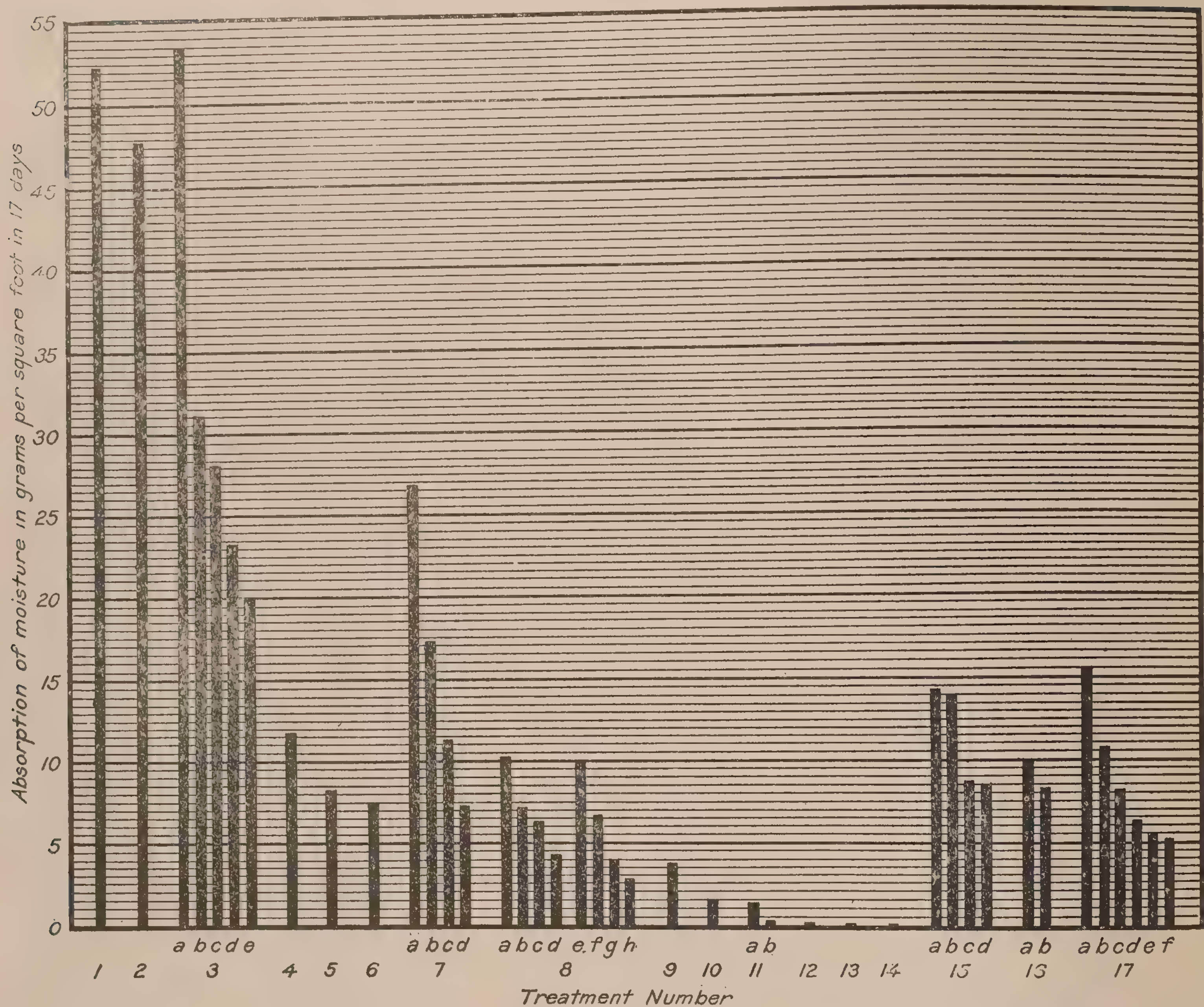

FIG. 1.—The relative effectiveness of different materials and methods of treatment in preventing the absorption of moisture by wood when exposed for 17 days to a humidity of 95 to 100 per cent.

TREATMENT.

1. Natural wood—no treatment.
2. 5 coats of linseed oil plus 2 coats of wax.
3. Impregnation treatments—
   - *a* Linseed oil (soaking).
   - *b* Paraffin and gasoline (vacuum and pressure).
   - *c* Beeswax (vacuum and pressure).
   - *d* Spar varnish (vacuum and pressure).
   - *e* Cellulose varnish (vacuum and pressure).
4. 3 coats of cellulose varnish.
5. Filler plus 3 coats of orange shellac.
6. Filler plus 3 coats of rubbing varnish.
7. Filler plus 3 coats of spar varnish.
   - *a* Poorest of 43 varnishes tested.
   - *b* Average of 43 varnishes tested.
   - *c* Average of 10 best varnishes tested.
   - *d* Best varnish tested.
8. Enamels—Filler plus—
   - *a* 2 coats enamel (red-lead pigment) plus varnish.
   - *b* 2 coats enamel (aluminum powder) plus varnish.
   - *c* 2 coats enamel (white-lead pigment) plus varnish.
   - *d* 2 coats enamel (barytes pigment) plus varnish.
   - *e* 3 coats commercial enamel (average of 11 brands).
   - *f* 3 coats commercial enamel (best brand).
   - *g* 2 coats laboratory mixture plus varnish (average of 10 best mixtures).
   - *h* 3 coats laboratory mixture plus varnish (best laboratory mixture).
9. Filler plus 3 coats of shellac and aluminum powder.
10. 5 coats of bakelite plus 5 coats of varnish.
11. Metal leaf coatings—Filler plus shellac or varnish under coat plus varnish size plus aluminum leaf plus 2 coats of varnish shellac or enamel.
    - *a* Average of all types.
    - *b* Best type.
12. Sprayed with copper or aluminum and coated with 3 coats of varnish.
13. Electroplated with copper.
14. Vulcanized rubber coating (1 small specimen tested).
15. Forced drying (filler and 3 coats)—
    - *a* Average of 23 varnishes (room dried).
    - *b* Same varnishes (dried at 110° F.).
    - *c* Average of 5 enamels (room dried).
    - *d* Same enamels (dried at 110° F.).
16. Brushed versus dipped coatings—
    - *a* Filler plus 3 brushed coats (average of 7 varnishes).
    - *b* Filler plus 3 dipped coats (average of same varnishes).
17. Filler plus different numbers of coatings—
    - *a* 2 brushed coats.
    - *b* 4 brushed coats.
    - *c* 6 brushed coats.
    - *d* 8 brushed coats.
    - *e* 10 brushed coats.
    - *f* 12 brushed coats.

REPORT No. 86

PROPERTIES OF SPECIAL TYPES OF RADIATORS

By S. R. PARSONS
Bureau of Standards

REPORT No. 86.

PROPERTIES OF SPECIAL TYPES OF RADIATORS.

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This report describes an investigation of properties of special types of radiators, conducted for the National Advisory Committee for Aeronautics at the Bureau of Standards.

In respect to the relation of power absorbed to power dissipated, flat plate radiators have been found to be markedly superior to other types commonly employed, for use in unobstructed positions on airplanes flying at the higher speeds.

For such use, a pitch of one-half inch (1.27 cm.) between plates gives a higher figure of merit than the closer spacings.

The depth of such a radiator should be considerably greater than of the honeycomb types, and the most efficient depth is greater with the higher speeds. For a radiator of one-half-inch pitch to be used at a speed of 120 miles per hour (53.6 meters per second), the most efficient depth appears to be about 13 inches (33 cm.). A small increase above this depth has but a slight effect, however, on the figure of merit, and if it is desirable to reduce the frontal area of the radiator to a minimum, the depth may be increased to 20 inches (51 cm.) or more without serious reduction of efficiency.

Equations and plots are given, by means of which the properties of flat plate radiators may be obtained for various depths and spacings.

The plates should be continuous from front to rear of the radiator, in order to avoid excessive head resistance caused by eddies in the air stream.

Radiators of the "fin and tube" types are characterized by high head resistance, and low heat transfer at high speeds. With the possible exception of certain compact types for use in the wing they are unsuitable for airplane use.

The construction of some types of radiators is such that at certain air speeds they produce a whistling sound. These so-called "whistling radiators" are characterized by the following points:

(1) Unusual conditions of air flow, resulting in irregularities in the relations between different properties. For example, mass flow of air through the radiator is not proportional to free air speed as in ordinary radiators, and head resistance is not proportional to the square of free air speed.

(2) High heat transfer *for a given mass flow of air through the radiator.*

(3) Very low flow of air through the radiator *for a given free air speed.*

(4) In many cases, low heat transfer *for a given free air speed.*

(5) Very high head resistance and horsepower absorbed.

(6) Low figure of merit.

If water tubes made of smooth flat plates, continuous from front to rear of the radiator, are substituted for the rows of tubes of the whistling radiators described, the figure of merit will be greatly increased.

Different radiators show different effects on being inclined at an angle to the direction of the air stream, but in general the results (for angles up to 30°) are as follows:

(1) Decrease in air flow through the core;

(2) Increase in head resistance;

(3) In some cases slight increase in the heat transfer, for angles up to 20° or 25°.

PROPERTIES OF FLAT PLATE RADIATORS.

In the course of the investigation of airplane radiators, being carried on at the Bureau of Standards for the National Advisory Committee for Aeronautics, it has become evident that careful consideration should be given to a type of radiator consisting of water tubes which are thin flat hollow plates placed edgewise to the air stream. In regard to the relation of heat dissipation to absorption of power, radiators of this type have been shown to be markedly superior to other types now commonly employed for use in unobstructed positions on the faster planes. These conclusions are based on a comprehensive series of tests made to determine what properties such radiators may be expected to possess.

For the study of heat dissipation, three radiators were used, each $9\frac{3}{4}$ inches deep and with plates one-sixteenth inch thick, spaced one-fourth, three-eighths, and one-half inch respectively, center to center. They are the types E–6, E–7 and E–8 of Technical Report No. 63, Part I, and their properties and characteristics are given in that report. The curves are reproduced in plots 2, 3, and 4.

For the study of mass flow of air through the core and of head resistance, a set of 22 dummy radiators were constructed, using depths of 2, 4, 8, and 12 inches, spacings of one-fourth, three-eighths, one-half, three-fourths, and 1 inch, and thicknesses of one-sixteenth and one-eighth inch. The plates were made by covering cardboard of the proper thickness with thin sheet copper, and were used again and again. They were 12 inches long and with the exception of the one-fourth inch spacing, enough were used to make a radiator 12 inches wide, so that the frontal area was 1 square foot. Since the copper was thin it was not possible to obtain as smooth a surface as might have been desired, but the condition was fairly representative of what would be found in commercially manufactured radiators.

Extrapolations for depths of radiators not covered by the laboratory tests were made with the use of three empirical equations, given below. All quantities are based on one square foot frontal area of the radiator and an air density of 0.0750 pound per cubic foot.

Mass flow of air is given by the equation [1]

$$M=0.110\sqrt{\frac{p-t}{p}}(V)\left(1-e^{-10.95\sqrt{\frac{p-t}{x}}}\right) \tag{1}$$

where M = mass flow of air in pounds per second per square foot frontal area of radiator;
V = free air speed, in miles per hour;
t = thickness of plates, in inches;
p = pitch, in inches;
x = depth of radiator, in inches; and
e = base of natural logarithms.

Heat dissipation is given by the equation

$$Q=34.8\ M\left(1-e^{-\frac{Bx}{M^A}}\right) \tag{2}$$

where Q = heat dissipation in H. P. per 100° F. temperature difference and A and B are empirical constants.

The constants A and B were determined from actual heat tests on the three radiators mentioned above, and are as follows:

$\frac{1}{4}$-inch pitch: $A=0.24$; $B=0.0616$;
$\frac{3}{8}$-inch pitch: $A=0.11$; $B=0.0283$;
$\frac{1}{2}$-inch pitch: $A=0.23$; $B=0.0258$.

[1] The basis for the empirical equations (1) and (2) may be briefly stated as follows:
The air flow is found to be proportional to the free air speed, and may be expected to be proportional to the density of the air ρ, so that

$$M=b\ \rho\ V \tag{1'}$$

in which b may be expected to be a function of the "free area" a, the depth x, and the distance between plates $(p-t)$. From the form of the equa-

It was assumed that the head resistance might be regarded as made up of two parts: (1) that due to the impact of air on the edges of the plates, including suction on the rear; and (2) that due to skin friction of the air passing over the plates. The results were very well represented by the equation

$$R=n\ V^2\ (0.00016\ t+0.0000025\ x) \tag{3}$$

where R=head resistance in pounds per square foot frontal area; and

n=number of plates per foot width of core.

The values given by this equation are a little low when n equals 48; i. e., with one-fourth-inch pitch, but for spacings greater than one-fourth inch, the resistance per plate was found to be constant for a given speed and air density.

The weights of the radiators and the water contained in them were estimated from geometrical considerations and the densities of copper and solder, assuming the Lepère type of construction. The weight is given by the equation

$$W=0.0557\ n\ x \tag{4}$$

where W=weight of core and contained water, in pounds per square foot frontal area.

A lift-drift ratio of 5.4 was assumed for the airplane, and the horsepower absorbed (per square foot) is

$$H.\ P.=\left(R+\frac{W}{5.4}\right)\left(\frac{V}{375}\right) \tag{5}$$

Figure of merit is the ratio of the power dissipated to the power absorbed.

$$F.\ M.=\frac{Q}{H.P.} \tag{6}$$

---

tion, b must be dimensionless, and therefore x and $(p-t)$, each having the dimension of a length, must enter as a ratio, and b may be of the form

$$b=f\left(a,\ \frac{p-t}{x}\right) \tag{2'}$$

Also $b=0$ when $a=0$ and $p-t=0$
and $b=1$ when $a=1$ and $x=0$.

These conditions are satisfied by the equation

$$b=\sqrt{a}\left(1-e^{-k\sqrt{\frac{p-t}{x}}}\right) \tag{3'}$$

and tests on about 60 types of radiators give for k a value of 10.95 for the units used. Substitution of equation (3′) in (1′), with proper units and the assumed value of air density, give equation (1).

For heat transfer, extending the above notation, let

T=temperature difference between water in the radiator and air passing through it
c=specific heat of the air
y=total perimeter of the air tubes (in unit frontal area) around a section perpendicular to the direction of air flow
q=heat transfer per unit surface per unit time per unit temperature difference
T_0=value of T at front face of the radiator.

If it is assumed (1) that the heat transfer varies as the temperature difference, (2) that the transfer coefficient q is independent of the depth x, and (3) that the temperature change in the water with depth of core is negligible in comparison with the temperature difference T, so that the change in T may be regarded as due entirely to the rise in air temperature; then equating the heat gained by the air to that dissipated from the surface gives

$$M\,c\,d\,T=-q\ T\,y\,dx \tag{4'}$$

which integrates to the form

$$T=T_0\,e^{-\alpha x} \tag{5'}$$

where

$$\alpha=\frac{qy}{c\,M}$$

The heat transfer is given by the equation

$$Q=\int^x q\ T\,y\,dx \tag{6'}$$

which gives the form

$$Q=c\ M\ T_0\left(1-e^{-\alpha x}\right) \tag{7'}$$

Experiment shows that q varies as M^{1-A}, where A is about 0.2, but varies with different types of radiators.

If T_0 is taken as 100°F., and proper units are used, equation (7′) reduces to equation (2).

The following example illustrates the use of the equations. Let it be required to obtain the figure of merit of a radiator with plates one-sixteenth inch thick, one-half inch pitch, 16 inches deep, and at 120 miles per hour.

In equation (1) for mass flow of air,

$$M=0.110\sqrt{\frac{p-t}{p}}(V)\left(1-e^{-10.95\sqrt{\frac{p-t}{x}}}\right)$$

p = pitch = 0.5

t = thickness = $\frac{1}{16}$ = 0.0625

x = depth = 16

V = speed = 120

$$M=0.110\sqrt{\frac{0.5-0.0625}{0.5}}(120)\left(1-e^{-10.95\sqrt{\frac{0.5-0.0625}{16}}}\right)$$

$$M=0.110\sqrt{0.875}\ (120)\ (1-e^{-1.81})$$

M = 12.35 (0.836) = 10.33 lb. per sq. ft. per sec.

In equation (2) for energy dissipated,

$$Q=34.8\ M\left(1-e^{-\frac{Bx}{M^A}}\right)$$

A = 0.23

B = 0.0258

$$A=34.8(10.33)\left[1-e^{-\frac{(0.0258)\ (16)}{10.33^{.23}}}\right]$$

$$Q=359\ (1-e^{-0.241})=359\ (0.215)$$

Q = 77.3 $H.\ P.$ per sq. ft. per 100° F.

In equation (3) for head resistance,

$$R=n\ V^2\ (0.00016\ t+0.0000025\ x),$$

n = 24

$$R=24\ (120)^2\left(\frac{0.00016}{16}+0.0000025\ (16)\right)$$

$$R=24\ (120)^2\ (0.00001+0.00004)$$

R = 17.27 pounds per square foot.

From equation (4),

W = 0.0557 (16) (24) = 21.4 lb. per sq. ft.

From equation (5),

$$H.\ P.=\left(R+\frac{W}{5.4}\right)\left(\frac{V}{375}\right)$$

$$H.\ P.=\left(17.27+\frac{21.4}{5.4}\right)\left(\frac{120}{375}\right)=6.80\ H.\ P.\text{ per sq. ft.}$$

From equation (6),

$$F.\ M.=\frac{Q}{H.\ P.}=\frac{77.3}{6.80}=11.4$$

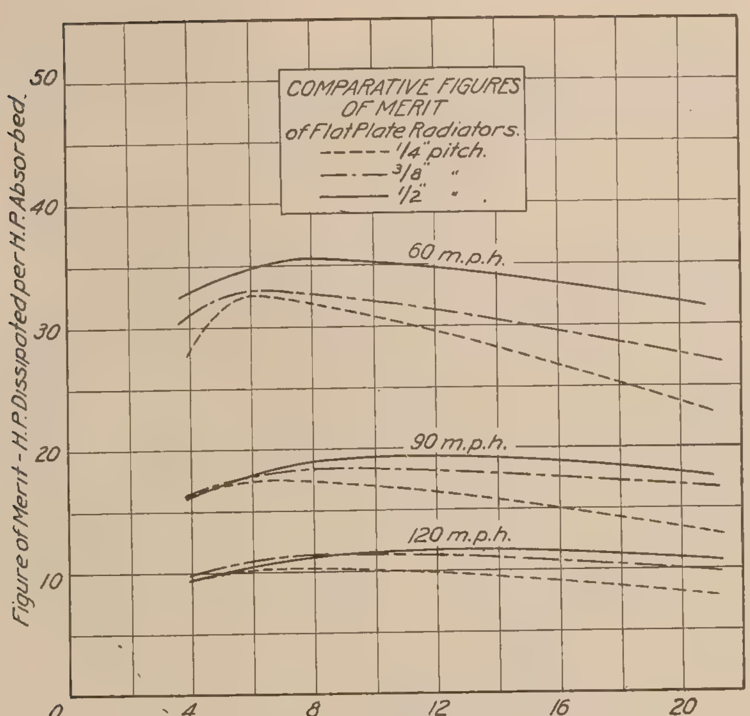
DESCRIPTION OF CURVES.

Plot 1 shows figure of merit computed with the aid of the above equations for various depths and for speeds of 60, 90, and 120 miles per hour, and for spacings of one-fourth, three-eighths, and one-half inch between plates. The curves illustrate the following points:

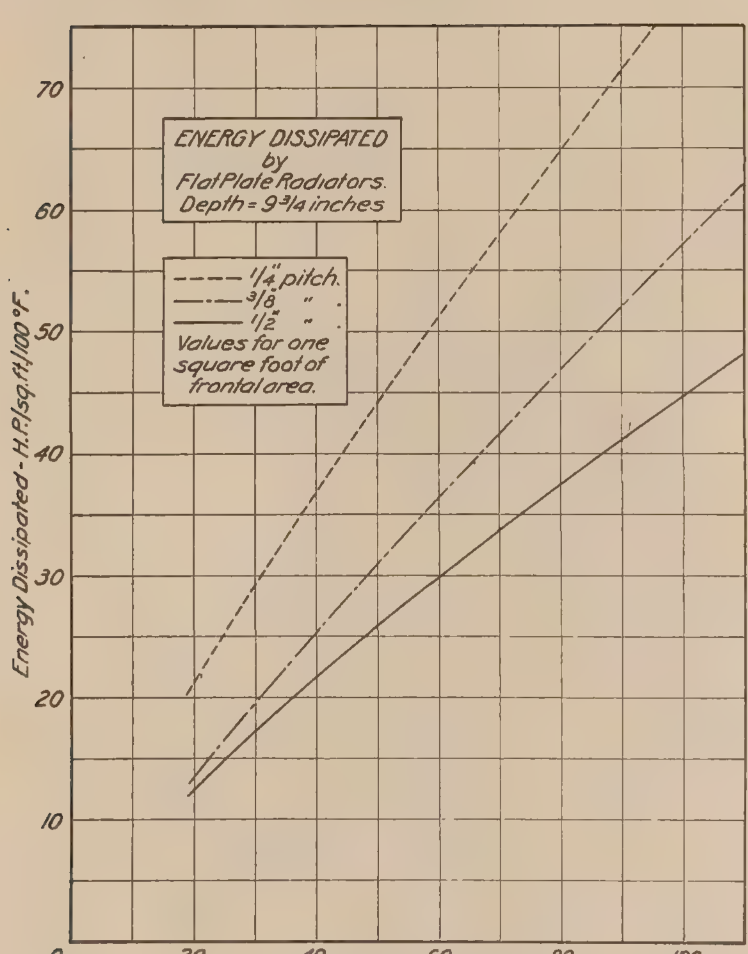
(1) The one-half inch pitch gives, in general, a higher figure of merit than those of closer spacings.

(2) For high speeds the radiator may be somewhat deeper than for low speeds.

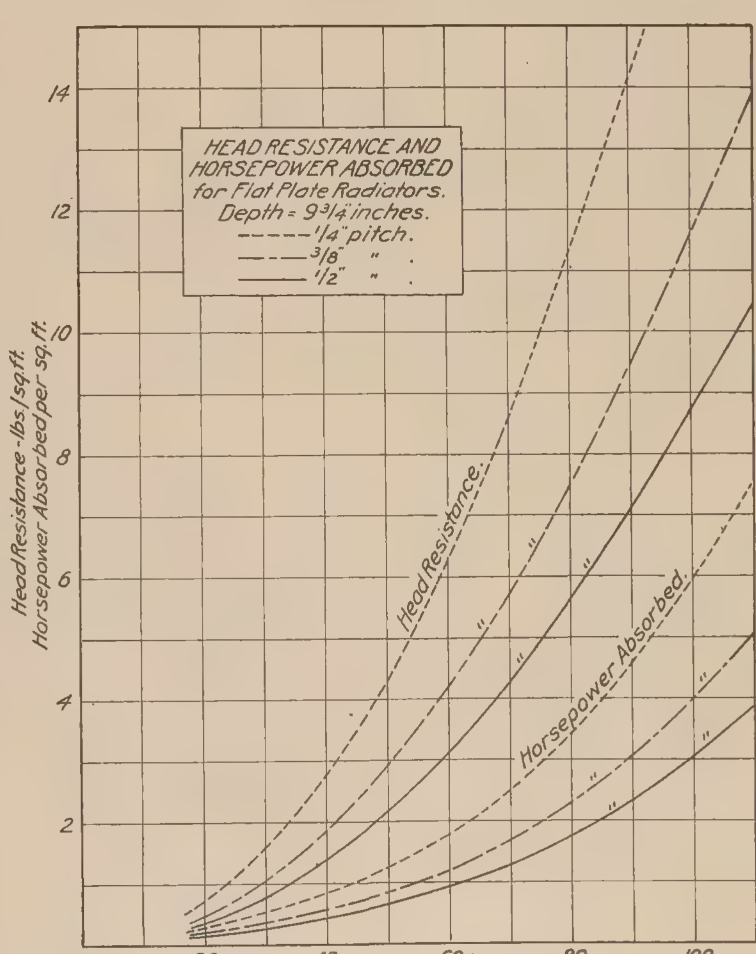
(3) For the higher speeds the most efficient depth is considerably greater than those in common use with the cellular types of core.



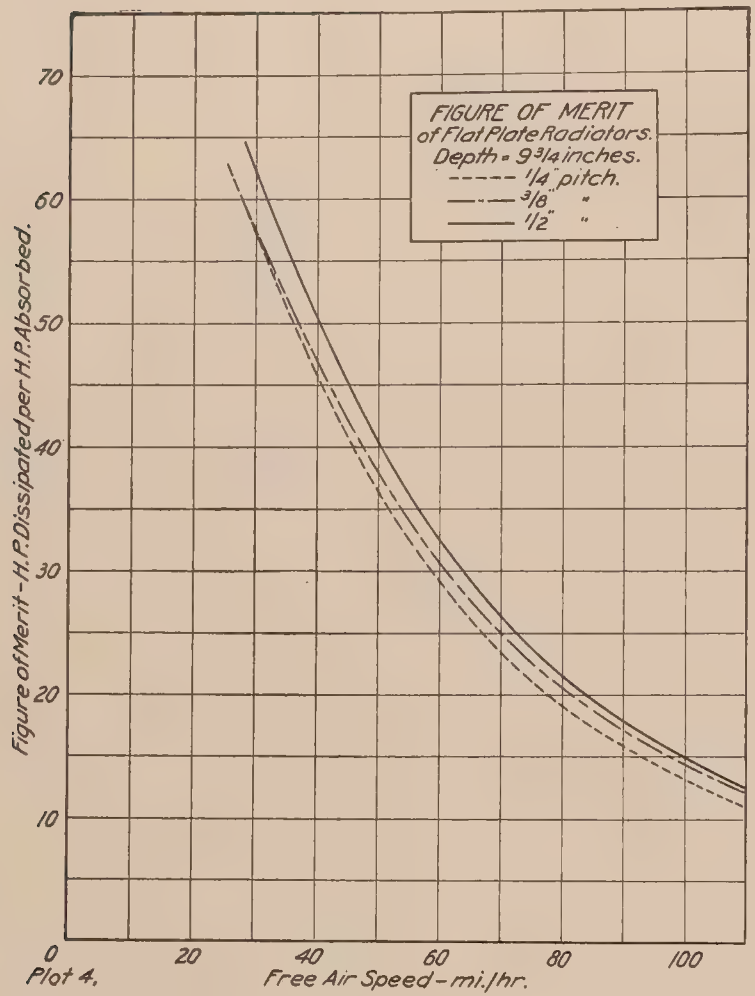
Plot 1.



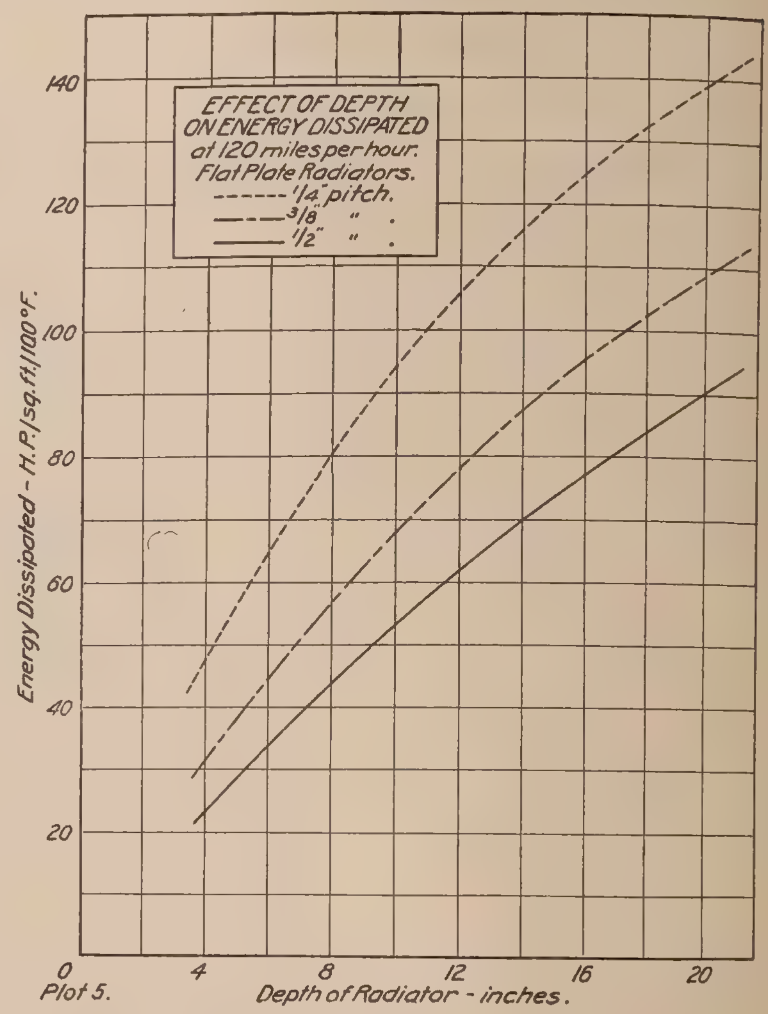
Plot 2.



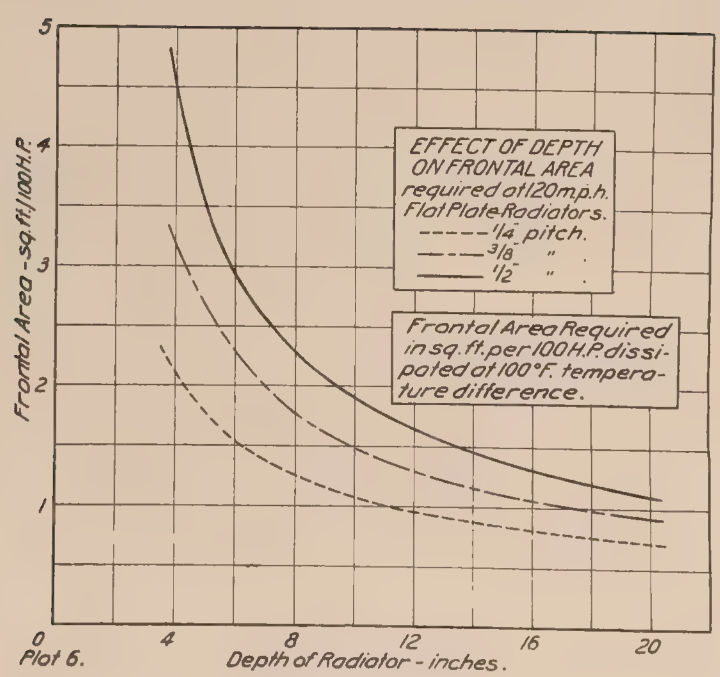
Plot 3.



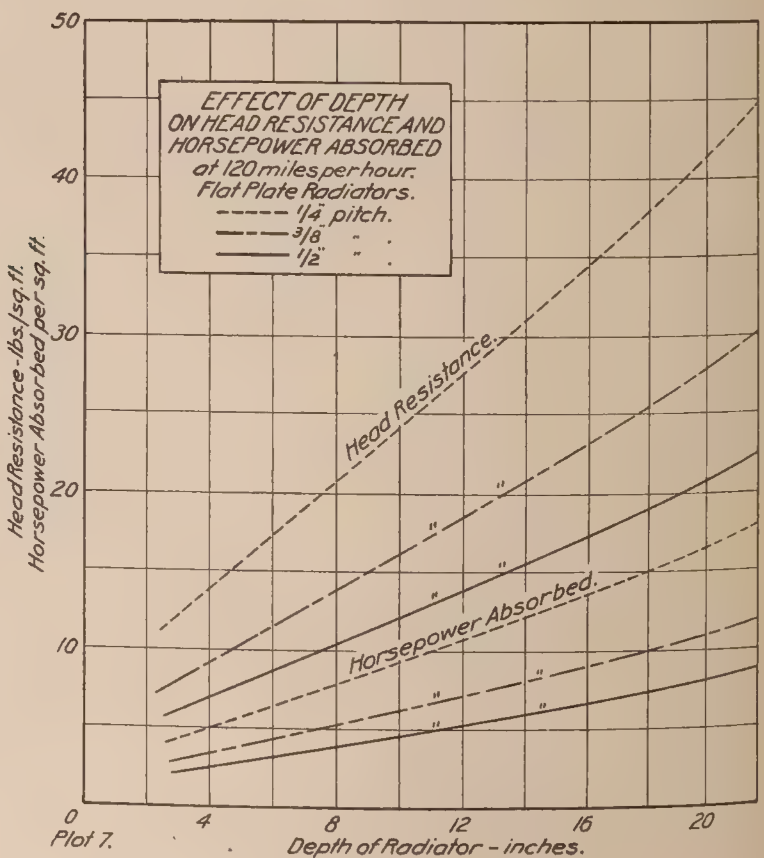
Plot 4.



Plot 5.



Plot 6.



Plot 7.

(4) For the higher speeds, the figure of merit is practically at its maximum value over a considerable range of depth, so that if considerations of compactness make it desirable to reduce the frontal area to a minimum, a reasonable increase in depth beyond the optimum will have but a small effect on the figure of merit.

The properties of the three radiators tested for heat transfer are reproduced from Technical Report No. 63, Part I, in plots 2, 3, and 4.

The effect of depth on the properties of the radiator is shown in plots 5, 6, and 7. The values of "area required per 100 H. P." in plot 6 are 100 times the reciprocals of the corresponding values of energy dissipated (in H. P./sq. ft.) of plot 5.

The following example illustrates one way in which the curves may be used. For a radiator with one-half inch pitch, to be used at 120 miles per hour, the maximum figure of merit is given on plot 1 as 11.7 at 13 inches depth. From plot 6, the frontal area of radiator required with 13 inches depth is 1.55 square feet per 100 horsepower. If, in order to reduce the frontal area, a depth of 20 inches should be used, the area required would be 1.10 square feet per 100 horsepower (from plot 6), and the figure of merit would be 11.0 (from plot 1). The horsepower absorbed would be increased from 5.6 to 8.1 per square foot (plot 7). Since, however, the frontal area may be reduced in the ratio $\frac{1.10}{1.55}$, the actual power absorbed that should be compared with the value for 13 inches depth would be $\frac{1.10}{1.55}$ (8.1), or 5.75. These results may be summarized as follows:

	13 inches.	20 inches.	Change.
			Per cent.
Area required, sq. ft./100 H. P.	1.55	1.10	−29
Horsepower absorbed, per square foot required at 13-inches	5.6	5.75	+ 3
Figure of merit	11.7	11.00	− 6

A careful distinction should be made between radiators whose water tubes are smooth, flat plates and other types using perforated plates, or deep and narrow tubes placed in rows, one behind the other. Holes in the water tubes, or spaces between them in the direction of the air flow, cause very great increase in head resistance and decrease in mass flow of air; and although the heat transfer per square foot of cooling surface may be increased by the great turbulence caused, it is at a very heavy cost in head resistance, and with a decrease in figure of merit.

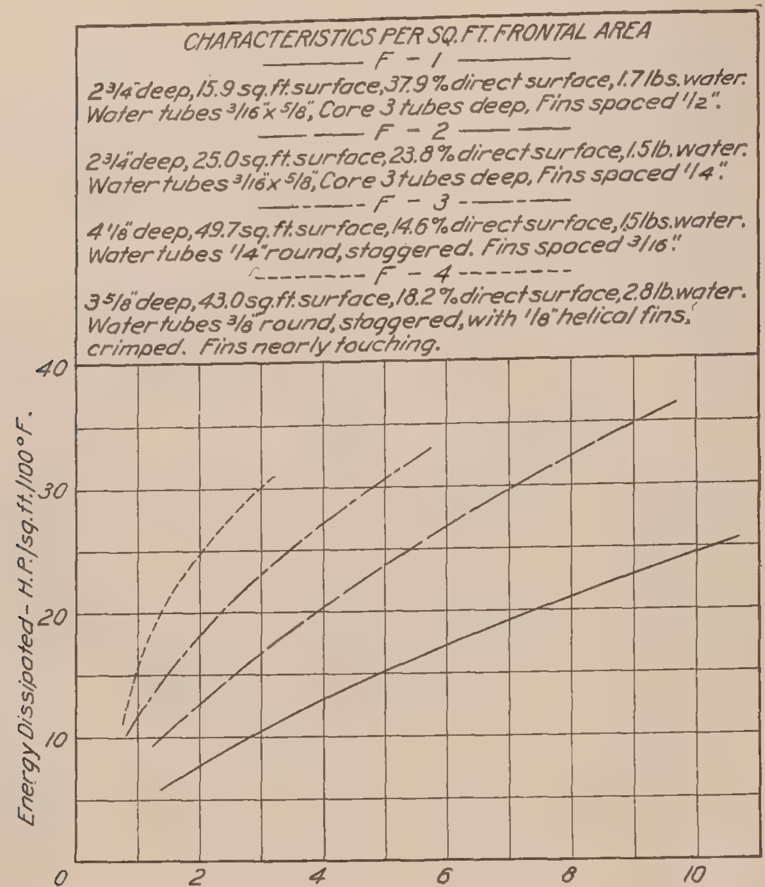
The effect of holes in the water tubes, and of spaces between them in the direction of the air flow is taken up later in this report under "Properties of whistling radiators," in which the properties of six such types are given.

PROPERTIES OF FIN AND TUBE RADIATORS.

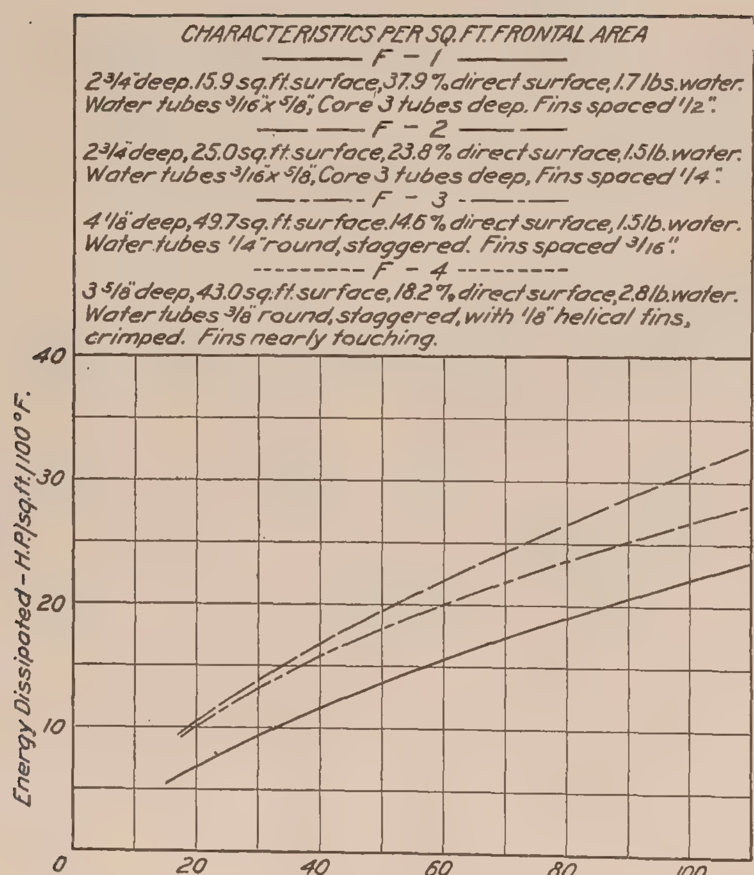
The properties of "fin and tube" types of cooling radiators are distinctly different from those of the better cellular types and warrant special mention.

In general, the fin and tube types are characterized by high head resistance and by low heat transfer at high speeds. The low heat transfer is accounted for by the small flow of air through the radiator and by the large amount of indirect cooling surface. For radiators as ordinarily made, with depths of 5 inches or less, head resistance has been shown to be due principally to the impact of the air on the front face and suction on the rear, and only to a small degree to skin friction on the walls of the air passages. With the fin and tube construction the effects of impact and suction are exaggerated; for each separate water tube is subjected to impact on one side and suction on the other, with the result that the total (projected) area subjected to impact and suction—on all tubes—is much greater than that necessitated in a radiator of the same size but of cellular construction. To the effect of this impact and suction must be added the effect of skin friction on the fins.

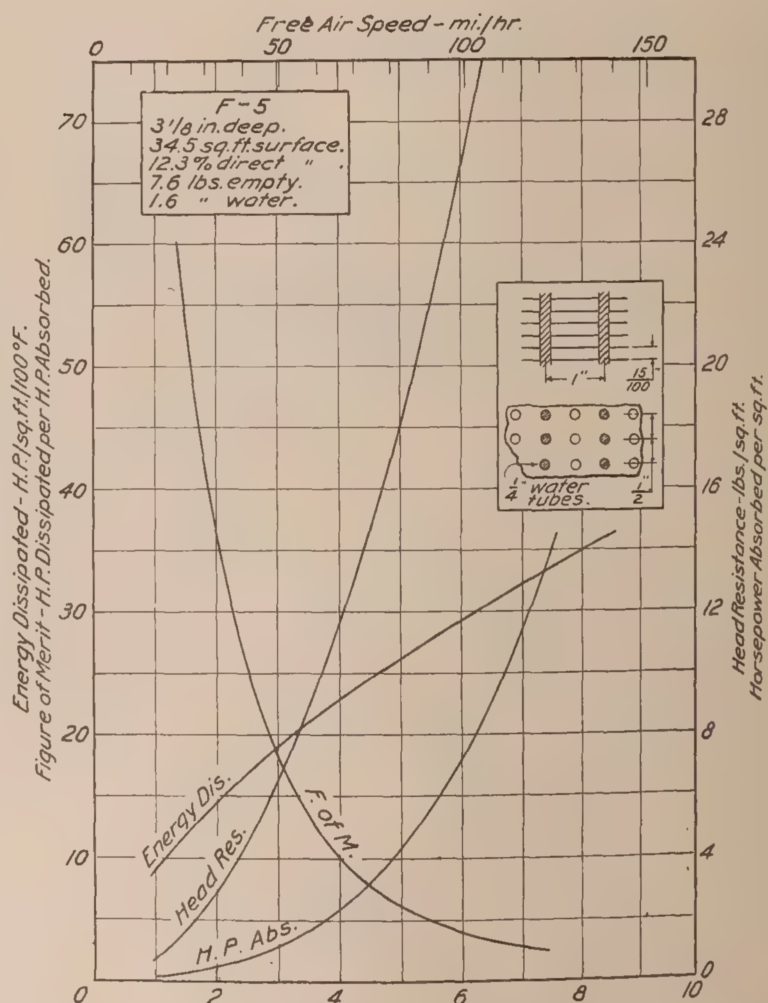
Excessive head resistance, accompanied by low heat transfer, make the fin and tube types unsuitable for use on an airplane where they would be exposed to a current of air at a high speed.



Plot 1.



Plot 2.



Plot 3.

The more compact types, however—notably F–4, which has large water tubes with crimped spiral fins nearly touching each other—show a relatively high rate of heat transfer at very low speeds, which justifies their extensive use on trucks and lower speed automobiles. Indeed the type F–4 would doubtless dissipate a considerable amount of heat with convection currents only.

The accompanying curves show the heat transfer (energy dissipated) for five types of core, in terms of the mass of air flowing through the core. The heat transfer is expressed in horsepower per square foot of frontal area, for a difference of 100° F. between the mean temperature of the water and the temperature of the entering air. Mass flow of air is expressed in pounds per second per square foot of frontal area. Energy dissipated is also shown in terms of free air speed; that is, the speed of the machine on which the radiator is mounted. Plot 3, reproduced from Technical Report No. 63, Part I, shows also the figure of merit of the radiator, which is the ratio of the horsepower dissipated to the horsepower absorbed.

The attempt to determine the relation between the mass flow of air through the core and free air speed was unsuccessful in the case of the type F–4 (with spiral fins); for the air flow was too small to be measured with the instrument used on the other radiators.

In general, it may be stated that fin and tube radiators are unsuitable for airplane use, with the possible exception of a type similar to F–4, placed in the wing, where the mass flow of air must be very small (even less than for the nose position), and consequently head resistance is not necessarily a detriment.

PROPERTIES OF WHISTLING RADIATORS.

Certain types of cooling radiators that whistle in an air stream show peculiar properties, and while radiators of this construction are not recommended, they are being used to some extent, and appear to be worthy of special mention.

TYPES OF CORE TESTED.

The whistling radiators tested at the Bureau of Standards fall into two general classes:

(1) Plain water tubes, and

(2) Perforated water tubes.

The photographs show the forms of construction. In each case the radiator is made up of separate water tubes about 2 inches deep (in the direction of air flow), arranged in rows. In addition, the plain tube type has fins spaced 2 inches apart.

The test sections included one of the plain tube type, designated as E–9, and five perforated tube types, made up in different depths, and with different spacings between rows of tubes. These sections are designated as follows:

E–1, $\frac{5}{16}$-inch pitch, 2 tubes deep.
E–2, $\frac{5}{16}$-inch pitch, 3 tubes deep.
E–3, $\frac{5}{16}$-inch pitch, 4 tubes deep.
E–4, $\frac{1}{2}$-inch pitch, 4 tubes deep,
E–5, $\frac{5}{8}$-inch pitch, 4 tubes deep.

CAUSE AND EFFECT OF THE WHISTLE.

The form of construction leaves continuous air passages *across* the radiator; that is, perpendicular to the direction of the air stream. In the plain tube type these air passages occur between the water tubes, and in the perforated tube types not only between the tubes but at each perforation. All through the radiator there are short columns of air, across the ends of which air is blowing, with the result that vibrations are set up in the short columns and perpendicular to the air stream. The resulting whistle will of course vary widely in intensity and in pitch as the speed of the air stream varies, and conditions of resonance have very marked effects, not only upon the sounds, but upon the properties of the radiator.

By the vibrations of the cross columns, air is alternately being forced into and withdrawn from the fast moving stream. Air drawn out of the stream will be retarded and air forced into it will be accelerated, thus acting as a drag on the stream. These two effects cause a great decrease in the flow of air through the radiator and a great increase in head resistance.

At the same time the very great turbulence caused in the air stream results in a high heat transfer per square foot of cooling surface *for a given mass flow of air through the radiator*, and this increase in heat transfer may be so great as to counterbalance the decrease in air flow, but is not great enough in any case observed to overcome the disadvantage of the increased head resistance.

DESCRIPTION OF CURVES.

The accompanying curves show the properties of the six types of radiator, expressed as follows (all values have been reduced to an air density of 0.0750 pound per cubic foot):

Free air speed, in miles per hour.

Mass flow of air through the radiator, in pounds per second per square foot frontal area.

Energy dissipated, in horsepower per square foot per 100° F., difference between the temperature of the entering air and the mean of the temperatures of the entering and leaving water.

Head resistance, in pounds per square foot frontal area.

Horsepower absorbed, in horsepower per square foot frontal area.

Figure of merit is the ratio of the horsepower dissipated to the horsepower absorbed.

Plot 1, reproduced from Technical Report No. 63, Part I, shows the properties of the plain tube type, E–9. Its heat transfer is high, but its head resistance is also high, and the figure of merit is low. The great weight of the radiator accounts for its very low figure of merit at low speeds.

The relation between the mass flow of air through the radiators and free air speed is shown for three of the perforated tube types in plot 2. For ordinary radiators this relation is linear, and it is practically so for the plain tube whistling type. The irregular form of these curves shows clearly the effect of the peculiar conditions of air flow in the radiator, the sudden changes in slope of the curve corresponding to sudden changes in tone of the whistle.

Plot 3 shows energy dissipated (heat transfer) in terms of mass flow of air. The curve for the type E–4 was determined by interpolation,[2] since the water boxes had been removed in order to measure its head resistance. Too much importance should not be assigned to the fact that some of these curves show a high heat transfer with a given mass flow of air, for Plot 4 shows that the highest curve of Plot 3—that for E–3, four tubes deep, five-sixteenth inch pitch—becomes low when heat transfer is plotted against free air speed. About 70 radiators tested at this bureau have been graded from "A" for very high heat transfer for a given free air speed to "E" for very low heat transfer, and these perforated tube types are graded "D" on such a scale.

Plots 5 and 6, reproduced from Technical Report No. 63, Part I, show the complete properties of the type E–4, in terms of mass flow of air, and in terms of free air speed, respectively. The curves are reliable within the ranges covered, but the irregular relation between mass flow of air and free air speed makes extrapolations extremely doubtful.

Head resistance of ordinary radiators is nearly proportional to the square of free air speed, but the irregularities in the head resistance curves of Plots 1 and 6 show that this is not the case with the whistling types.

COMPARISON BETWEEN WHISTLING RADIATORS AND FLAT PLATE RADIATORS.

The foregoing statements should not be interpreted as applying in any degree to radiators whose water tubes are flat plates with smooth surfaces, and continuous from front to rear of the radiator.

The whistling types are characterized by low air flow and often low heat transfer (for a given free air speed), by high head resistance, and by a low figure of merit; but the flat plate types are characterized by high air flow and heat transfer (for a given free air speed), by low head resistance, and by a high figure of merit. The following comparative tables show the superiority of the flat plate types over the whistling types.

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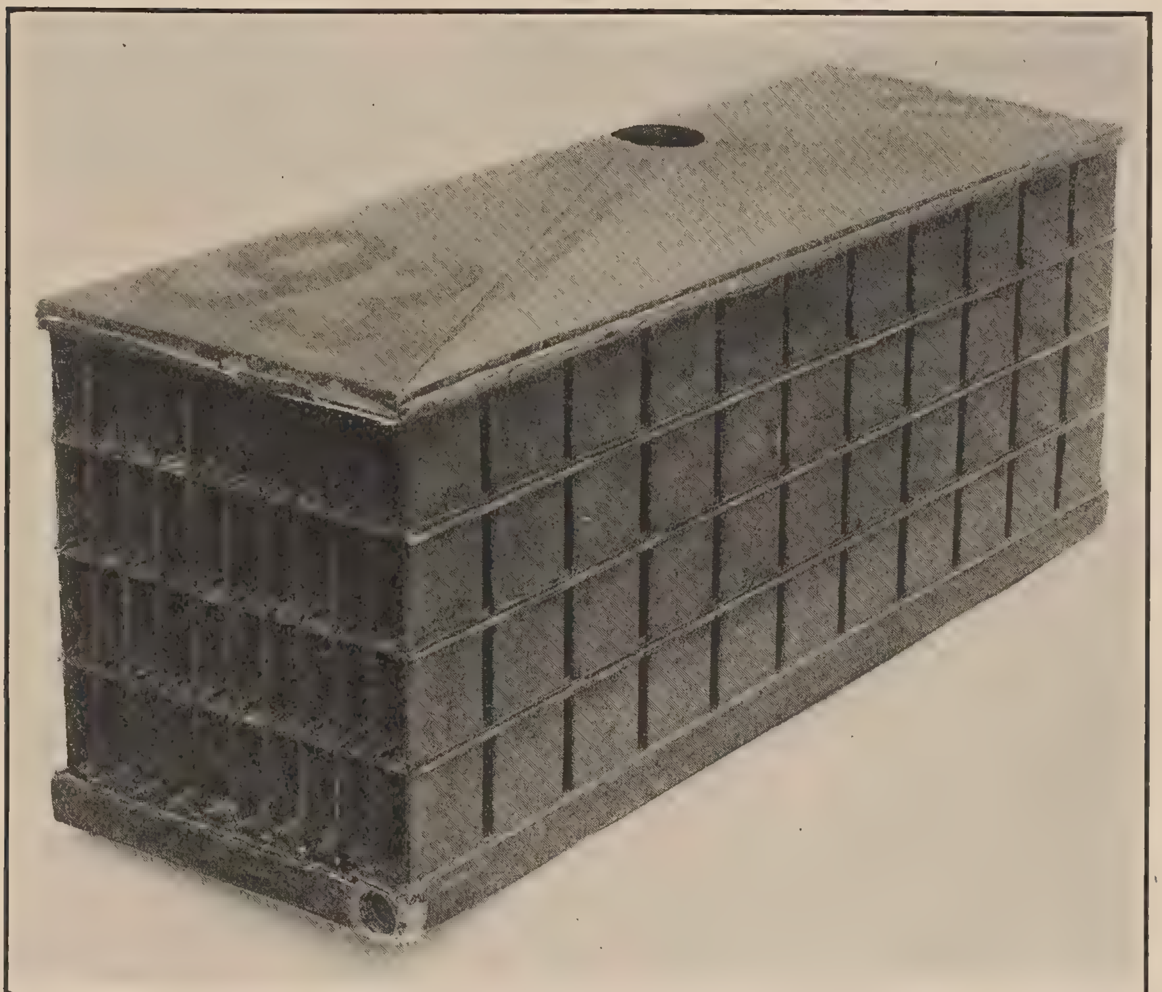
[2] Types E–3, E–4, and E–5 are each four tubes deep, and are five-sixteenths, one-half, and five-eighths inch pitch, respectively. The heat transfer for E–3 and E–5 was found to be proportional to the number of tubes per foot width of front, and this proportionality was used in interpolating for E–4.



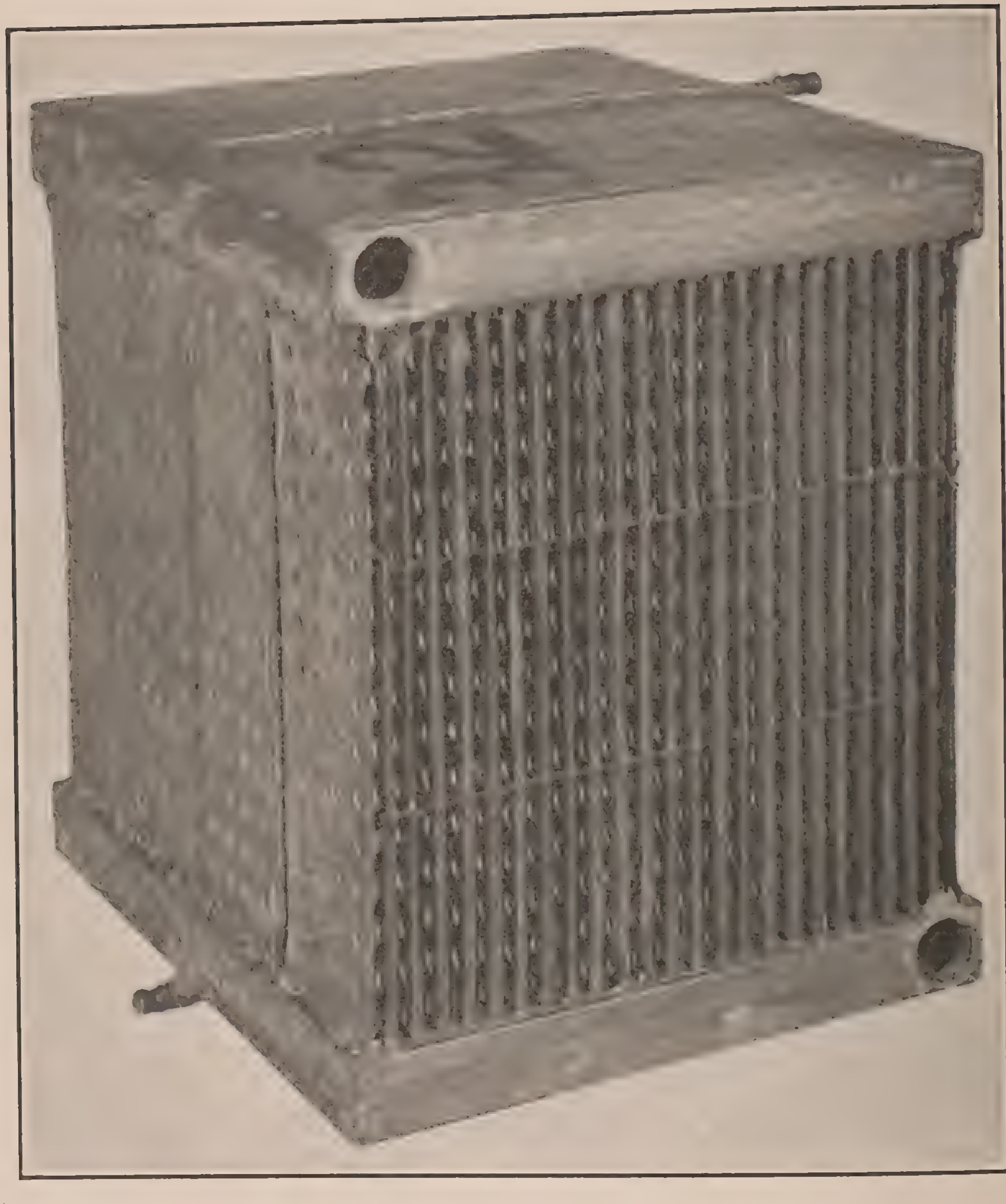


TABLE I.—*Perforated tube type and flat plate type of the same pitch and practically the same depth.*

Type.	E-4, perforated tube.	E-8, flat plate.	Type.	E-4, perforated tube.	E-8, flat plate.
Pitch, inches	0.5	0.5	Energy dissipated, horsepower per square foot per 100° F	30.8	29.8
Depth, inches	9.50	9.75	Head resistance, pounds per square foot	6.73	3.12
Cooling surface, square feet, per square foot front	29.5	39.2	Horsepower absorbed per square foot	1.75	.92
Mass flow of air, pounds per second per square foot	3.70	5.48	Figure of merit	17.6	32.4

Test data are not available for a direct comparison of plain tube and flat plate types of radiator of the same pitch, but the plain tube whistling type E–9 of three-fourths inch pitch may be compared with flat plate types of one-half inch pitch. It was previously shown that for flat plate types a pitch of one-half inch is better at high speeds than the closer spacings, and there seems to be no reason to suppose that a pitch of three-fourths inch would be less efficient. A comparison of the two types would appear to give an advantage to the one of three-fourths inch pitch. Two depths of the flat plate type are considered, giving approximately the same direct cooling surface (not including fins), and the same actual depth of tubes, respectively, as the whistling types. A plain tube whistling type H–3, described in a French report, is also included.

TABLE II.—*Comparison of flat plate and plain tube radiators.*

FREE AIR SPEED, 120 MILES PER HOUR.

Type.	Flat plate.	Flat plate.	E-9, plain tube.
Pitch, inches	0.5	0.5	0.75
Depth, inches	16.	20.	23.25
Direct cooling surface	61.9	80.5	61.7
Mass flow of air	10.33	9.91	8.96
Energy dissipated	77.4	90.5	81.5
Head resistance	17.1	20.8	25.9
Horsepower absorbed	6.7	8.2	11.2
Figure of merit	11.5	11.0	7.3

FREE AIR SPEED, 90 MILES PER HOUR.

Type.	Flat plate.	H-3, plain tube.	Flat plate.	E-9, plain tube.
Pitch, inches	0.5	0.57	0.5	0.75
Depth, inches	16.	15.8	20.	23.25
Figure of merit	18.9	12.5	18.0	10.8

CONCLUSIONS.

The above tables seem to show clearly that any increase in heat transfer caused by perforations in the water tubes or spaces between them in the direction of air flow is at the expense of a great increase in head resistance and is accompanied by a decrease in the figure of merit. The same result has been found in the case of turbulence vanes in cellular radiators, and indeed no type of radiator is known to this bureau in which an artificial increase of turbulence is not accompanied by a decrease in figure of merit. For use in obstructed positions, such as the nose of the fuselage, it may be necessary to sacrifice figure of merit of the radiator core for the benefit of heat transfer, but for unobstructed positions it appears that smooth straight air passages through the radiator should be provided.

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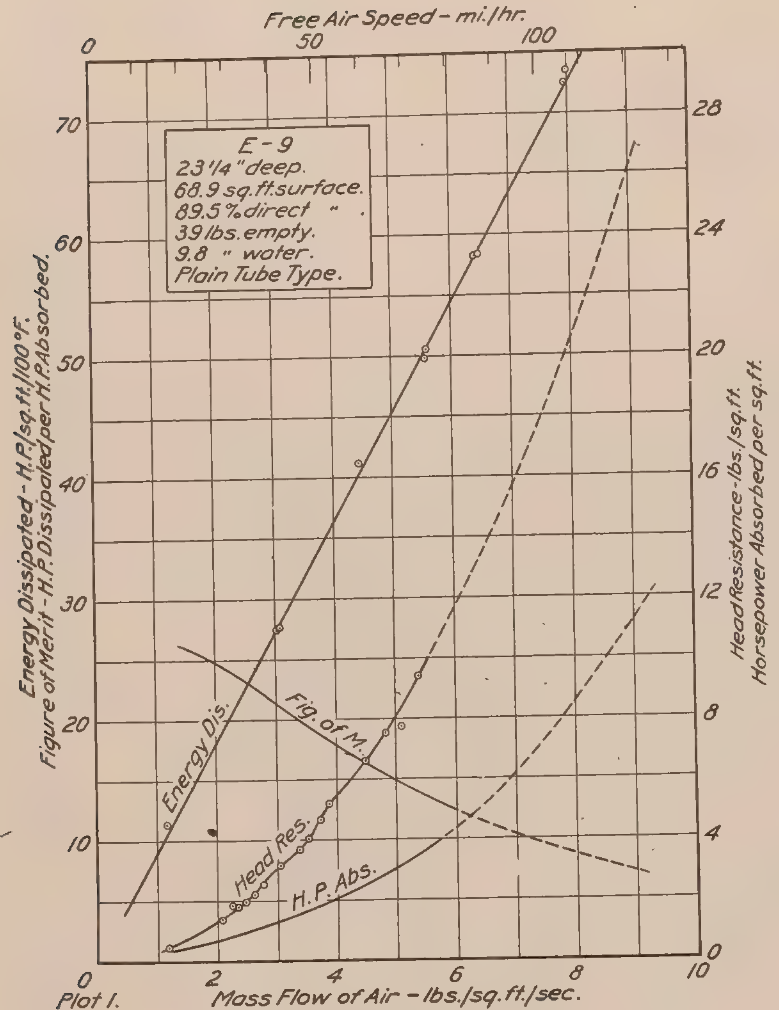
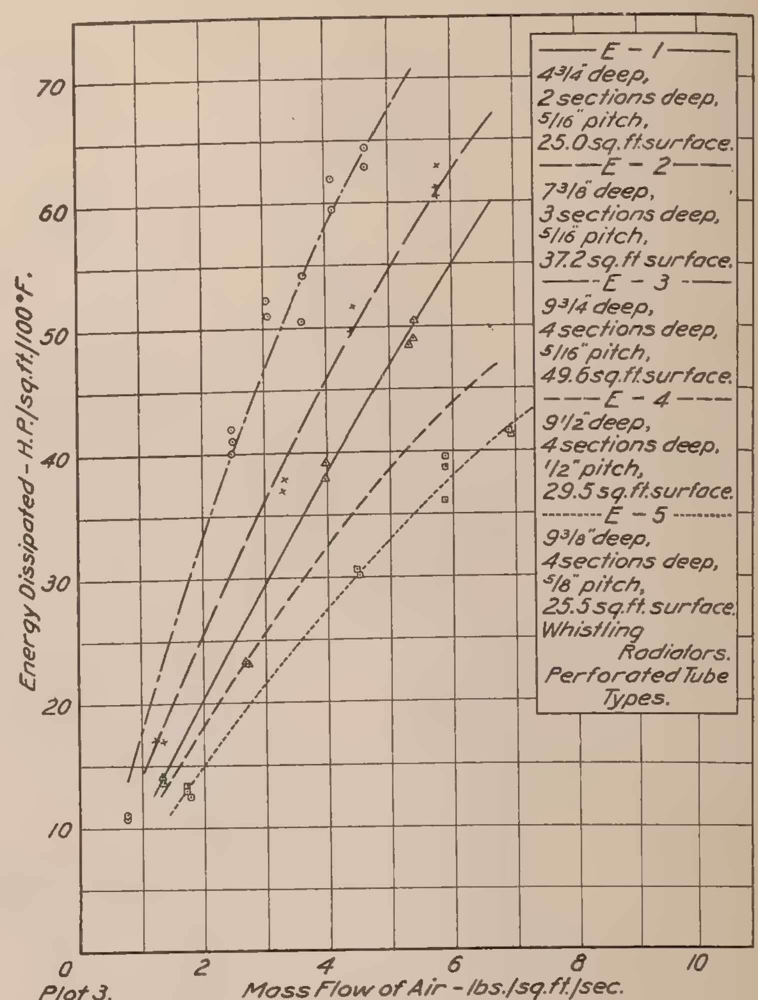


Plot 1.



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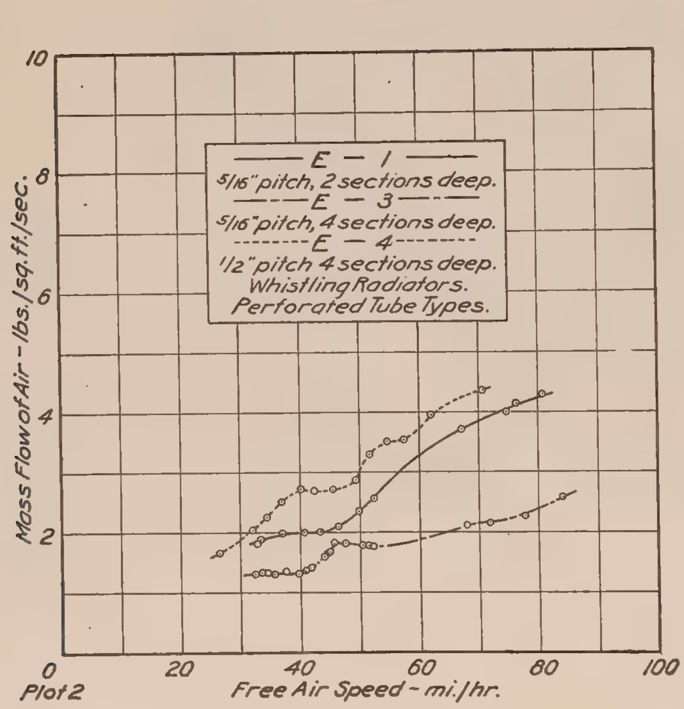
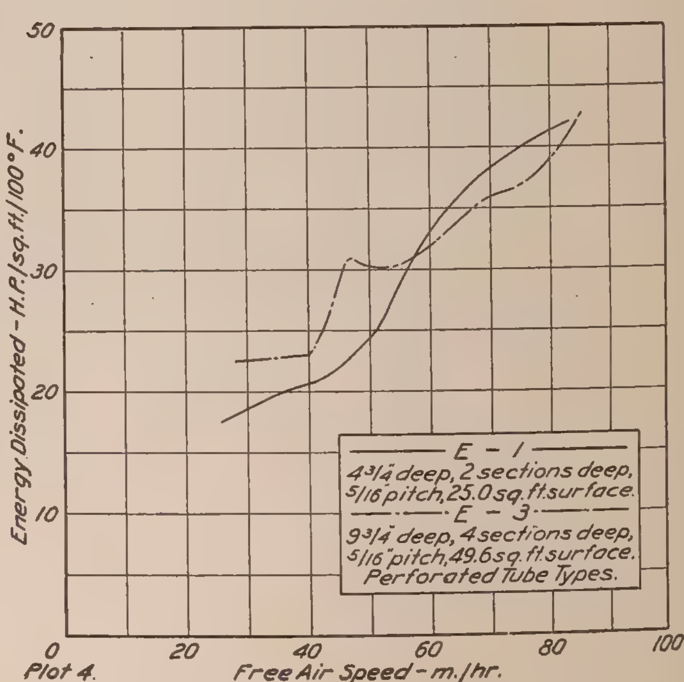
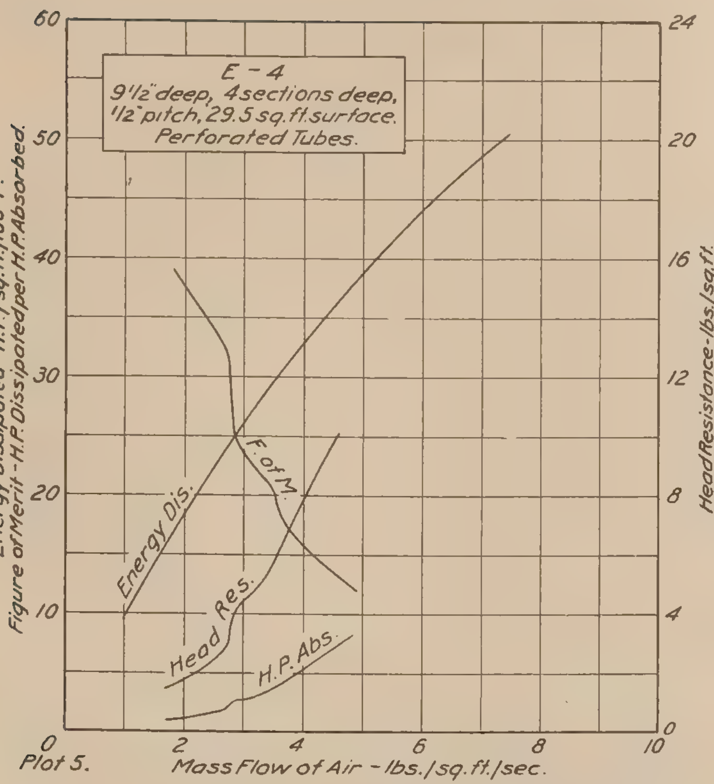
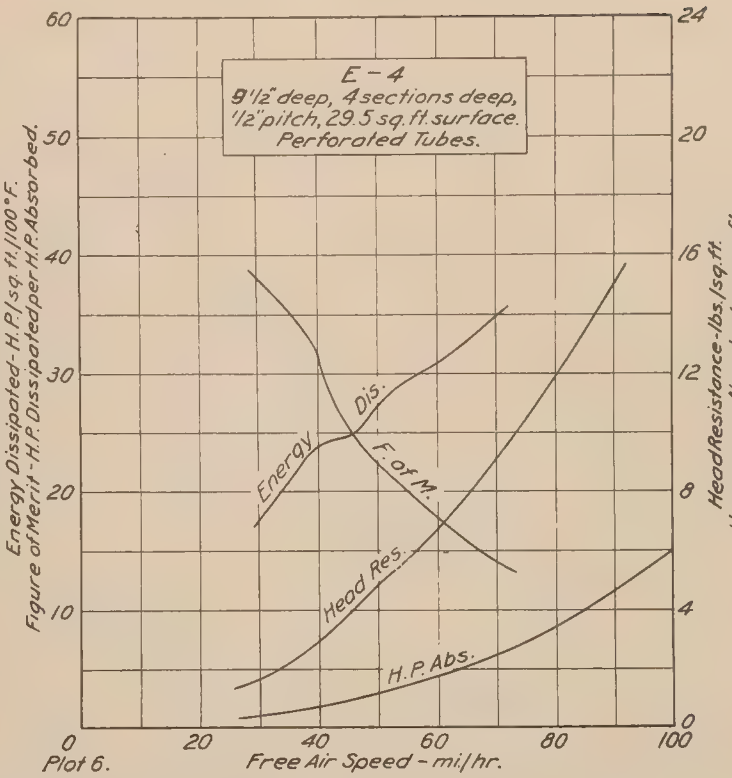


Plot 2



Plot 4.





EFFECTS OF YAWING AIRPLANE RADIATORS.

Wind-tunnel tests on radiators for aeronautic engines have usually been made with the face of the radiator normal to the direction of the wind, so that although local eddies might be set up, the general direction of the air stream was straight as it passed through the radiator. If the axes of the air passages are inclined at an angle with the general direction of the air stream; or, in other words, if the face of the radiator is not normal to the air stream, the properties of the radiator will be somewhat changed, and it is the purpose of this report to show the general form of these changes.

The effect of inclining the radiator at an angle with the air stream, or yawing the radiator, is of interest in connection with the following conditions:

(1) Radiator mounted in the propeller slip stream, where the air strikes the radiator at other angles than normal to its face.

(2) Radiator mounted in any position (such as the wing) where the axes of its passages for the air are not parallel to the direction of motion of the plane.

(3) Radiator pivoted about an axis perpendicular to the direction of motion of the airplane. This construction represents one method that has been suggested for the regulation of cooling capacity.

The effects of yawing a radiator through angles from 0° to 45° fall into three classes, namely, the effects on (1) air flow through the core, (2) heat transfer, and (3) head resistance.

(1) EFFECT ON AIR FLOW THROUGH THE CORE.

For ordinary types of radiator the mass flow of air through the core is directly proportional to the free air speed (that is, to the speed of the air stream in which the radiator is placed) when the face of the radiator is normal to the air stream. Measurements of air flow have been made on two sections in yawed positions and the results obtained with one of the sections are shown in Plot 1, which indicates that the relation between mass flow of air through the core and free air speed is still linear between 30 and 90 miles per hour, but the line points below the origin. The two types tested are of square cell construction, about one-fourth inch in diameter.

A similar effect, but much more pronounced, was observed in the case of a single type of core when in its normal (not yawed) position. The result of this test is also shown in Plot 1. This type, which is the type G–4, of Technical Report No. 63, Part I, is made up with irregular shaped cells inclined at an angle of 4½° to the normal to the face, alternate rows of cells being inclined on opposite sides of the normal. It is illustrated in the accompanying photograph.

The type G-4 a 'd types t at whistle in the air stream are the only radiators noted in which the air flow through the core is not directly proportional to the free air speed.

(2) EFFECT ON HEAT TRANSFER.

No tests have been made at this bureau to determine the effect of yawing the radiator on heat transfer, but a French report mentions a section (type not specified, but probably cellular) which showed an increase in heat transfer as the angle of yaw increased from 0° to 20° and then a decrease, bringing the heat transfer back to its normal value with an angle of about 40°. The maximum increase over the normal value was about 7 per cent. The French report further states that "with radiators of water-plate type the increase in heat transfer is not so clearly shown, but the decrease in effectiveness is hardly perceptible below 40°."

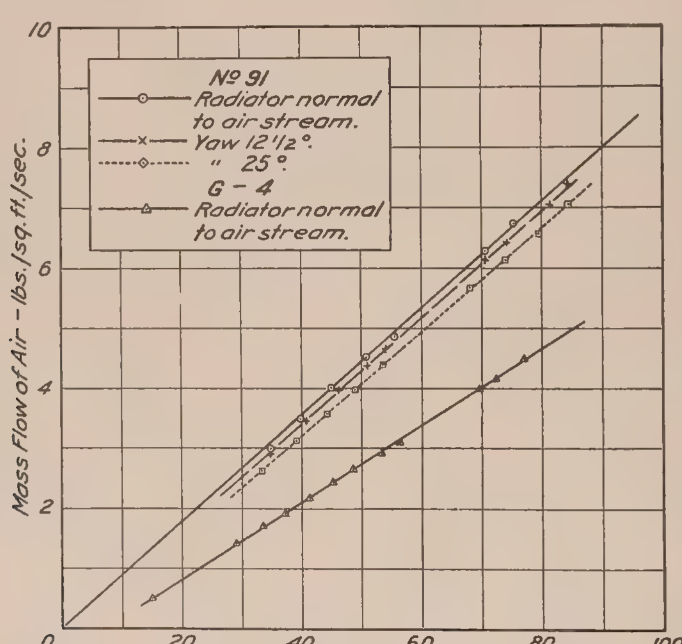
A British report states that three honeycomb types, each 12 inches square, showed a maximum increase in heat transfer of somewhat over 20 per cent with angles of yaw of about 45°. A large honeycomb section with 4.8 square feet frontal area gave a maximum of about the same per cent at 25°.

(3) EFFECT ON HEAD RESISTANCE.

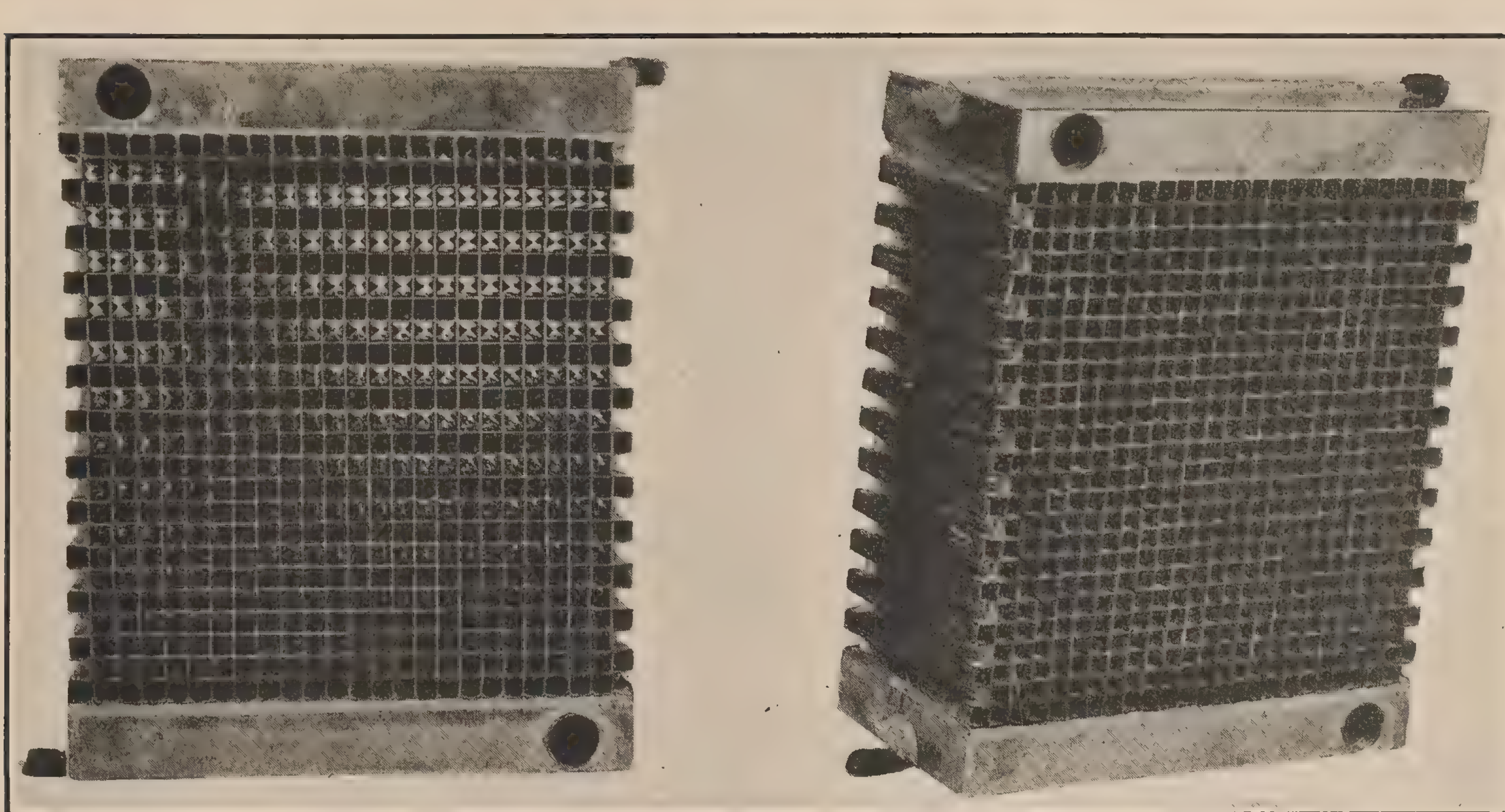
This effect has been treated in Technical Report No. 61, Part I, "Head resistance of radiator cores," and the accompanying curves are taken from that report.

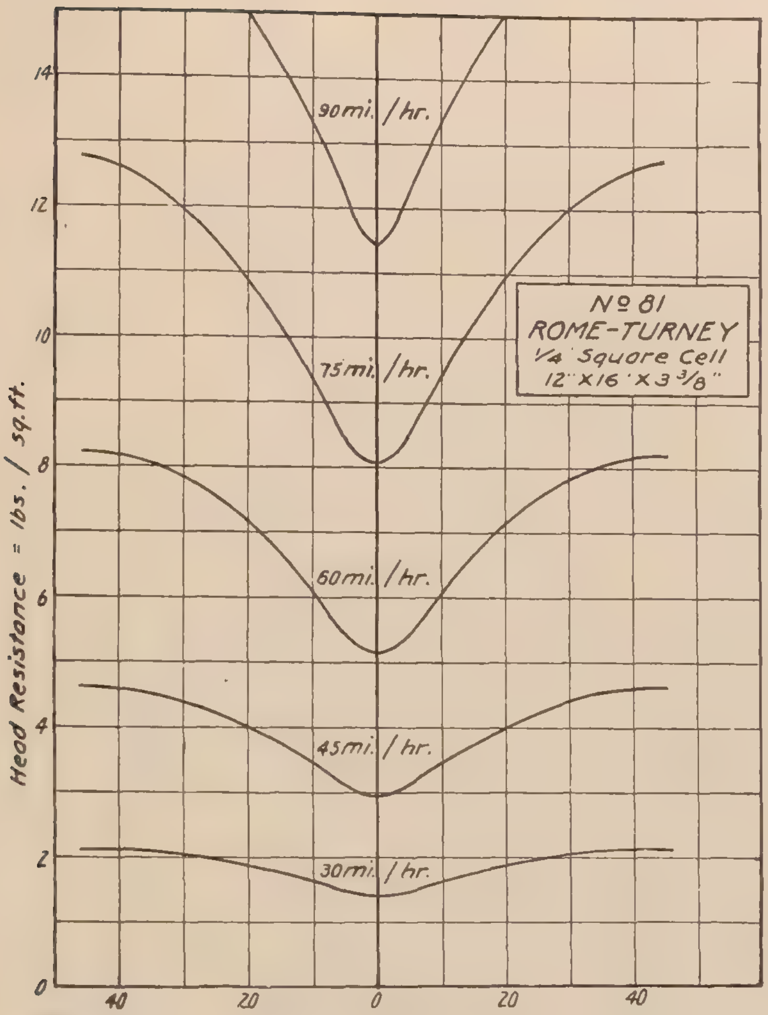
Plots 2 and 3, for one-fourth inch square cell (No. 81) and three-eighths-inch elliptical cell (No. 73), respectively, are given as typical of ordinary cellular types. The head resistance increases rapidly with the angle of yaw up to at least 30°.

Plots 4 and 5 are for special types which show the variations in the form of the curves. The type No. 80 has spiral air passages, and its head resistance shows very little change for angles up to about 30°. The type No. 79, shown in the photographs, is the type whose peculiar air flow is mentioned above. This shows a large increase in head resistance with increase in angle of yaw, and in particular it shows a very sudden increase with small angles.

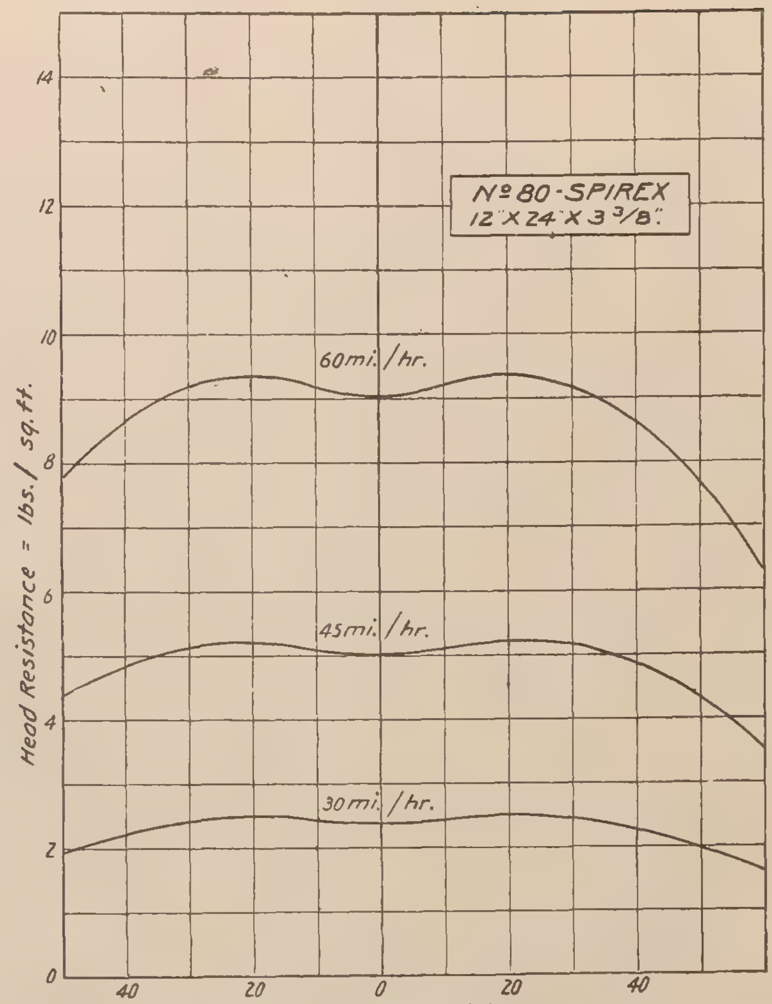


Plot 1.

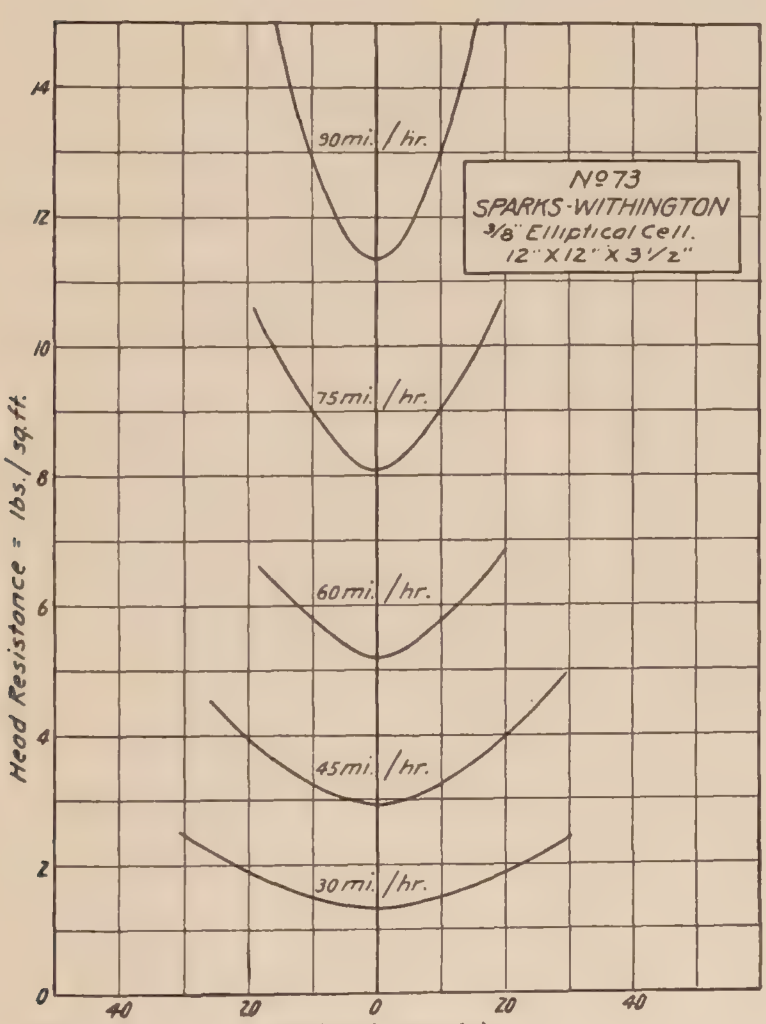




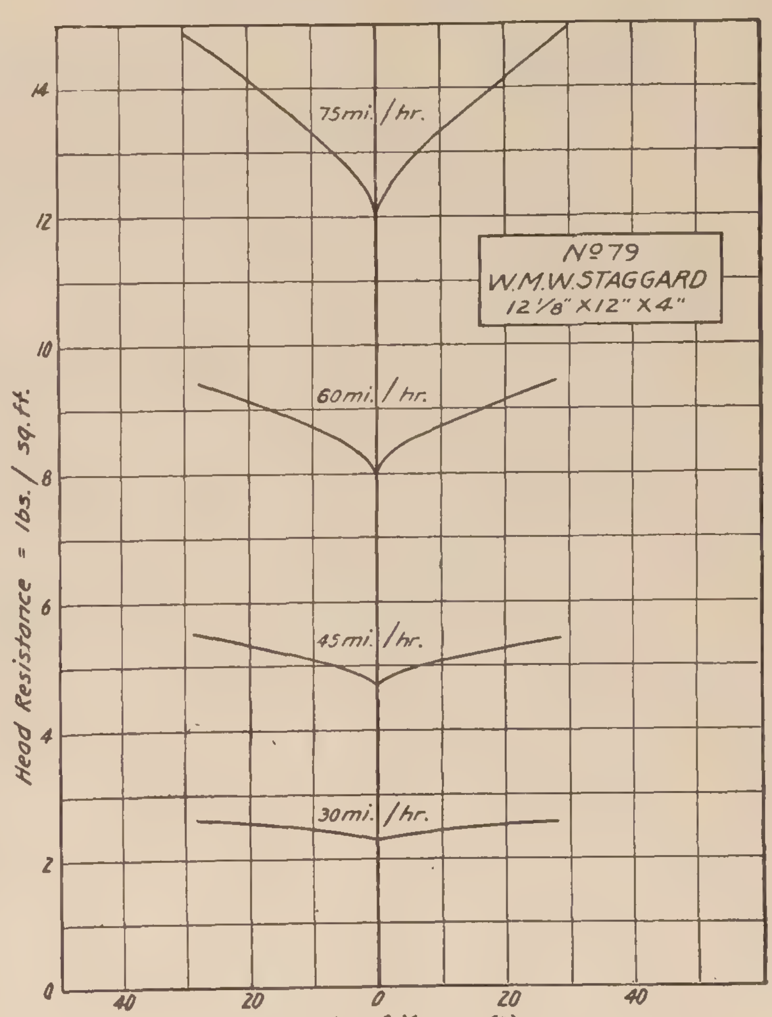
Plot 2.



Plot 4.



Plot 3.



Plot 5.

REPORT No. 87

EFFECTS OF NATURE OF COOLING SURFACE ON RADIATOR PERFORMANCE

BY

S. R. PARSONS and R. V. KLEINSCHMIDT
Bureau of Standards

REPORT No. 87.

EFFECTS OF NATURE OF COOLING SURFACE ON RADIATOR PERFORMANCE.

By S. R. Parsons and R. V. Kleinschmidt,
Bureau of Standards.

RÉSUMÉ.

This report describes an investigation of effects of nature of cooling surface on radiator performance, conducted for the National Advisory Committee for Aeronautics at the Bureau of Standards.

Cooling surfaces in radiators should be kept clean.

An accumulation of oil and dust on the surface will have a very harmful effect on the performance of the radiator.

The following remarks apply only to conditions in which the cooling surfaces are reasonably clean:

1. Heat transfer from an ordinarily smooth surface may be increased 17 per cent for a given air flow, by giving the surface a high polish; or it may be decreased 10 per cent or more by smoking the surface; but

2. Surfaces likely to be obtained in radiators, if fairly clean, will not differ in smoothness enough to give appreciable difference in heat transfer, with a given flow of air through the core.

3. Heat transfer from a radiator may be considerably decreased if the surfaces are not kept reasonably clean.

4. Heat transfer from a radiator (at a given airplane speed) may be slightly increased if special attention is given to smoothness of surface, on account of a small increase in air flow through the core.

5. Heat transfer is practically unaffected by a light coating of clean oil on a smooth surface.

6. Pressure gradient is practically independent of the roughness of the surface over a considerable range.

7. Pressure gradient is practically unaffected by a light coating of clean oil on a smooth surface.

8. Head resistance of a radiator may be slightly decreased by polishing the surfaces (8 per cent observed in one case).

9. Flow of air through the core of a radiator may be somewhat increased by polishing the surfaces (5 per cent observed in one case).

10. Figure of merit of a radiator may be somewhat increased by polishing the surfaces (6 to 10 per cent observed in one case).

11. In general, the performance of a radiator may be improved by polishing the surfaces; but if they are fairly smooth *and clean,* a considerable polish is required to produce much change in the properties of the radiator, and there is a question whether or not such a method for improvement be practicable.

Since the performance of an aircraft radiator depends upon its capacity for transfer of heat from cooling surfaces to moving air, and upon resistance offered to the passage of the air stream, it follows that the nature of the cooling surfaces is a factor worthy of consideration in connection with the properties of the radiator. For direct cooling surface, i. e., for surface backed by flowing water, the effect on heat transfer of the composition of the metal need not be considered, except as one metal is capable of taking a better surface than another, because almost

any metal will conduct heat through the thin walls of the water tubes as rapidly as it can be transferred from the surface of the tubes to the air. The composition and the thickness of the metal are of some importance in the case of surface not backed by flowing water, but it will not be considered here, as this report will deal only with the mechanical condition of the surface.

It will be shown that the degree of roughness or smoothness of the surface is capable of causing important effects on rate of heat transfer and on head resistance, but that in actual radiators having fairly clean surfaces, the differences between various degrees of smoothness are not sufficient to give the effects on radiator properties that might be obtained if it were practicable to have the surfaces highly polished. If, however, the cooling surfaces become coated with oil and dust, the decrease in rate of heat transfer may be very great.

The experimental work on which this report is based consisted of the following measurements, which are described in detail below:

I. Heat transfer from a single tube, with different conditions of surface.
II. Heat transfer from two radiators, each with rough and somewhat smoothed surfaces.
III. Pressure drop in a single tube with different conditions of surface.
IV. Pressure drop in two radiators of similar construction, but one with rough, and the other with somewhat smoothed, surfaces.
V. Head resistance of one radiator section, before and after the surfaces had been somewhat smoothed.
VI. Mass flow of air through the core of one radiator section, before and after the surfaces had been somewhat smoothed.

I. EFFECT OF SURFACE ON HEAT TRANSFER FROM A SINGLE TUBE.

The tube used was of brass, 41.5 cm. (16.3 inches) long, with an inside diameter of 7.8 mm. (0.31 inch), and with walls approximately 1 mm. (0.04 inch) thick. Eight thermocouples were soldered into shallow slots on the outside of the tube, at points 2, 7, 12, and 17 cm. from the ends, and heat was supplied electrically from a coil of No. 32 copper wire wound closely the entire length of the tube, and carefully insulated with baked shellac. The tube was wrapped in hair felt with a corrugated paper covering to within 1 cm. of each end; and the ends were wrapped with several layers of friction tape and inserted tightly into two wooden boxes or chambers in which the properties of the air could be measured as it entered and left the tube. These chambers were 7.6 cm. (3 inches) square and 15.2 cm. (6 inches) long, and each was divided into three compartments by screens. The air entering the first compartment of the inlet chamber passed through a series of screens of coarse mesh wire and finally through a thin screen of hair felt into the second compartment, where its temperature was measured. It then passed through another screen into a third compartment, into which the end of the tube projected about 1 mm. (0.04 inch). This compartment was connected to one side of a vertical oil gauge used to measure the pressure drop through the tube. On leaving the tube, the air passed through the first compartment of the exit chamber, which was connected to the other side of the oil gauge, and then through a screen of wire and a layer of hair felt into the thermometer section, and finally through another screen into the last compartment, which was connected to the inlet of the fan. The exit chamber was very carefully lagged with 2.5 cm. of cork on the outside and 0.5 cm. of hair felt on the inside to prevent the turbulent air from striking the wooden walls directly.

The mass of air flowing through the tube was computed from the heat input to the coil, the rise in temperature of the air and the specific heat of the air—that is, by using the tube itself as a Thomas meter—with the exception that pressure drop through the tube was used for some of the runs after it had been calibrated against the tube as a Thomas meter. The pressure drop method was used in some of the earlier runs when mercury thermometers were used to measure the temperature rise, but was abandoned when thermocouples were used for this measurement. The thermocouples were read on a "pyrovolter," and the heat input to the coil was obtained from readings of a voltmeter and an ammeter.

Care was taken to obtain steady temperature conditions before beginning any set of readings.

Five conditions of surface were used, viz:

1. Original surface (as the tube was drawn).
2. Surface polished (considerable time and effort were expended in getting a high degree of polish).
3. Polished surface lightly oiled.
4. Polished surface lightly smoked.
5. Surface roughened with fine sandpaper.

The results are shown in Plot 1 and in the following table, which shows heat transfer in watts per degree centigrade of difference between the mean temperature of the tube and the temperature of the entering air, and per cent of increase or decrease of heat transfer over that for the original surface:

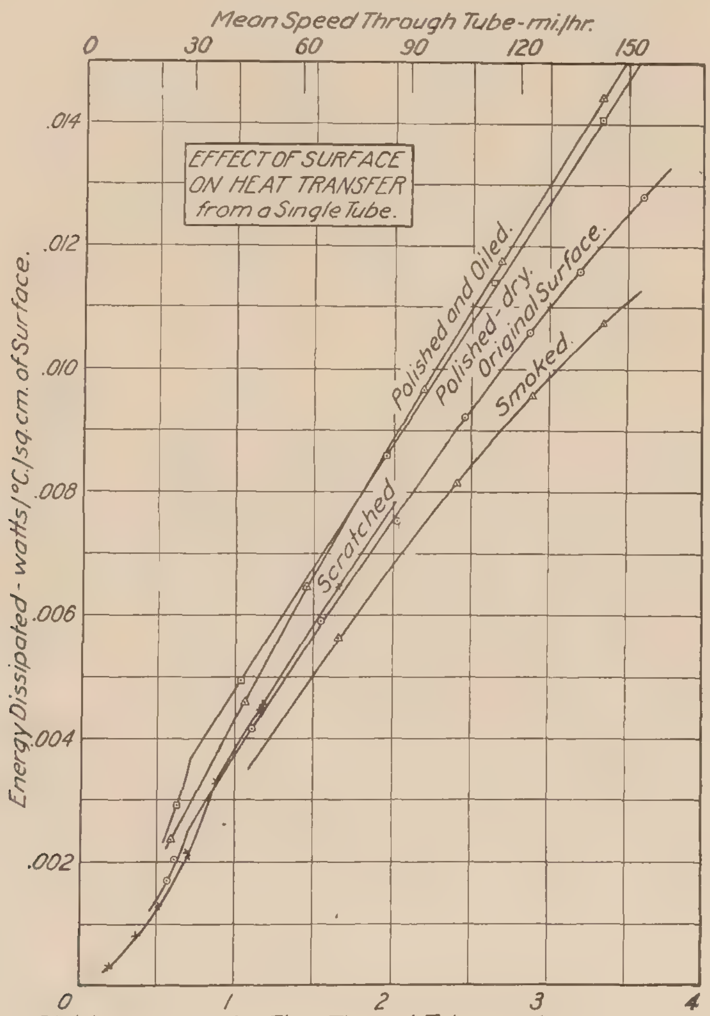
Heat transfer.

[In watts per °C.]

Air flow gm./sec.	Original.	Polished.	Oiled.	Smoked.	Rough.
1	0.378	0.487+28.8%	0.447+18.2%	0.335−11.4%	0.390+3.2%
2	.752	.882+17.3%	.892+18.6%	.686− 8.8%	.780+3.7%
3	1.10	1.28 +16.7%	1.30 +18.6%	.988−10.2%	1.14 +3.6%

The table shows the following points:

1. The highly polished surface dissipated about 17 per cent more heat (at the higher speeds) than one fairly rough.
2. The smoked surface dissipated about 10 per cent less heat than the fairly rough one.
3. Oiling the polished surface had very little effect on its heat transfer.
4. The roughened surface was not much different in its heat transfer from the original surface.



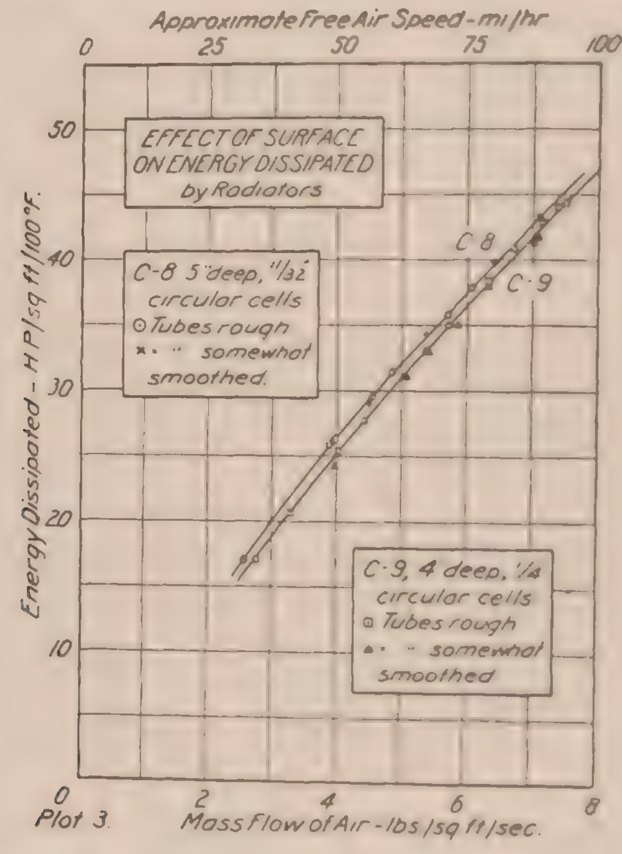
Plot 1.

The considerable difference in heat transfer between the fairly rough and the highly polished surfaces may be accounted for by the fact that roughness allows a blanket of more or less stagnant

air to cling to the surface, and thereby to some extent prevents the scouring of the surface that is required for the most rapid transfer of heat.

II. EFFECT OF SURFACE ON HEAT TRANSFER FROM A RADIATOR.

The usual test of heat transfer in terms of air flow was made on two radiator sections 4 and 5 inches (10.2 and 12.7 cm.) deep, and with $\frac{1}{4}$-inch and $\frac{11}{32}$-inch (0.64 cm. and 0.87 cm.) circular cells, respectively, each before and after the cooling surfaces had been somewhat smoothed. The original surfaces were considerably rougher than those of many well-made radiators, and the smoothed surfaces were somewhat better than those usually found in radiators, but did not even approximate to the high polish obtained in the single tube mentioned above. The curves, shown in Plot 3, indicate no difference in heat transfer greater than the limit of experimental error, and it appears that although it is possible to increase the heat transfer considerably by giving the surfaces a high polish, it is nevertheless true that any surface likely to be obtained in commercial production will not have a sufficiently high polish to take advantage of this fact.



Plot 3.

III. EFFECT OF SURFACE ON PRESSURE GRADIENT IN A SINGLE TUBE.

A brass tube 105 cm. (41.3 inches) in length and with an inside diameter of 0.95 cm. ($\frac{3}{8}$ inch) was used for the measurement of pressure drop with different conditions of surface. Small holes were drilled in the tube at 10-cm. intervals beginning 5 cm. from each end, and tubes were attached to read static pressure at each of these 11 positions. Since it was necessary to remove the burr from the inside of the tube after the holes were drilled, the original surface was not used, and the first measurements were made with the tube polished, though not to the same degree as that obtained in the tube used for heat transfer.

The air flow was measured by means of a small Thomas meter made for that purpose. The meter was made with considerable care, and while it was not calibrated, because of the lack of convenient apparatus, it was without doubt good for comparative purposes at least.

Pressure gradients were obtained by plotting the pressures read at the 11 static holes against their respective positions, and were expressed in grams per square centimeter per centimeter length of tube. The observations on the smoothed and oiled surfaces were very consistent, but when the tube was smoked or roughened the observations were less consistent, probably because of effects of the smoking and the roughening on the static-pressure openings. The errors due to irregularities, however, do not exceed 2 per cent. Corrections for the effect of changing density of the air ranged around 1 per cent and were omitted.

The pressure gradient for a given surface is very nearly proportional to the square of the air flow, and for purposes of comparison a constant "k" was computed for the equation

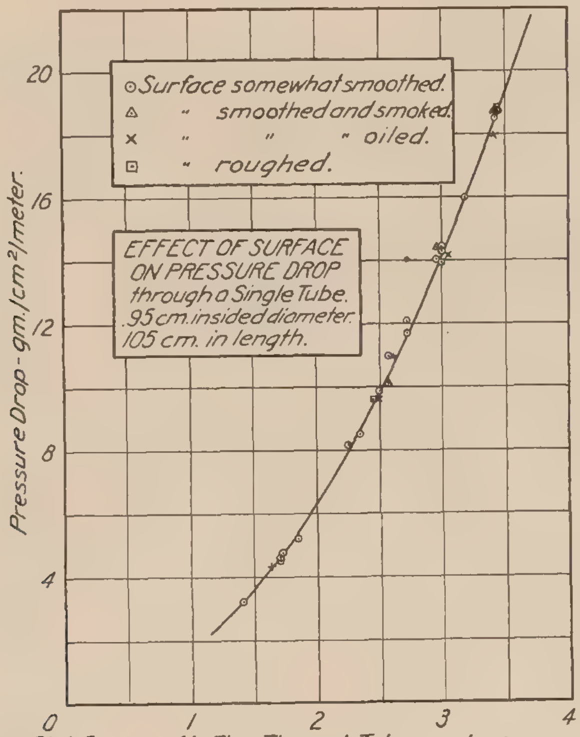
$$P = kM^2$$

where P = pressure gradient in gm. per sq. cm. per cm., and
M = air flow in gm. per sec.

The mean value of this constant for each surface is tabulated below, together with the per cent of difference from the constant of the smoothed surface. These differences are within the range of experimental error, with the possible exception of the smoked surface, which would probably have shown a greater difference if the tube had been more thoroughly smoked. The length and small diameter of the tube made the roughening of the surface somewhat inconvenient, and the pressure drop would without doubt have been considerably increased if the surface had been made considerably more rough, as is indicated by the results (described below) of the work on a tube in a radiator. These constants may be interpreted to mean that no noticeable difference in pressure drop is to be expected between different surfaces that vary between fairly wide limits of smoothness. The corresponding values of pressure gradient and air flow are shown in Plot 2.

"k" in equation $P = kM^2$.

Surface.	k	Difference, per cent.
Smoothed	0.0159	....
Smoothed and oiled	.0156	−1.9
Smoothed and smoked	.0163	+2.5
Roughened	.0158	−0.6



Plot 2.

IV. EFFECT OF SURFACE ON PRESSURE GRADIENT IN A TUBE OF A RADIATOR.

Two radiators were used, each 4 inches (10.2 cm.) deep, and with ¼-inch (0.64 cm.) circular cells. The first, with tubes somewhat polished, was one of those mentioned under "Heat transfer from a radiator." The second had very rough tubes, similar to those of the first before they had been smoothed.

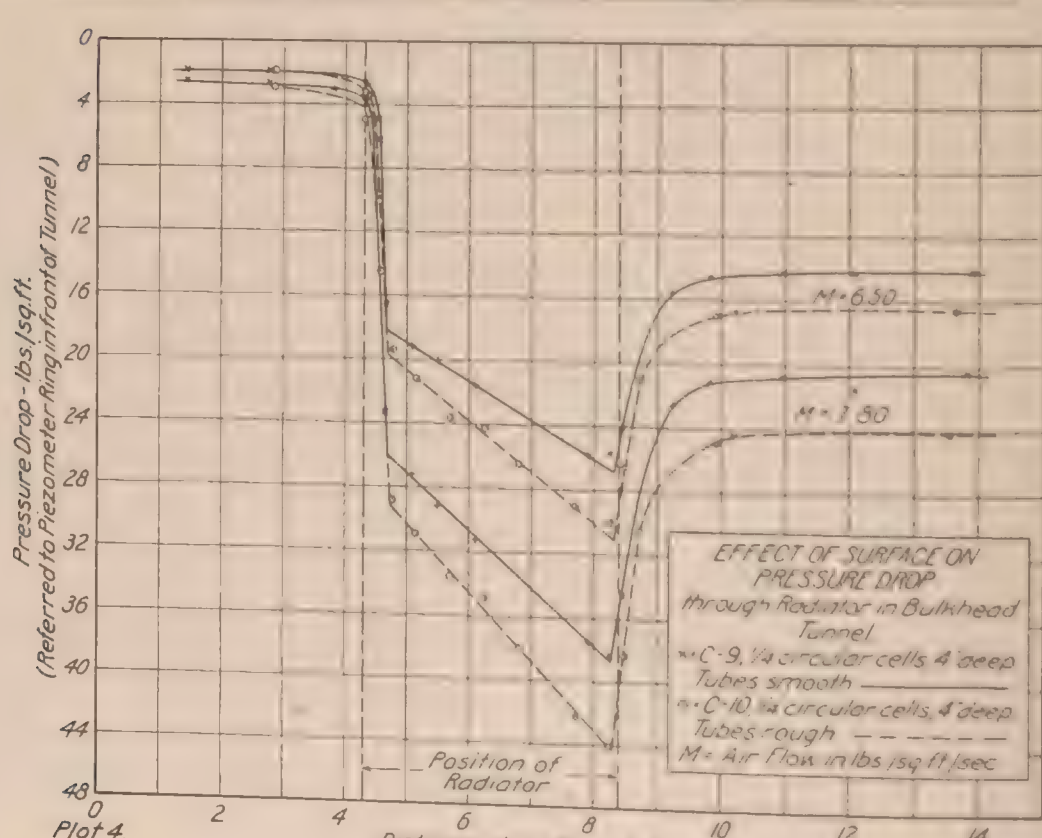
The radiator was placed in the 8 by 8 inch (20.3 cm.) wind tunnel used for measurement of heat transfer, and pressure was measured by means of a steel tube 0.04 inch (1 mm.) in outside diameter and 20 inches (51 cm.) long, with a static-pressure opening near the center. This static-pressure exploring tube was passed through an air tube near the center of the radiator,

and moved forward or backward to obtain the pressure at different positions. It was supported by two pieces of piano wire, which were attached to the ends, stretched over crossbars set in the tunnel 2 or 3 feet in front of and behind the radiator, and carried outside of the tunnel, to facilitate moving the tube forward and backward. One side of an inclined water gage was connected to the rear end of the exploring tube, and the other was connected to a piezometer ring in front of the radiator. It was found by trial that consistent results could be obtained if only ordinary care was used in centering the exploring tube inside of the air tube of the radiator.

Pressure drop between the piezometer ring and the exploring tube was expressed in pounds per square foot; and the air flow, in pounds per second per square foot of frontal area of radiator core. Previous work in a wind tunnel under partial vacuum has shown that the pressure drop between piezometer rings before and behind the radiator is inversely proportional to the air density at the front ring (for a given air flow), and this relation was used to correct for variations in density during the time of the observations.

Curves for two rates of air flow are shown in Plot 4, and the following table shows the values of the pressure gradients inside the radiator tube, both in pounds per square foot per inch, and in grams per square centimeter per centimeter, together with the per cent by which the gradient in the rough tube exceeds that in the smooth tube. The per cent of difference indicated is somewhat too high, because the two radiators were not quite identical, the one with rough tubes having a free area about 3 per cent less, and a head resistance about 7.5 per cent higher, than the other had before its tubes were smoothed.

Approximate free air speed, mi./hr.	Air flow, lb./sec. per sq. ft.	Pressure gradient.				Increase, per cent.
		gm./sq. cm. per cm.		lb./sq. ft. per inch.		
		C–9 smoothed.	C–10 rough.	C–9 smoothed.	C–10 rough.	
82	6.50	0.449	0.585	2.34	3.05	30
88	7.00	.512	.664	2.67	3.46	30
99	7.80	.641	.827	3.34	4.31	29

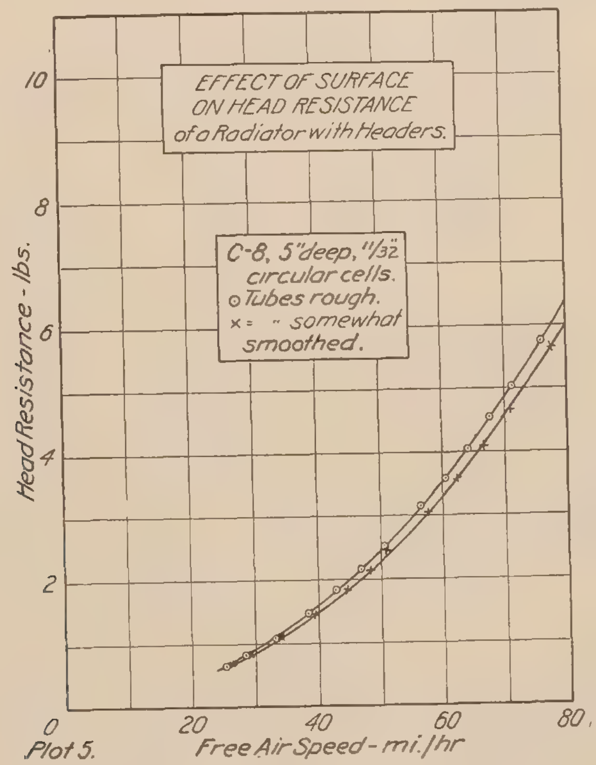


Plot 4.

V. EFFECT OF SURFACE ON HEAD RESISTANCE OF A RADIATOR.

The radiator C–8, 5 inches (12.7 cm.) deep, with $\frac{11}{32}$-inch (0.87 cm.) circular cells, was also tested for head resistance with the two conditions of surface described under "Heat transfer from a radiator," viz, very rough, and somewhat smoothed. The difference in head resistance

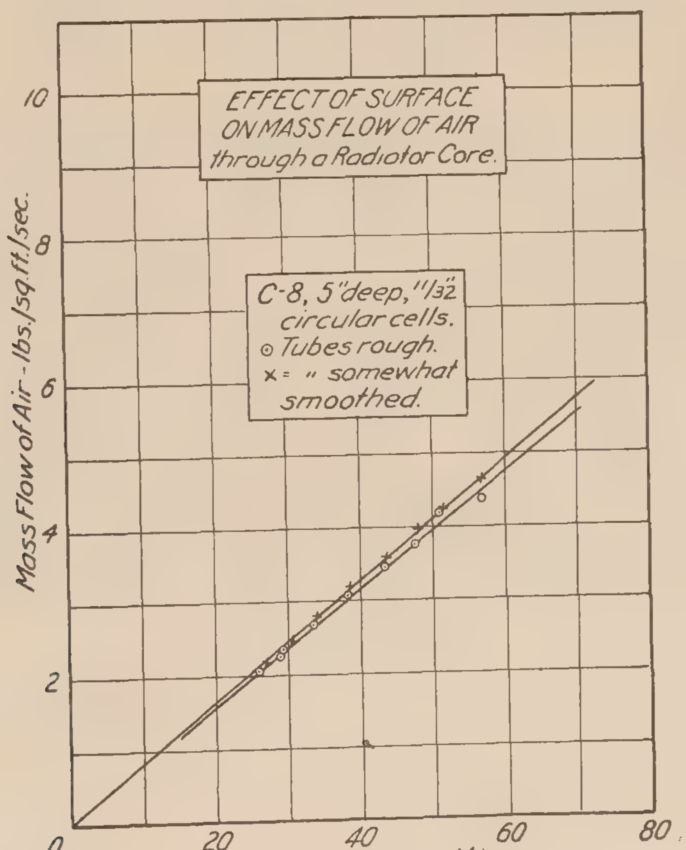
observed—about 6 per cent—is not as great as should be shown, because of the fact that the section used included attached water boxes, and the effect of the water boxes is partially to mask any effect of changes in the core. A good estimate of the change in head resistance may be made by comparing the air flow with Plot 1 of Technical Report 63, Part I, which shows a general relation between head resistance and air flow for radiators with straight air passages. The "mass flow factors" corresponding to the smoothed and rough surfaces were respectively 0.0820 and 0.0790, and the curve shows that the corresponding "head resistance constants" are 0.00153 and 0.00165, which gives a difference of 8 per cent, and is probably correct within 1 per cent. The observed values of head resistance are shown in Plot 5.



Plot 5.

VI. EFFECT OF SURFACE ON AIR FLOW THROUGH A RADIATOR.

As implied above, the radiator C-8 was tested not only for head resistance but for mass flow of air through the core, in terms of free air speed. The increase in air flow with the smoothed surface was 5 per cent, and is indicated in Plot 6.



Plot 6

EFFECT OF SURFACE ON FIGURE OF MERIT.

The tests made on the section C-8, 5 inches (12.7 cm.) deep with $\frac{11}{32}$-inch (0.87 cm.) circular cells, permit a computation of figure of merit with the two conditions of surface. The figure of merit is the ratio of the rate at which the radiator dissipates heat (expressed in horsepower) under specified conditions of temperature and water flow, to the horsepower absorbed by the radiator because of its head resistance and weight. The following results apply to the radiator when in such a position on the airplane that the flow of air through and around the radiator is not affected by other parts of the plane. The per cents of difference are based on the values with the tubes smoothed.

Free air speed, mi./hr.	Figure of merit.		Difference, per cent.
	Rough.	Smooth.	
30	39.2	41.8	6.2
60	19.5	21.4	8.9
90	10.3	11.5	10.3
120	6.0	6.7	10.3

EFFECT ON RADIATOR PERFORMANCE OF OIL AND DUST ON THE COOLING SURFACES.

The tests described above have dealt mainly with the degree of smoothness of the cooling surfaces, and so far as they go, they seem to indicate that in actual radiators the differences between conditions of surface encountered will usually not be great enough to show any very great difference in the properties of the radiators. But the results of these tests should be interpreted with a little caution, for they do not include the condition of surface caused by a coating of oil and dust, such as sometimes occurs in actual radiators. The "oiled" surfaces mentioned above were first polished, and then *lightly* coated with clean oil, and such a surface is evidently not representative of the of the heavy coat of oil and dust that sometimes accumulates. The smoked surface used with the single tube for heat transfer probably gives the nearest approach in the tests, to the condition of surface with oil and dust. Even though the tube was highly polished before being smoked, the lightly smoked surface caused an insulating blanket of smoke particles and nearly stagnant air that was sufficient to reduce the heat transfer to 10 per cent less than that with an ordinary surface; and a coating of oil filled with dust may be expected to cause an insulating blanket that will reduce the heat transfer even more. In fact, it is well known that even in automobiles such a surface interferes with the performance of the radiator.

CONCLUSIONS.

Cooling surfaces in radiators should be kept clean. An accumulation of oil and dust on the surface will have a very harmful effect on the performance of the radiator. The following remarks apply only to conditions in which the surface is reasonably clean.

The lack of any quantitative measure of the condition of the surface complicates the problem of correlating the various results, but in Plot 7 an attempt is made to show the relations between the different quantities, by indicating the per cent of difference between values of heat transfer, pressure drop, etc., corresponding to different conditions of surface. The head and the tail of each arrow indicate the conditions of surface considered, and the arrow points away from the quantity on which the percentage is based. For example, the arrow under "head resistance" indicates that in passing from the smoothed to the very rough surface, head resistance was increased by 8 per cent of its value with the smoothed surface.

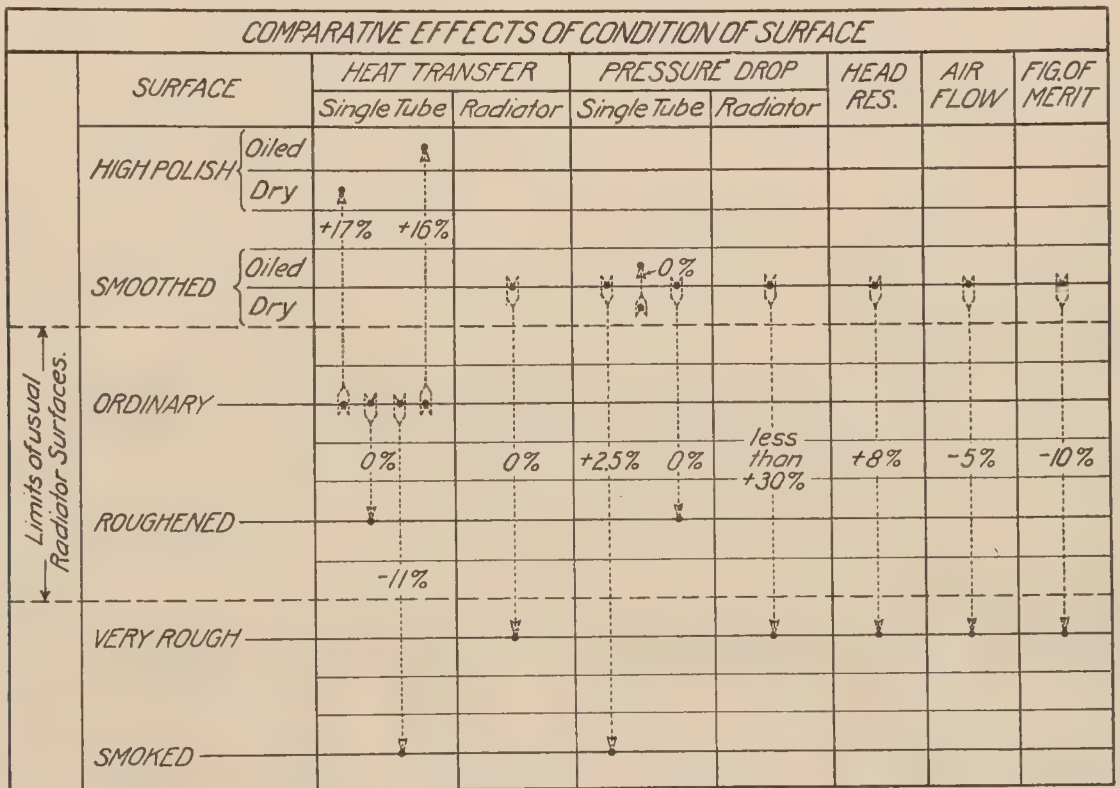
The results of the tests may be summarized as follows:

1. The degrees of smoothness usually found in radiators (not including the surface coated with oil and dust) are entirely within the range of the degrees of smoothness covered by most of the tests, so that with a few exceptions the per cents of difference shown in the diagram and in the tables are greater than what would usually be obtained by comparison of different radiators as they come from the manufacturers.

2. Heat transfer from an ordinarily smooth surface may be increased 17 per cent for a given air flow by giving the surface a high polish; or it may be decreased 10 per cent or more by smoking the surface; but

3. Surfaces likely to be obtained in radiators, if fairly clean, will not differ in smoothness enough to give appreciable difference in heat transfer, with a given flow of air through the core.

4. Heat transfer from a radiator may be considerably decreased if the surfaces are not kept reasonably clean.



Example:- Head resistance with very rough surface is 8 % greater than with smoothed surface.
Note:- This chart applies only to surfaces that are reasonably clean. Plot 7.

5. Heat transfer from a radiator (at a given airplane speed) may be slightly increased if special attention is given to smoothness of surface, on account of a small increase in air flow through the core.

6. Heat transfer is practically unaffected by a light coating of clean oil on a smooth surface.

7. Pressure gradient is practically independent of the roughness of surface over a considerable range.

8. Pressure gradient is practically unaffected by a light coating of clean oil on a smooth surface.

9. Head resistance of a radiator may be somewhat increased by polishing the surfaces (8 per cent observed in one case).

10. Flow of air through the core of a radiator may be somewhat increased by polishing the surfaces (5 per cent observed in one case).

11. Figure of merit of a radiator may be somewhat increased by polishing the surfaces (6 to 10 per cent observed in one case).

12. In general, the performance of a radiator may be improved by polishing the surfaces; but if they are fairly smooth and clean, a considerable polish is required to produce much change in the properties of the radiator, and there is a question whether or not such a method for improvement is practicable.

REPORT No. 88

PRESSURE DROP IN RADIATOR AIR TUBES

BY

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Bureau of Standards

REPORT No. 88.

PRESSURE DROP IN RADIATOR AIR TUBES.

By S. R. Parsons.

Bureau of Standards.

RÉSUMÉ.

This report describes an investigation of effects of pressure drop in radiator air tubes, conducted for the National Advisory Committee for Aeronautics at the Bureau of Standards.

A small steel tube—0.04 inch (1 mm.) in outside diameter and 20 inches (51 cm.) long, with a static pressure opening near the center—was stretched through an air tube of a radiator and used to measure static pressure in the stream of air passing through the radiator tube. The measurements lead to the following conclusions:

1. The drop in static pressure in the air stream through a cellular radiator, and the pressure gradient in the air tubes, are practically proportional to the square of the air flow, for a given air density. The observed values of skin friction agree approximately with those found by other investigators for long pipes. These facts appear to indicate that the air flow is turbulent, even in the short tubes of the radiators.

2. The difference between head resistance per unit area and fall of static pressure through the air tubes of a radiator, noted by various observers, is shown to be apparent rather than real.

3. Radiators of different types differ widely in the amount of contraction of the jet at entrance.

4. Frictional resistance is found to be two-thirds of head resistance for one type of $\frac{11}{32}$ inch (0.87 cm.) circular cells, 5 inches (12.7 cm.) deep; and one-half of head resistance for one type of $\frac{5}{16}$-inch (0.79 cm.) square cells, 4.8 inches (12.2 cm.) deep.

5. Supplying heat to the radiator increased the pressure gradient in the tubes of one type, of $\frac{1}{4}$-inch (0.64 cm.) circular cells, 4 inches (10.2 cm.) deep, by about 15 per cent for a mean temperature difference of 110° F. (61° C.) between the water and the entering air.

INTRODUCTION.

It has been noted in the course of investigations of aircraft radiators that the drop in static pressure in the air stream between the front and rear faces of the radiator seems not to be equal to the head resistance per unit area of the section, as measured on an aerodynamic balance. In some of the earlier investigations, both in this country and abroad, an attempt was made to measure this pressure drop, and the results obtained were assumed to be equal to the head resistance per unit area of the radiator for the same air flow. But as soon as aerodynamic balances became available for the work, and actual head resistance was measured, a considerable difference was found between observed head resistance and head resistance computed from pressure drop, and no satisfactory explanation of this difference was at once apparent. Since the air emerges from the radiator in a turbulent condition, its static pressure must be measured under unfavorable conditions, and it was natural to question the reliability of the measurements taken. An attempt to measure dynamic pressure before and behind the radiator was made, but with no better success.

The investigation described in this report was accordingly undertaken, in order to make independent, and if possible more reliable, measurements of static pressure at various points in the air stream; and to throw light on the difference, if any exists, between pressure drop and head resistance per unit area.

EXPERIMENTAL METHOD.

1. *Ordinary measurements of pressure drop.*

The bulkhead tunnels used for calorimetric tests of the radiators were fitted with piezometer rings before and behind the test section.[1] In the tunnel inclosed in the steel tank these rings were 1¼ inches (3.2 cm.) from the faces of the radiator and in the "steam tunnel" there were two pairs of rings, 1¼ and 6 inches (3.2 and 15.2 cm), respectively, from the faces of the section. Readings from the 1¼ inch and the 6 inch rings were practically identical for a number of sections.

2. *Special measurements of pressure drop.*

The special measurements of pressure drop indicated in the attached curves were made in two wind tunnels: The "steam tunnel" or closed tunnel, in which the radiator core completely fills the channel; and the open tunnel, which is 54 inches in diameter, and represents conditions in free air. The measurements were obtained with the use of a small steel tube, 0.04 inch (1 mm.) in outside diameter and 20 inches (51 cm.) long, with one end closed, and a static pressure opening near the center. This tube will be referred to below as "the pressure tube." It was passed through an air cell near the center of the radiator and moved forward or backward to obtain the pressure at different points. One side of an inclined water gauge was connected to the rear end of the pressure tube and the other side to the static pressure tube of the Pitot used to measure the velocity of the air stream in the channel.

The pressure tube was supported by two pieces of piano wire which were attached to the ends, passed over crossbars set in the closed tunnel, and rings held by wires in the open tunnel—at some distance before and behind the radiator—and passed through holes in the tunnel floor, to facilitate movement forward and backward. One of the supporting wires was wound around a spool held by a ratchet, and both were kept taut by a weight of about 4 pounds (1.8 kg.) hung on the other wire. The various positions of the pressure tube were indicated by marking a point on one of the supporting wires and measuring its distance from some convenient point, such as the floor of the tunnel. In the open tunnel the positions were checked by frequent measurements (inside the tunnel) of the distance of the pressure opening from one face of the radiator; but in the closed tunnel such measurements could not be made, and only relative positions were obtained, the actual position of the radiator on the plot being estimated from the form of the curve after the latter had been drawn. Preliminary trial showed that consistent results could be obtained with only ordinary care in centering the pressure tube inside of the radiator air cell. In the different sections used the pressure tube occupied between 1 and 3 per cent of the area of the cell through which it passed.

In most cases water was not passed through the radiator sections, and they were at the same temperature as the air; but in one case—type C 9, ¼-inch (0.64 cm.) circular cells 4 inches (10.2 cm.) deep—after the section had been used at room temperature hot water was pumped through it as in regular calorimetric tests, and a mean temperature difference of about 110° F. (61° C.) was maintained between the water and the entering air.

3. *Computation.*

In the closed tunnel the airflow was expressed in pounds per second per square foot of tunnel section (or of radiator core), and pressure difference was expressed in pounds per square foot. Previous work in a wind tunnel under partial vacuum has shown that pressure drop between piezometer rings before and behind the radiator is inversely proportional to the air density at the front ring, for a given mass flow of air; and proportional (very nearly) to the square of the mass flow of air, for a given density. These two relations were used to reduce the observations to a common density, and to correct for small variations in air flow. Observations were taken at from three to six air velocities on each section.

In the open tunnel pressure drop was expressed in pounds per square foot, as before, but the air flow was expressed in miles per hour of the stream through the tunnel, and the corresponding mass flow of air through the radiator was computed from the relation between these

[1] The two tunnels are described in detail in Technical Report No. 60.

two quantities previously found in the regular tests of the radiator.[2] Correction for air density was made as explained in Technical Reports Nos. 60 and 63, for measurements in the open tunnel, and corrections for small variations in velocity was made on the assumption that pressure drop, like head resistance, is proportional to the square of the air velocity. Observations were taken at only one velocity on each section, except that in the case of the Spery type 10 velocities were used with each of two positions of the pressure tube, and the results showed the assumed relation between pressure drop and velocity to be approximately true.

It was found, however, that the pressure gradient along the tube of the radiator seemed not to be the same in the open and closed tunnels for the same mass flow of air and the same density, and this difference was interpreted as indicating an error in the measurement of either the pressure gradient or the air flow. The measurement of the air flow was known to be subject to errors as high as 3 to 5 per cent in different radiators, and it appeared reasonable to regard the pressure gradient within the tube as a good indication of the air flow. Accordingly, for comparison of data obtained in the two tunnels, the results were reduced not to the same air flow as indicated by the usual measurements, but to the same pressure gradient in the tubes (this gradient being proportional to the square of the air flow), and this condition was taken to represent equality of air flow in the two tunnels. The differences between the air flows previously computed and those given by this procedure were from 5 to 8 per cent.

DESCRIPTION OF CURVES.

In the accompanying curves pressure drop in pounds per square foot is plotted against distance along the axis of the air tube of the radiator, the distance being measured in inches forward and backward from the rear face of the section. The location of the two faces are indicated by heavy dotted lines; and for each of the three sections tested in both the open and closed tunnels, head resistance corresponding to the indicated air flow is shown by a solid vertical line marked "R."

Plots 1–5 show results of tests in the closed tunnel, and in plots 6–8 results of tests in the open tunnel are shown, with the closed tunnel curves of equal pressure gradients superposed upon those of the open tunnel.

Plot 2 shows the effect of imparting heat to the air as it goes through the radiator. In this case the mean temperature difference between the water in the radiator and the entering air was about 110° F. (61° C.).

Plot 3 gives a comparison of pressure drop in two radiators whose tubes are of very nearly the same dimensions, but with different conditions of surface. In one (C–10) the cooling surfaces (walls of the air tubes) were very rough, and in the other (C–9) they were somewhat smoothed, though not highly polished.

It will be noted that plots 5 and 8, representing the Spery type (illustrated in the photographs, figs. 9 and 10), are plotted on twice the scale used for the other sections, in order to show clearly the loops in the curves as they follow the four constrictions in the air tubes.

DISCUSSION OF RESULTS.

1. The pressure measurements were of course made under unfavorable conditions, and the results obtained should be used with a little caution, but the consistency of the observations seems to indicate a fair degree of reliability, and a comparison of pressure gradients in the three radiators with circular cells shows a reasonable agreement with values of skin friction found for long tubes by Stanton & Pannell, Saph & Schoder, and others, the values here obtained being slightly higher than those of the other observers, for smooth tubes.

2. The drop in pressure in the air stream through the radiator and the pressure gradient in the tubes are practically proportional to the square of the air flow, for a given air density. This fact and the approximate agreement of observed skin friction with that found by other investigators appear to indicate that there is turbulent flow in the short tubes of the radiators.

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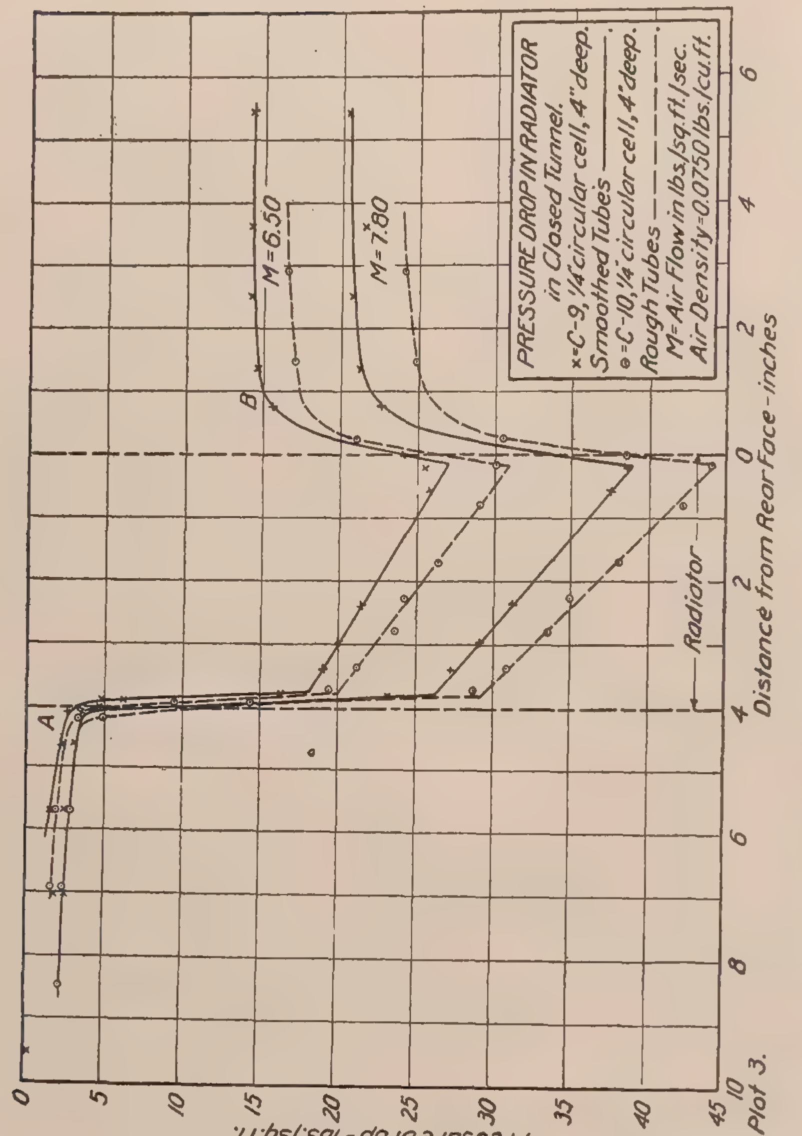
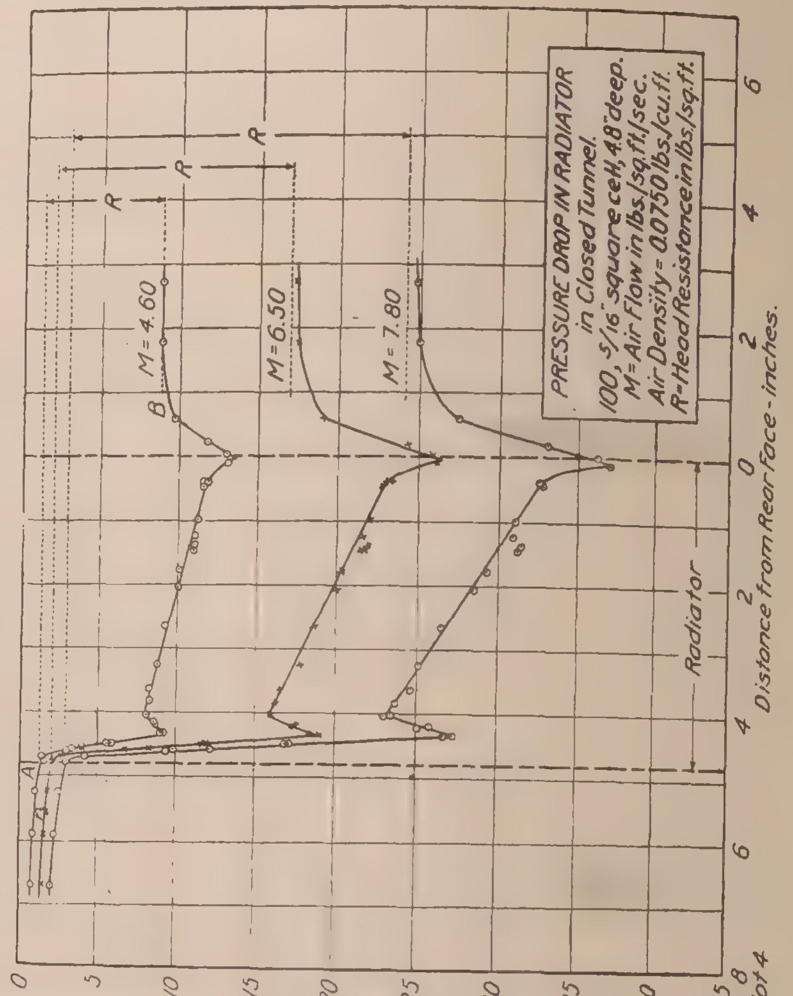
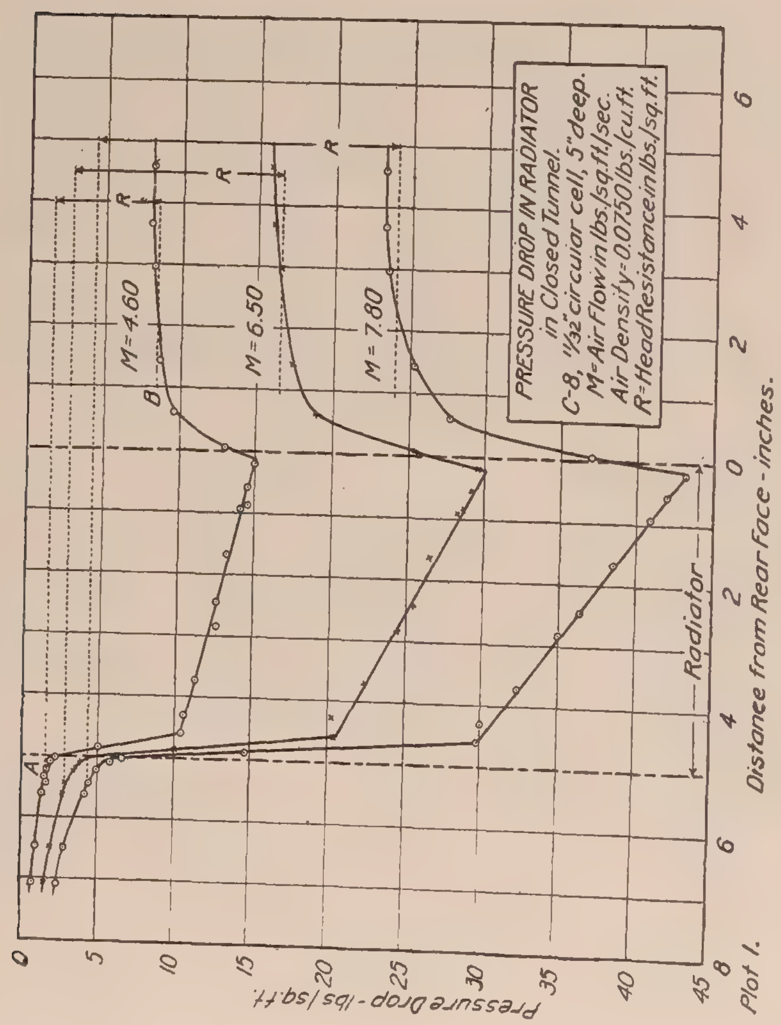
[2] The regular measurement of mass flow of air was made with a special air Venturi meter, described in Technical Report No. 60.



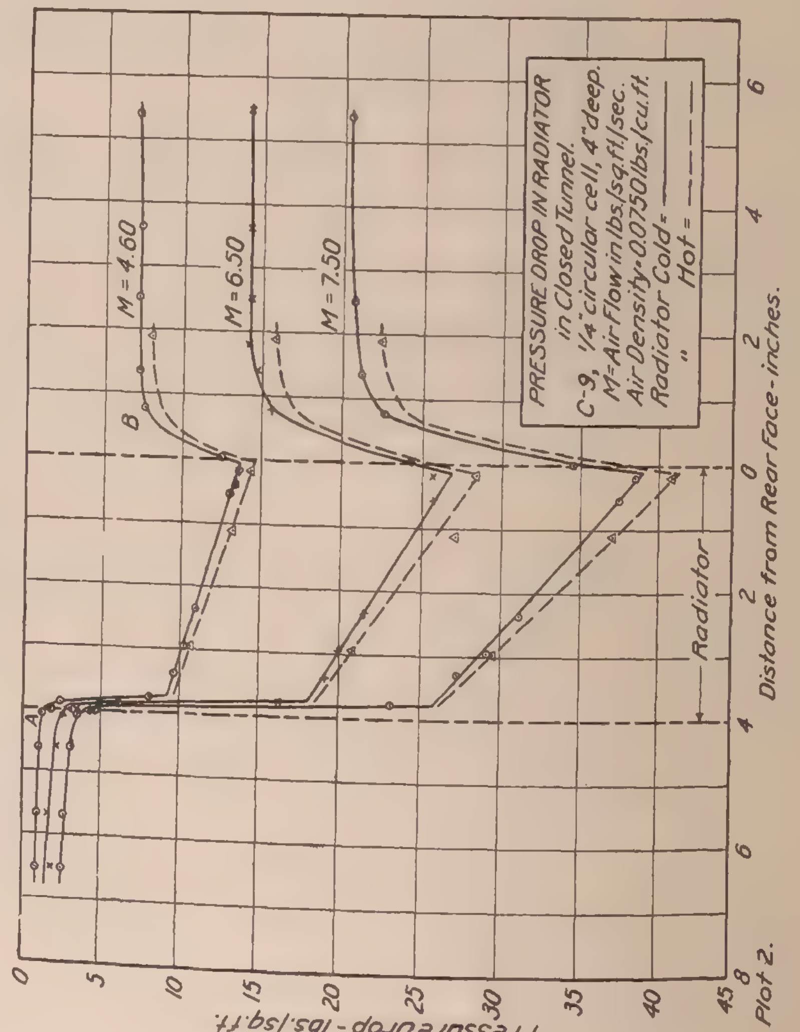
Plot 3.



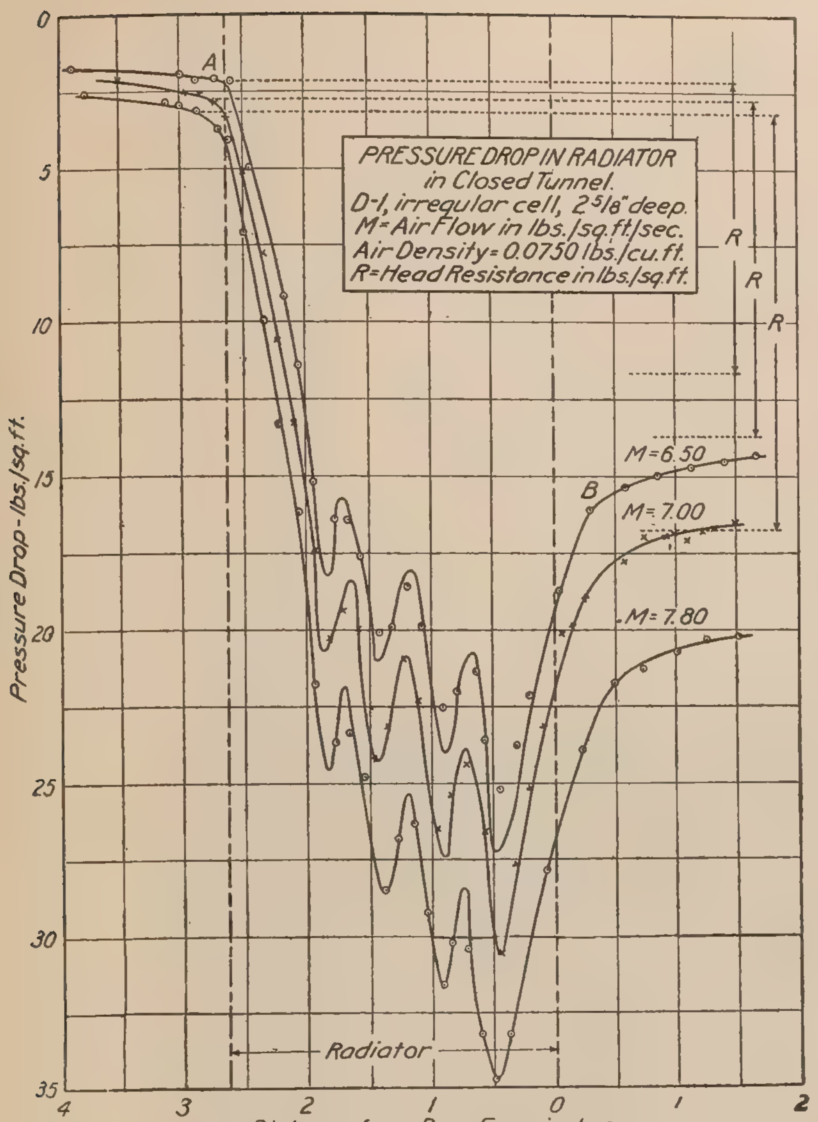
Plot 4.



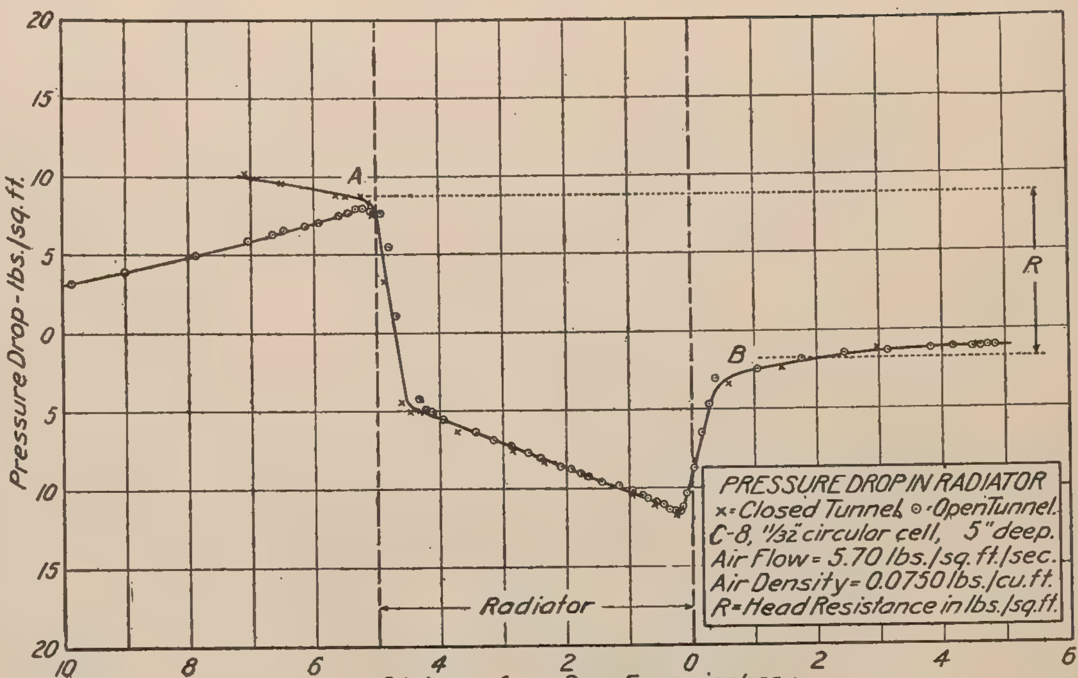
Plot 1.



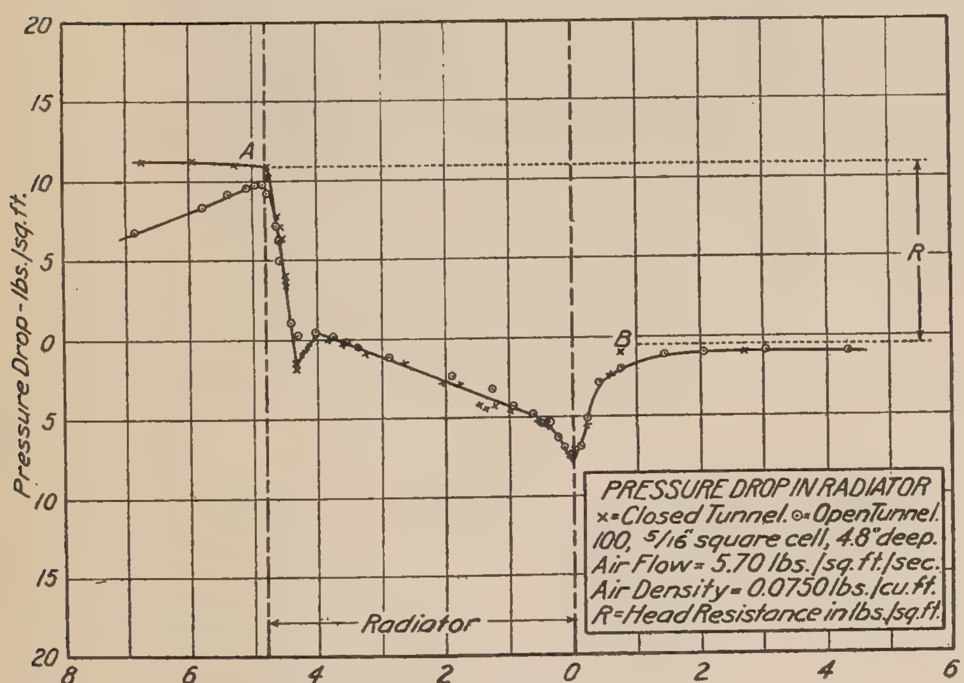
Plot 2.



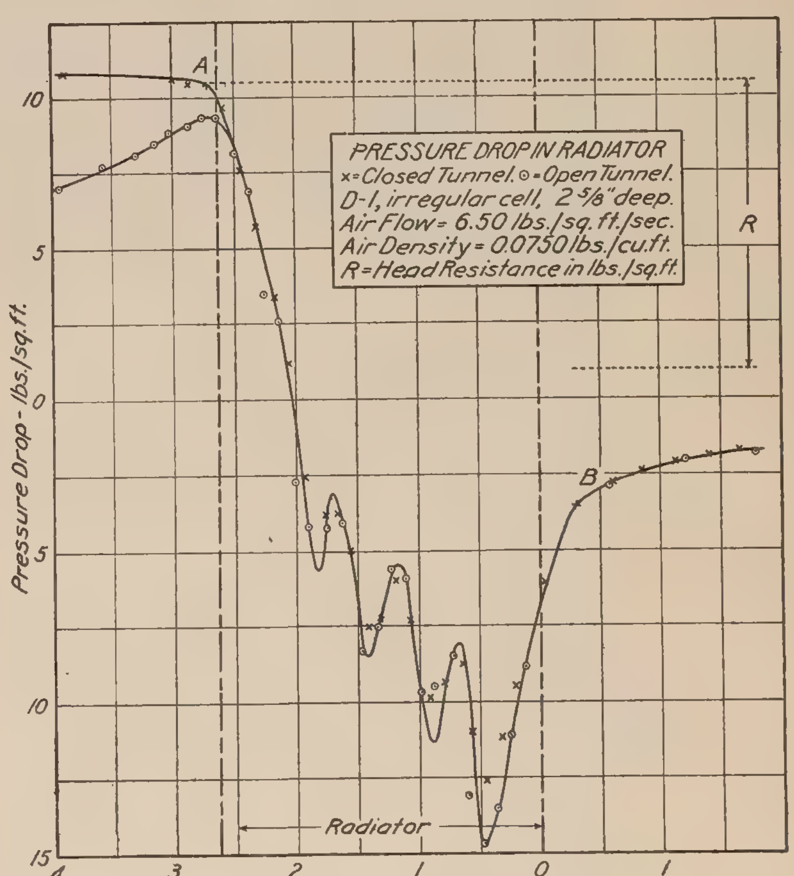
Plot 5



Plot 6.



Plot 7.



Plot 8.

3. The results seem to indicate that the differences formerly observed between pressure drop and head resistance per unit area were apparent rather than real, for the present measurements fail to show convincing evidence to prove that pressure drop is unequal to head resistance per unit area, provided the former is measured at such sections across the air stream as will eliminate the effects of changing velocity at entrance and exit of the radiator. On the other hand, the curves show clearly that if pressure drop is measured between two piezometer rings 1¼ inches (3.2 cm.) from the faces of the radiator (as in the tunnels described), the result will be a value somewhat higher than the head resistance per unit area, and the discrepancy will be greater for some types of core than for others; for in some radiators the high-velocity jets issuing from the tubes persist for a greater distance than in others, before uniform distribution of flow across the channel is regained. In the earlier work, the air flow through the radiator was used as a basis for comparison between pressure drop and head resistance, and errors in the measurement of air flow through the core in the open tunnel are responsible for some of the discrepancies noted; because the measurement of air flow is attended with some difficulty, and is subject to errors that appear to run as high as 8 per cent, and which lead to errors in head resistance about twice as great, since head resistance is proportional to the square of the air flow.

4. A comparison of the curves for the types C–8 and No. 100 indicated a marked difference between radiators in the amount of contraction of the jet at entrance and exit. The type C–8 is made of circular cells expanded to hexagonal form at the ends, and the transition from one form to the other furnishes a very crude streamline form, but one that is much better than that of the type No. 100, which is made with square cells, the ends of the water tubes being merely pressed together to form a joint, with no approach to a streamline form. The difference in the form of the curves for these two types is doubtless due in part to this difference in form of entrance, and also in part to the fact that one has circular and the other square cells. If, in the curves for No. 100, the loops at entrance and exit are interpreted as representing the contraction of the jet, and if it is assumed that the air stream fills the tube in the part for which the pressure curve is straight, Table II shows that the tube is filled for only about 82 per cent of its length. Whether this condition indicates that 82 per cent of the walls of the tubes is scoured by the air stream, and the remaining 18 per cent covered by air that is turbulent but not rapidly changed, is a question that must be postponed until more is known about the conditions of turbulent flow in the radiator tubes.

5. If it is assumed that the loss of head in the part of the tube for which the pressure curve is straight is due to skin friction, Table I shows that the frictional loss is two-thirds of the head resistance for C–8—$\frac{11}{32}$-inch (0.87 cm.) circular cells, 5 inches (12.7 cm.) deep—and one-half of the head resistance for No. 100—$\frac{5}{16}$-inch (0.79 cm.) square cells, 4.8 inches (12.2 cm.) deep.

6. The effect of supplying heat to the radiator is to increase the pressure gradient and the pressure drop between the front and rear faces, the gradient being increased by about 15 per cent in the case of the type C–9—¼-inch (0.64 cm.) circular cells, 4 inches (10.2 cm.) deep—for a mean temperature difference of 110° F. (61° C.) between the water in the radiator and the entering air.

The statement has been made by other investigators that increase in head resistance due to temperature difference can be computed from the increase in momentum of the air as it is heated and consequently made less dense; and it may be well to point out that while such reasoning may apply to pressure drop for a given air flow through the radiator, there is some question about its applicability to head resistance at a given airplane speed.[3] The expansion of the air while it is being heated in the radiator tube tends to do two things: To push the air out from the rear face at a higher velocity than it had at entrance; and to develop a back pressure, acting against the pressure that drives it through the tube. This back pressure tends to retard and reduce the air flow, and by so doing, to decrease the skin friction and consequently

[3] In considering this problem, it must be borne in mind that there are two speeds concerned—the rate of flow of air through the radiator, which may be expressed in pounds per second per square foot of frontal area; and the linear velocity with which the radiator may be regarded as passing through still air. The comparisons made throughout this report are based on equal rates of flow through the radiator.

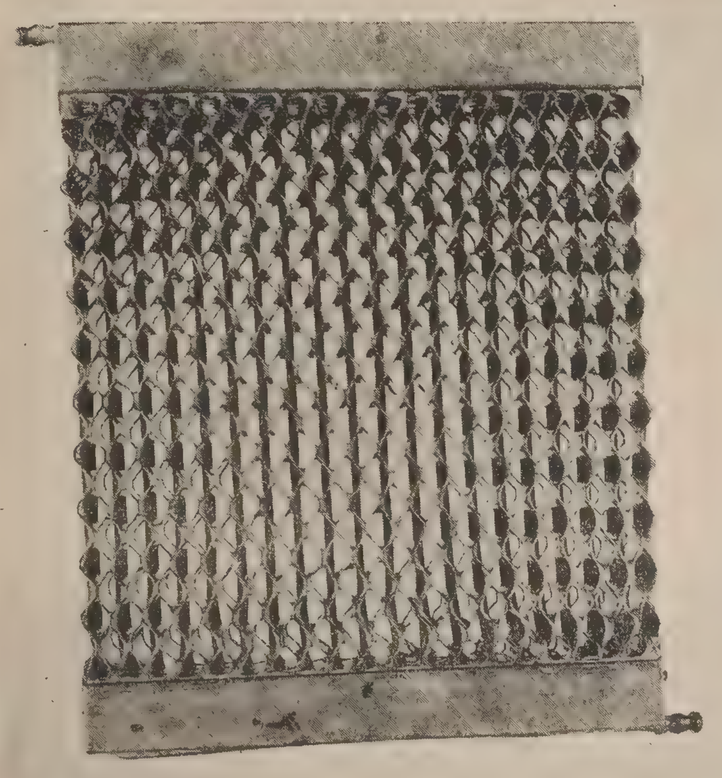


FIG. 9.

FIG. 10.

Spery radiator. See plots 5 and 8.

the head resistance. On the other hand, a decrease in air flow through the radiator requires that a greater part of the approaching air shall be deflected around it, and this condition tends to increase the head resistance. Until experimental evidence is available, the question of the effect of temperature difference on head resistance will be left open.

TABLE I.—*Mass flow of air is in lb. per sec. per sq. ft. frontal area.*

Radiator.	Mass flow of air.	Gradient, lb./ft.² per in.	Length over which gradient is uniform, in.	Friction loss, lb./ft.²	Head resistance, lb./ft.²	Friction loss as % of H. R.
C-8	4.60	1.02	4.3	4.4	6.9	64
	5.70	1.57		6.75	10.5	64
	6.50	2.16		9.3	13.7	68
	7.00	2.46		10.6	15.9	67
	7.80	3.02		13.0	19.7	66
Mean						66
No. 100	4.60	1.02	3.6	3.7	7.4	50
	5.70	1.58		5.7	11.3	50
	6.50	1.95		7.0	14.7	48
	7.00	2.27		8.2	17.0	48
	7.80	2.60		9.4	21.1	45
Mean						48
C-9 (cold)	4.60	1.19	3.6	4.3		
	5.70	1.79		6.45		
	6.50	2.40		8.65		
	7.00	2.67		9.6		
	7.80	3.60		13.0		
C-10	5.70	2.67	3.5	9.3		
	6.50	3.03		10.6		
	7.00	3.46		12.5		
	7.80	4.00		14.0		

TABLE II.

	C-8.	No. 100.
Depth of radiator ... inches	5.0	4.8
Length of air tube ... do	4.6	4.4
Length over which gradient is uniform ... do	4.3	3.6
Per cent of tube length over which gradient is uniform	94	82

TABLE III.—*Effect of heat, radiator C-9.*

Air flow.	Gradient. Cold.	Gradient. Hot.	Per cent increase.
4.60	1.19	1.35	13
5.70	1.79	1.97	10
6.50	2.40	2.78	16
7.00	2.67	3.18	19
7.80	3.60	4.14	15
Mean			15

REPORT No. 89

COMPARISON OF ALCOGAS AVIATION FUEL WITH EXPORT AVIATION GASOLINE

By V. R. GAGE, S. W. SPARROW, and D. R. HARPER, 3d
Bureau of Standards

REPORT No. 89.

COMPARISON OF ALCOGAS AVIATION FUEL WITH EXPORT AVIATION GASOLINE.

By V. R. Gage, S. W. Sparrow, and D. R. Harper, 3d,
Bureau of Standards.

RÉSUMÉ.

This report was prepared for the National Advisory Committee for Aeronautics and describes an investigation conducted for the Navy Department at the altitude laboratory at the Bureau of Standards and its publication is authorized by the Navy Department.

Mixtures of gasoline and alcohol when used in internal-combustion engines designed for gasoline have been found to possess the advantage of alcohol in withstanding high compression without "knock," while retaining advantages of gasoline with regard to starting characteristics. Tests of such fuels for maximum power producing ability and fuel economy at various rates of consumption are thus of practical importance, with especial reference to high-compression engine development.

Aviation alcogas, prepared by the Industrial Alcohol Co., of Baltimore, Md., for trial by the Navy Department and by the latter submitted to the Bureau of Standards for test, was a mixture apparently of about 40 per cent alcohol, 35 per cent gasoline, 17 per cent benzol, and 8 per cent other ingredients. This is not the alcogas prepared for commercial or passenger car use. The exact composition and methods of manufacture are a trade secret.

The tests made for the Navy Department consisted in a direct comparison, in a 12-cylinder Liberty engine, between alcogas and standard "X" (export grade)[1] aviation gasoline with respect to maximum power attainable, and fuel consumption with the leanest mixture giving maximum power. The tests were made in the altitude laboratory at the Bureau of Standards, where controlled conditions simulate those of any altitude up to 30,000 feet. The speed range covered was from 1,400 to 1,800 revolutions per minute and the altitude range from ground level to 25,000 feet. Two series of comparisons were made, one with 5.6 compression ratio pistons and one with 7.2 compression ratio pistons.

The results of the tests showed the following performance of alcogas in comparison with X gasoline as a standard:

(1) At 5.6 compression the same maximum power production at ground level and a general average of 4 per cent more power at altitude, the maximum difference being about 6 per cent at 6,400 feet and 1,800 revolutions per minute.

(2) At 7.2 compression an average and fairly uniform increase of 4 per cent in power at altitude, no comparative figure for X gasoline at ground level being determined with this compression.

(3) A fuel consumption per brake horsepower of from 10 per cent to 15 per cent more by weight to secure this maximum power at any altitude or speed with either compression ratio. Owing to 12 per cent higher density of alcogas, the fuel consumption in terms of volume per brake horsepower is practically the same as with X gasoline.

(4) Thermal efficiency superior by about 15 per cent. A pound of alcogas contains about 22 per cent less heat units than a pound of gasoline, so that in securing more power with 15 per cent greater weight of fuel it is evident that the available energy of alcogas is more fully utilized than that of gasoline.

Considering the high rate of fuel consumption by weight for alcogas in regard to its effect on plane operation it is to be borne in mind that the weight of fuel is usually about one-seventh that of the total plane weight, so that a 15 per cent greater fuel supply is only a 2 per cent increase in the total weight to be lifted and propelled. This is more than compensated by the

[1] Meeting specification of 1917 for the aviation gasoline shipped abroad for the A. E. F.

5 per cent increase in power obtainable. The necessity of greater fuel supply, by weight, of alcogas in comparison to gasoline for a given mileage does not entail the sacrifice of any additional space, as the density of alcogas is 12 per cent greater than that of gasoline.

Since a 7.2 compression ratio is generally considered too high for gasoline, a comparison is desired of the changes of brake horsepower, fuel consumption, and required radiator capacity (ratio of jacket heat loss to brake horsepower) between alcogas at this compression and X gasoline at 5.6 compression. This comparison shows that alcogas with 7.2 compression develops about 15 per cent greater power with the same weight of fuel per unit power. The radiator capacity required per brake horsepower remains the same.

There is no tangible way of comparing "smoothness" of operation of an engine, but the testing staff expressed the opinion that at all times the alcogas gave a smoother running engine than did X gasoline. The use of the fuel was not continued over an interval sufficiently long to give any data in regard to the effect on the engine of continued operation.

OBJECT OF TEST.

This report is a record of a direct comparison of performance of alcogas aviation fuel and standard X (export grade) aviation gasoline in a 12-cylinder Liberty engine. The comparison was made at the request of the Navy Department to determine the relative merits of the two fuels for aviation use, with particular reference to use in extremely high compression engines. Comparison was therefore made at 7.2 compression ratio, as well as at 5.6 compression ratio (about the common ratio in aviation engines of the present date). The measurements made were brake horsepower and fuel consumption for maximum power.

DESCRIPTION OF FUELS.

The physical properties of the two fuels used in these tests are given in Table I and figure I. The distillation figures were determined by the Bureau of Mines method, as described in their Technical Paper No. 214 (Motor Gasoline, Properties, Laboratory Methods of Testing, and Practical Specifications).

The gasoline was the standard reference fuel of this laboratory ("X" gasoline), prepared for the Bureau of Standards by the Atlantic Refining Co. from Pennsylvania crude oil. It complies with Specification No. 3512 of the Bureau of Aircraft Production for export aviation gasoline for the use of the American Expeditionary Forces, 1918.

TABLE I.—*Distillation and other properties of alcogas and X gasoline.*

	Aviation alcogas.	X gasoline.
Heating value (total):		
British thermal units per pound	15,910	20,340.
Calories per gram	8,840	11,300.
Appearance	Clear lavender	Clear water white.
Odor	Alcohol and ether	Gasoline.
Specific gravity at 15.6° C	0.799	0.710.
Distillation:		
Initial boiling point	60° C	59° C.
10 per cent	65° C	72° C.
20 per cent	67° C	77° C.
30 per cent	69° C	82° C.
40 per cent	71° C	87° C.
50 per cent	73° C	92° C.
60 per cent	74° C	97° C.
70 per cent	76° C	103° C.
80 per cent	78° C	111° C.
90 per cent	145° C	127° C.
95 per cent	177° C	150° C.
Dry point	184° C. (97 per cent)	153° C. (96 per cent).
Residue, per cent	1	1.5.
Loss, per cent	2	2.5.
Reaction to litmus	Slightly acid	
Corrosion	Black deposit	
Gum, per cent	0.02	

The alcogas fuel was prepared by the Industrial Alcohol Co. of America, Baltimore, Md., for aviation use. It was not the same mixture as that prepared for commercial or for passenger car use. Although the composition and methods of manufacture are a trade secret, it is probable that the composition of the aviation alcogas used in these tests was about 40 per cent alcohol, 35 per cent gasoline, 17 per cent benzol, 8 per cent toluol, ether, etc. Since commercial alcohol and gasoline are not readily miscible, it is necessary to add some ingredient, such as benzol, to secure homogeneity and another must be added to lower the freezing point. Alcogas is a mixture somewhat similar to the Taylor fuel, 33 per cent alcohol, 40 per cent gasoline, 27 per cent benzol, tested by the Bureau of Mines in experimental aviation engines and found capable of withstanding a compression of 8.2 without knock. It is this characteristic which makes mixtures of this type worthy of special study as engine fuel.

DESCRIPTION OF TEST PLANT.

The tests were made in the altitude chamber at the Bureau of Standards, which is designed to give conditions of pressure and temperature such as may be found at any altitude from ground to 30,000 feet. An air pump reduces the pressure in the chamber and in the engine exhaust piping. A refrigerating equipment, together with electric heating coils, give ready control of the temperature conditions at the engine. All controls and measuring devices are located outside of the chamber. A detailed description of the altitude chamber is given in Report No. 44 of the National Advisory Committee for Aeronautics.

A Liberty 12-cylinder aviation engine was used, manufacturers' No. 323, rebuilt at the Bureau of Standards. The oil used was Mobile B. When running with the regular 5.6 compression ratio the engine was standard except that it was equipped with two Stromberg 2-inch duplex carbureters, permitting extreme latitude in the adjustment of air to fuel ratio of the mixture supplied to the engine. Special pistons were fitted to give the 7.2 compression ratio. In each case, the clearance volume was determined by filling the compression space with oil. The compression pressures were measured with a check valve and gage. With the 5.6 compression ratio at 900 revolutions per minute, the compression pressure was about 125 pounds per square inch. With a 7.2 compression ratio, at the same speed, it was about 170 pounds per square inch.

Fuel consumption was measured by direct weighing, noting the time to consume a predetermined weight of fuel. The laboratory is equipped with two fuel tanks, each on a platform scale, and a valve in the intake supply line of the carbureter shifts the supply from one tank to the other.

DESCRIPTION OF TEST PROCEDURE.

These tests were made for the specific purpose of comparing alcogas to X gasoline as an aviation fuel; nevertheless many observations were made that have only indirect bearing upon the fuel comparison, but which, in connection with similar data from other tests, may lead to a more complete understanding of the many factors entering into the various problems connected with internal combustion engines.

The manner of conducting the tests was, briefly, as follows: The engine was started on one of the fuels, and the air, load, speed, oil, jacket, etc., conditions adjusted. Starting with a mixture known to be rich, the fuel supply was gradually reduced and the maximum torque noted, the leaning of the mixture being continued until the torque was appreciably below its maximum value; then the fuel flow was increased only enough to again obtain the maximum torque. All the data in this test were secured with engine throttles wide open. When conditions and adjustments were as desired, observations were made of the speed, load, various pressures and temperatures and quantities, while the time required to use a certain weight of the fuel was noted. At the end of the run on one of the fuels the valves were turned so as to supply the engine with the other fuel. After sufficient time to be sure that none of the previous fuel remained in the line unused, the carbureter was again adjusted for maximum torque with minimum fuel, in the manner described above. By following this procedure there was very little chance for any change of engine condition to enter into the comparative results from the

two fuels. After the tests with ordinary (5.6) compression ratio, the engine was taken down, the special 7.2 compression pistons were put in, the engine was thoroughly cleaned, overhauled and some replacements of parts made. This overhaul had no influence on the comparison of the two fuels with either one of the compression ratios, all such comparative runs being made according to the procedure just described, which eliminates engine changes. Other comparisons such as that of the engine performance under different compression ratios, may be affected to some slight degree by the overhaul, and deductions from such data will not have quite as high a degree of accuracy as they would from a test conducted with primary attention to constancy of engine conditions. This fact should be borne in mind in examining figures 24, 25, 26, and 27, although it should not be inferred that these curves fail to merit reasonable confidence.

RESULTS OF TEST.

The test data have been summarized in the curves forming figures 2 to 27. The first group, figures 2 to 12, include the data obtained with the higher or 7.2 compression ratio pistons. Figures 13 to 23 include the lower of 5.6 compression ratio results. Figures 24 and 25 compare the two compression ratios, with either fuel (fig. 24, alcogas, and fig. 25, X gasoline) as to effect on power, thermal efficiency, and fuel consumption. Figures 26 and 27 are a comparison of alcogas at 7.2 compression raio and X gasoline at 5.6 compression ratio, in regard to brake horsepower, pounds of fuel per brake horsepower hour, and heat lost to jacket per brake horsepower.

On figure 2 and figure 13 are plotted brake mean effective pressures versus revolutions per minute. The points are computed directly from observed data. The faired curves are used as the basis of the curves shown in figures 3 and 14, brake horsepower versus revolutions per minute on which the points shown were computed directly from test data, without previous averaging in any respect.

The fuel consumption is shown on figures 4 and 15 in total weight consumed per hour, and on figures 5 and 16, in pounds per brake horsepower hour. The first named curves have been used as an aid in judging engine performance and as a check on other curves. They do not contribute the basis of the second set, which are faired from points computed directly from the original data. In locating faired curves less weight is given to those observation points which the notes made during test show may have been subject to uncertainties of engine behavior.

The scattering of the points on figures 5 and 15 indicate inconsistencies in fuel consumption data rather out of proportion to the exactness with which the power and other measurements could readily be made. This may be attributed to a certain slowness in response of the engine when changes of mixture were made, a sluggishness distinctly less in evidence at 7.2 compression than at 5.6 compression.

Even under the most favorable conditions, considerable change in mixture is possible for a very slight change in power, so that great accuracy is not possible in duplicating the condition of minimum fuel for maximum power. This, not lack of precision in measuring the fuel, is the explanation for the scattering of the points on figures 5 and 16. It was observed that with the 7.2 compression less change of mixture was required to produce a noticeable change in power than with the 5.6 compression. Consistency in fuel consumption data could have been secured had a single arbitrary carburetor adjustment been chosen and used for each fuel, but then the results would have been of no value for the object of these tests. It was desired to find the best performance of the engine with each fuel, absolutely independent of the characteristics of the carburetor. Incidentally, the weight of air used was also determined, so that the data can be used to obtain information as to what the carburetor characteristics should be, for all conditions existing in these tests. The values of thermal efficiency and fuel consumption per unit power, figures 24 and 25, both include factors dependent upon the carburetor adjustment, and are subject to a possible error of about 2 per cent for this reason, even though the actual data were obtained with greater precision.

Measurements of pressure and temperature in intake and exhaust manifolds, valve conditions, etc., give qualitative indications in regard to richness of mixture, and these incidental test data afford a slight assistance in locating the faired curves of figures 5 and 16, by indicating relative weight to be given to scattered points. No attempt has been made, however, to apply hypothetical corrections for unknown conditions, and the points are as observed.

The faired curves of figures 5 and 16, fuel consumption per unit power versus speed are used as a basis for the plot in the other coordinate, fuel consumption per unit power versus altitude in figures 6 and 17.

The heat balance, or record of relative utilization of the heat supplied by the fuels, is shown on figures 7 to 10 and figures 18 to 21. Four partitions of energy are diagrammed, namely (1) the percentage measured as appearing in brake horsepower, (2) that discharged as heat in exhaust gases, (3) that absorbed in the jacket cooling water, and (4) the residual or difference between the sum of the three foregoing percentages and 100 per cent. The residual heat is in reality greater than this value because of the unmeasured heat obtained from combustion of the lubricating oil. The fuel which escapes unburned in the exhaust gases is accounted for in the "residual" values, the exhaust losses including only the sensible heat of the gases and the latent heat of the water vapor. The points shown are computed directly from test data. There being no data on "residual" heat, these curves are located from the faired curves of the other three plots, rather than from points computed as residual to the observation points at the basis of these plots.

Although the curves of heat balance are plotted against speed as abscissæ, no significance attaches to the slope as rate of change of heat distribution with change of speed. For example, no inferences from figures 18 and 19 that brake horsepower, jacket and exhaust heats certainly ncrease with speed at 1,250 feet altitude and decrease at 6,400 feet altitude would be justifiable; there being many variables (including the air to fuel ratio) which affect heat distribution quite independent of speed, and which may have conspired accidently to slope one set of curves up and the other down. What the curves do show is the relative situation as regards the two fuels, and the general magnitude for both at any particular speed.

The curves indicate a pronounced difference in the two fuels, as regards heat appearing in the exhaust gases and in the residual group. When using alcogas, a greater percentage of the heat supplied appeared in the exhaust, more was converted to useful work, and less wasted unburnt (part of the residual), than when using X gasoline.

Selecting the normal engine speeds of 1,600 revolutions per minute and 1,700 revolutions per minute, the heat balance curves of figures 7 to 10 and 18 to 21 are recast into plots of heat balance versus altitude, figures 11 and 22. The curves correspond exactly to the faired curves of the parent set, rather than the estimated smoothest curve through the points, which are computed directly from observed data, and it will be noted that they match remarkably closely.

The slope of the curves should be interpreted with the same degree of reservation noted above for the companion set of curves, nevertheless it seems justifiable to accept the reverse curvature, which is rather marked, as a real reversal with altitude and not a mere accidental coincidence of some undetermined cause depressing or raising values. This conclusion is partly from results of other tests (with different fuels) where in numerous instances evidence has appeared that most complete combustion of gasoline is secured at conditions corresponding to the altitude of 10,000 to 15,000 feet.

In figures 12 and 23 (compression ratios 7.2 and 5.6, respectively) are summarized the differences between the engine performance of the two fuels. The performance of X gasoline is used as the reference zero in each case, and the percentage increase or decrease obtained with alcogas, as shown by the faired curves of the preceding figures, is plotted. Comparison is made of brake horsepower, fuel consumption per brake horsepower, and thermal efficiency at all speeds and altitudes of the test.

The curves of figures 24, 25, 26, and 27, relating to engine performance rather than directly to fuel comparison, have been discussed in the closing paragraph of the section entitled "Description of test procedure."

CONCLUSIONS.

Brake horsepower. (Figs. 12 and 23).—The alcogas shows a better maximum power-producing ability than X gasoline at all speeds and altitudes, except at ground, the maximum difference being 6 per cent. At ground level the two fuels gave the same result at 5.6 compression, while at 7.2 compression comparison was omitted because of the tendency of gasoline to knock at such high compression. The most common difference, omitting the extremely high and low speeds and considering all altitudes, is about 4 per cent, which may be accepted as the figure for superiority in brake horsepower of alcogas over X gasoline.

Fuel consumption.—The gain in power-producing ability noted above for alcogas is at the expense of considerable increase in fuel consumption. Figure 12 shows differences reaching 20 per cent. The general average is an excess consumption, per brake horsepower, of alcogas exceeding 10 per cent (by weight) at 5.6 compression ratio and nearly 15 per cent at 7.2 compression ratio. (Comparison by volume is noted below.)

Thermal efficiency.—Alcogas shows about 15 per cent higher thermal efficiency than gasoline. This figure, as a general average, is taken from figures 12 and 23. Stated in terms of brake thermal efficiency of an engine, 15 per cent superiority of alcogas over gasoline means that if an engine using gasoline is 25 per cent efficient, it would be 28 to 29 per cent efficient on alcogas.

Comparisons of alcogas and X gasoline by volume.—Alcogas is 12 per cent more dense than gasoline; consequently all the above figures are very different when comparison is made on the basis of the pint or gallon as a unit instead of the pound. The maximum brake horsepower attainable is independent of this unit, so that the figure is 4 per cent, as before. The excess fuel consumption per brake horsepower of 10 to 15 per cent by weight becomes practically zero on the volume basis. The total heating value per gallon of alcogas is about 106,000 British thermal units and of gasoline 120,000 British thermal units, a difference of 12 per cent referred to gasoline as a base, instead of 22 per cent difference as by weight. This figure is seen to be of the same order of magnitude as the difference in thermal efficiencies of the fuels. Computing the effective useful work obtainable (product of British thermal units supplied and thermal efficiency) it is found to be the same from a gallon of either alcogas or gasoline.

General engine performance.—While there is no tangible method of comparing the "smoothness" of operation of the engine, the testing staff felt that alcogas gave a "smoother" running engine at all times than did the X gasoline. No tests were made to determine the condition of the engine after continued use of alcogas fuel, but no evidence was found of any evil effects.

Apparently the change in compression ratio has about the same effect, no matter which of the two fuels is used, until the temperature and pressure conditions are such as to cause poor engine operation with gasoline. The main advantage of alcogas seems to be that it is known to be free from tendency to knock on ground level when using the 7.2 compression with wide-open throttle.

The numerical values for effect of changing compression ratio, figures 24, 25, 26, and 27 are subject to an undetermined uncertainty, because of the overhauling of the engine (see section on test procedure), but it is probable that this uncertainty is small and that it is safe to state that the increase of brake horsepower at 7.2 compression over that at 5.6 compression averages at least 10 per cent for all speeds and altitudes, and that the fuel economy for maximum power is improved, so that the fuel consumption per brake horsepower and the thermal efficiency are at least 10 per cent better with the higher compression. It may be of interest to note that the "air-standard efficiency" (based on an ideal engine following Otto cycle) increases about 10 per cent upon raising the compression ratio of the Liberty 12-cylinder aviation engine from 5.6 to 7.2 results in about the expected change in efficiency, power, etc.

It is generally considered that a 7.2 compression is too high for gasoline fuel. Therefore it is of interest to compare the engine performance using gasoline with the 5.6 compression with performance when using alcogas with 7.2 compression. A general comparison of the change of brake horsepower, fuel consumption, and required radiator capacity (ratio of jacket

heat loss to brake horsepower) under these conditions is given on figures 26 and 27. Alcogas with the 7.2 compression pistons gives a general average of about 15 per cent more power than X gasoline with the 5.6 compression pistons. The pounds of fuel per unit power is about the same, perhaps favoring slightly the use of alcogas with the higher compression. Figure 27, comparing the ratio of heat in jacket water to power, shows this ratio to be the same, but as the power obtained from alcogas in a 7.2 compression engine is greater, more radiator capacity would be required than when using X gasoline in a 5.6 compression engine.

NOVEMBER 25, 1919.

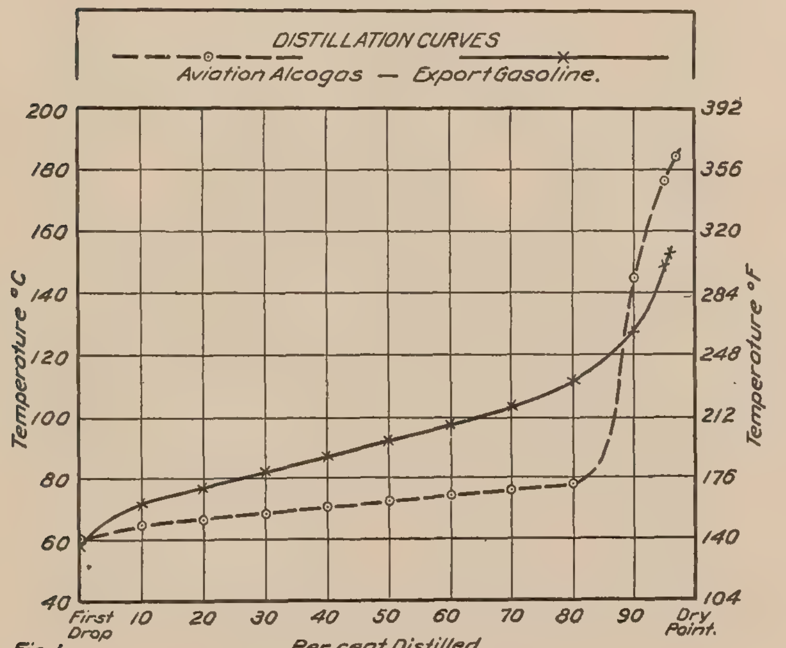


Fig. 1.

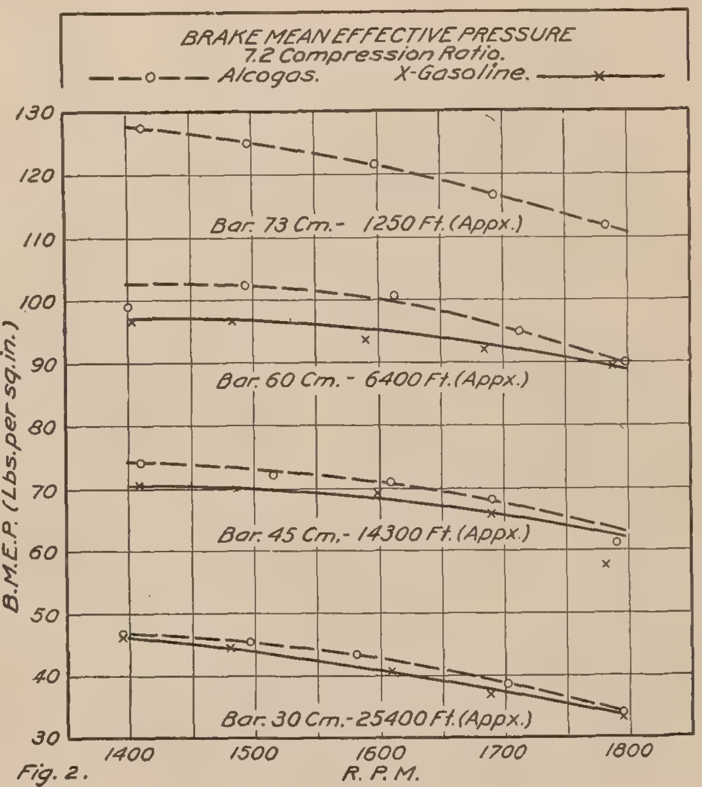


Fig. 2.

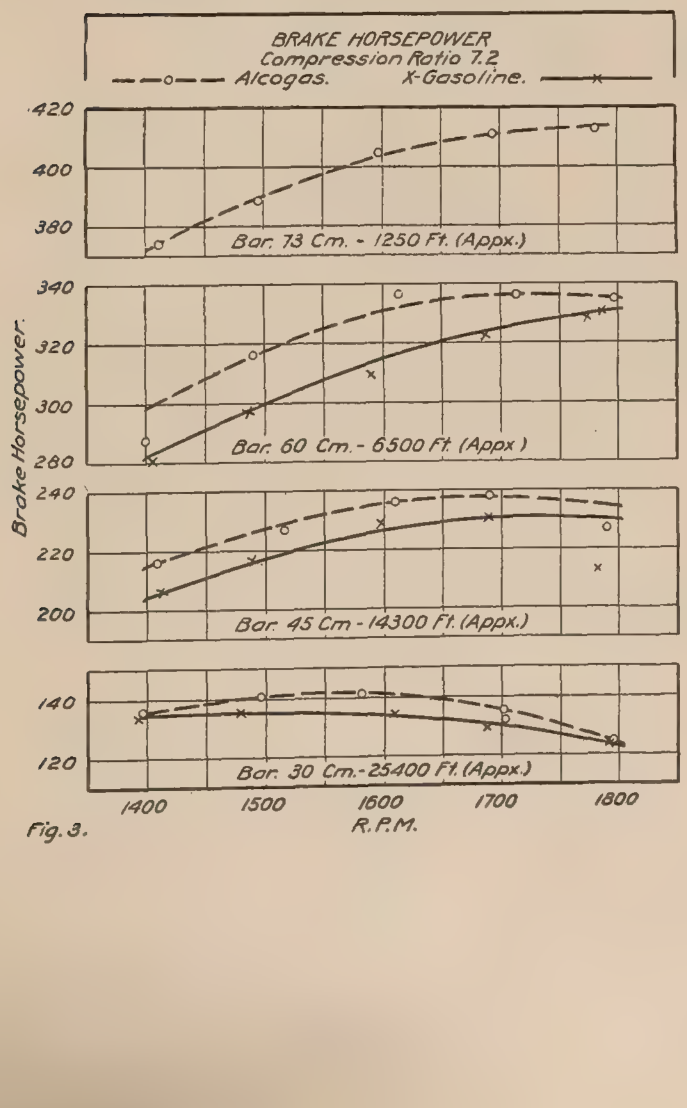


Fig. 3.

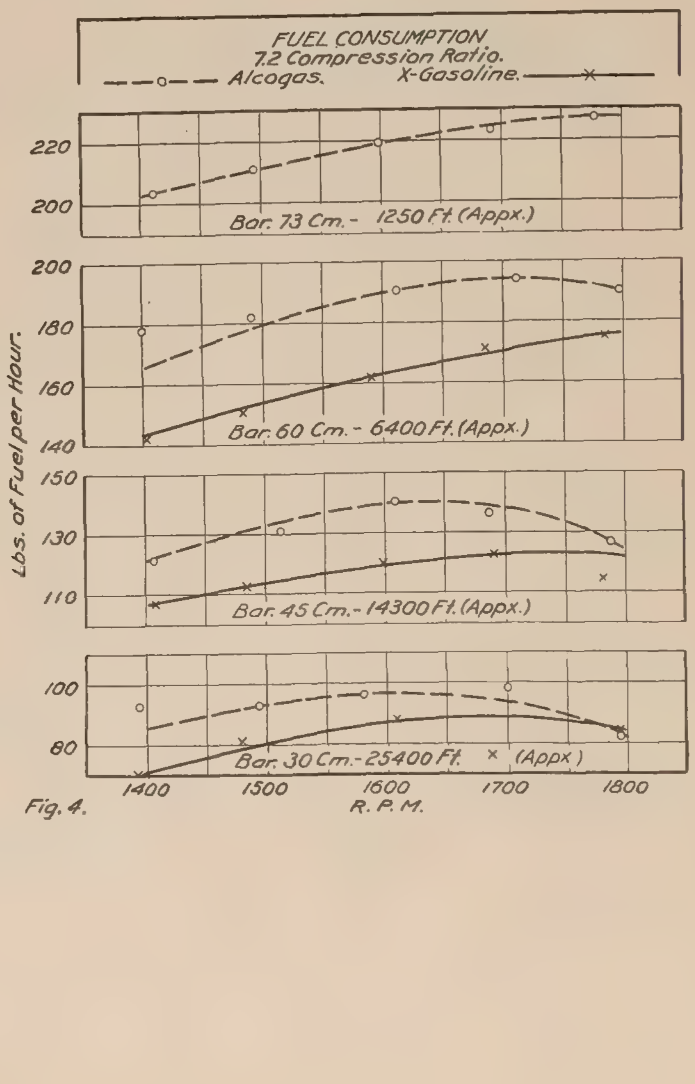


Fig. 4.

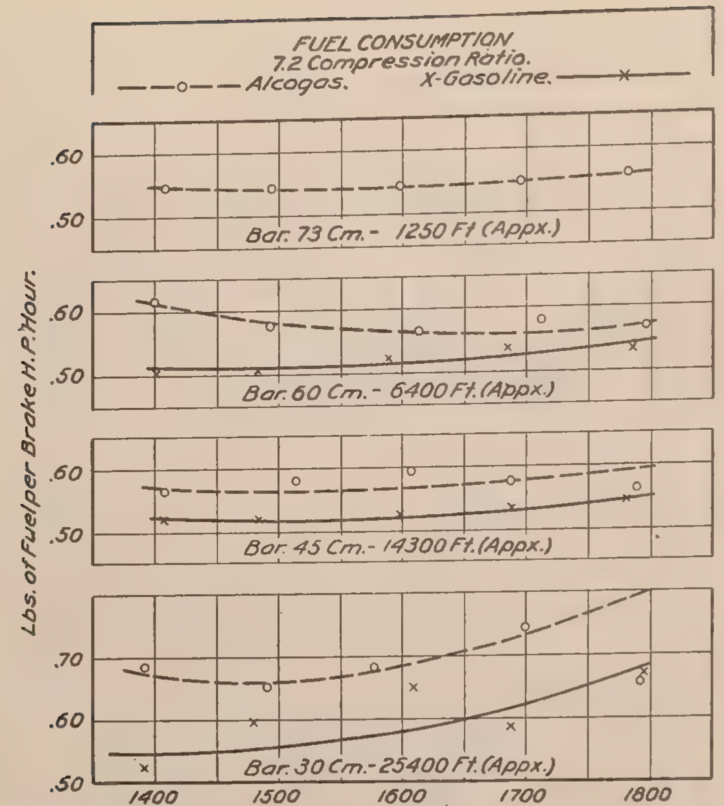


Fig. 5.

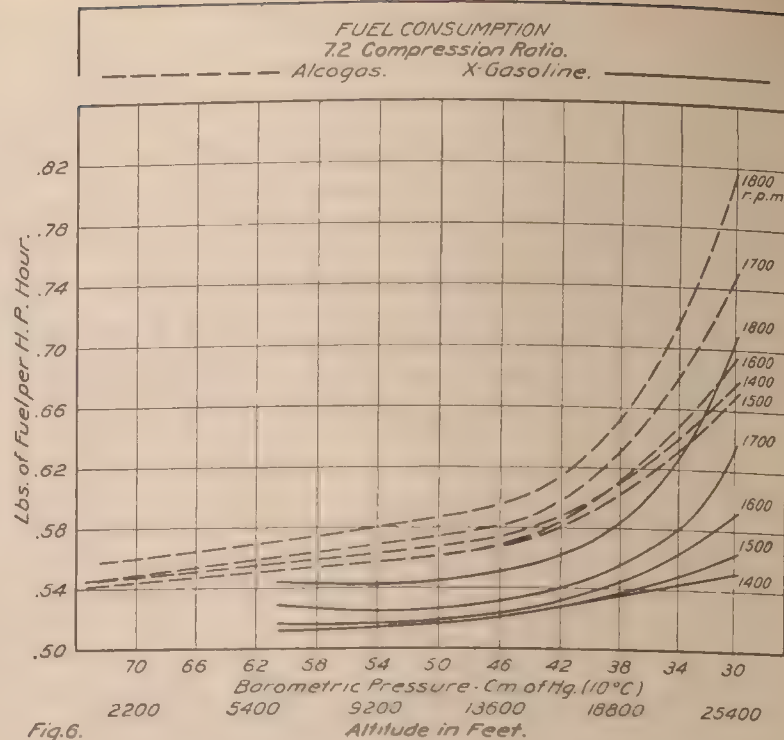


Fig. 6.

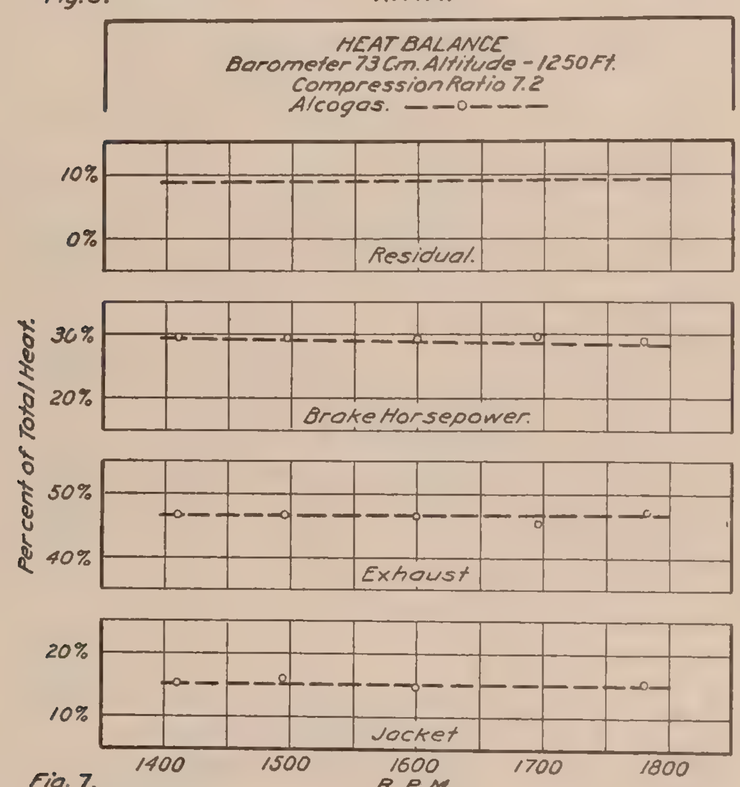


Fig. 7.

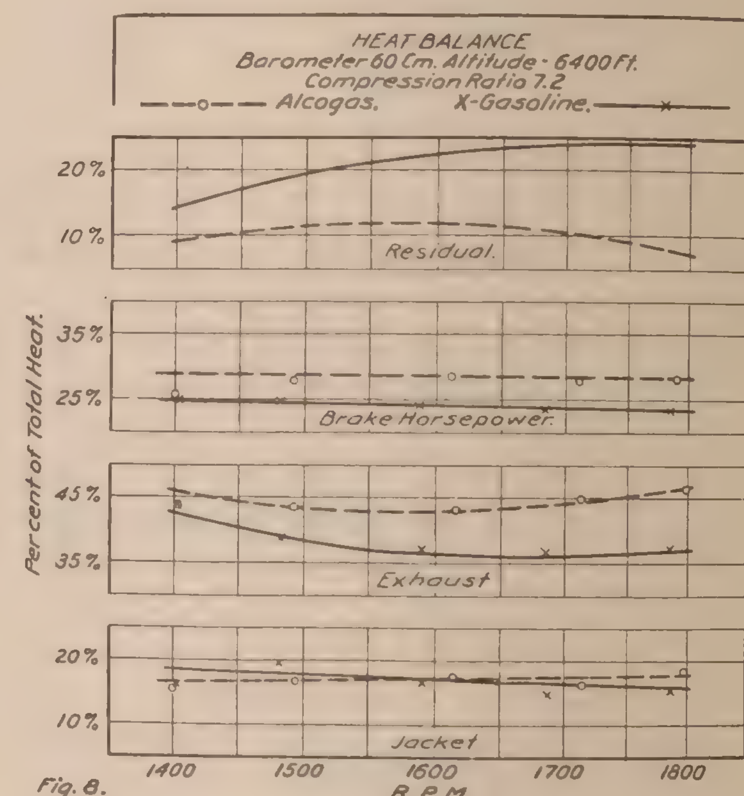


Fig. 8.

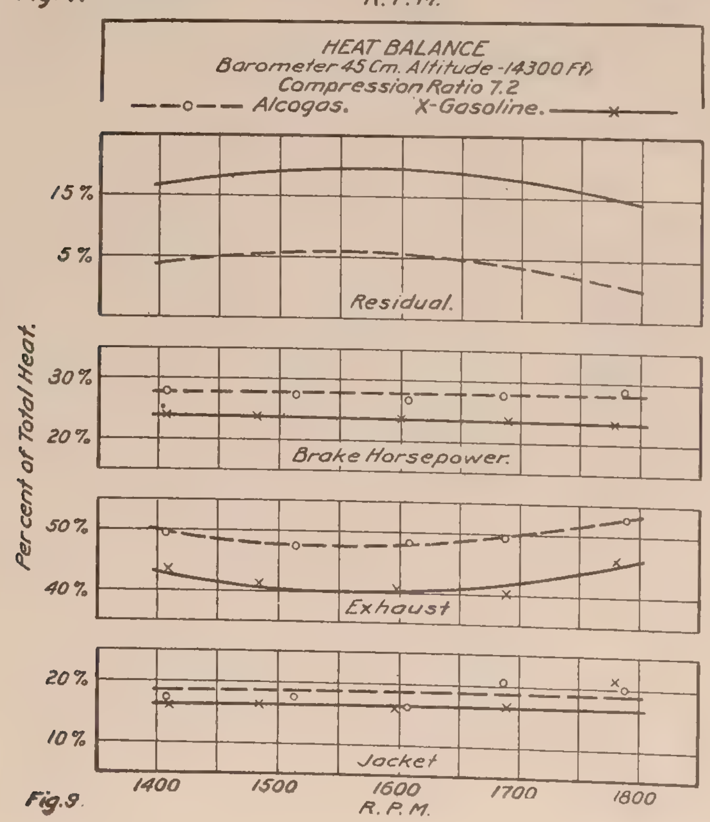


Fig. 9.

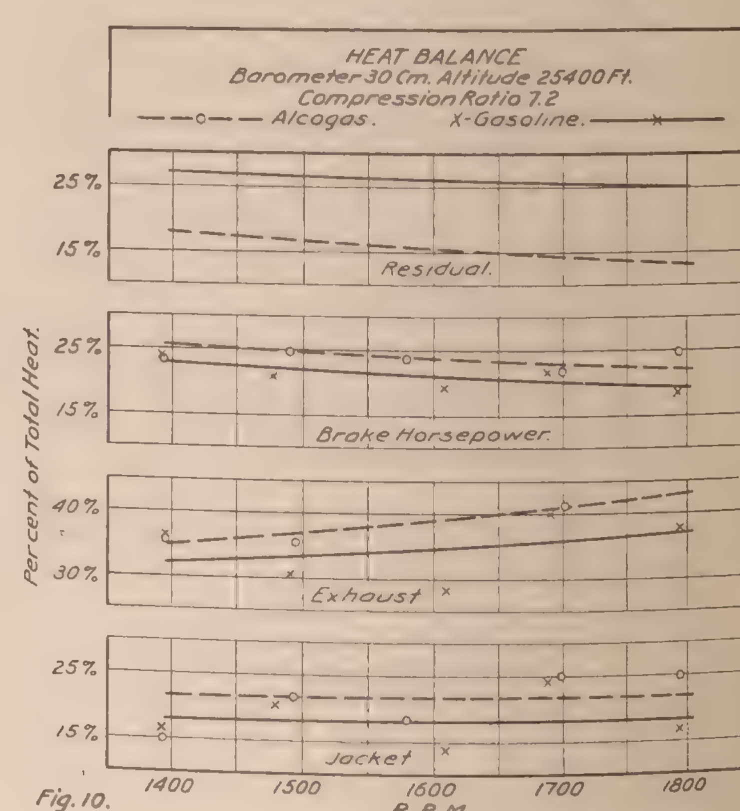


Fig. 10.

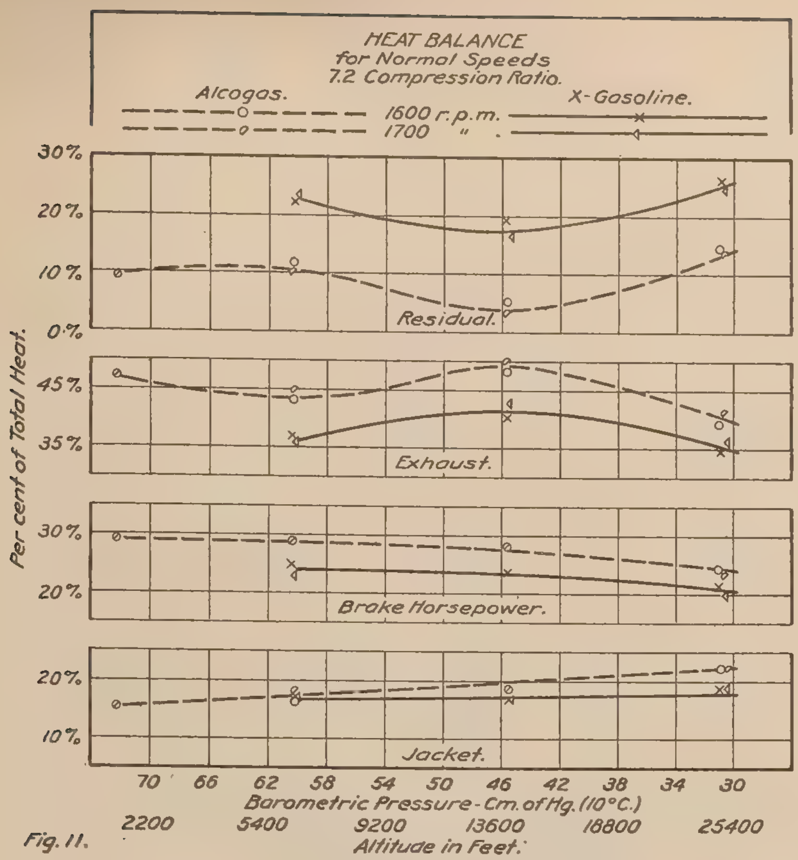


Fig. 11.

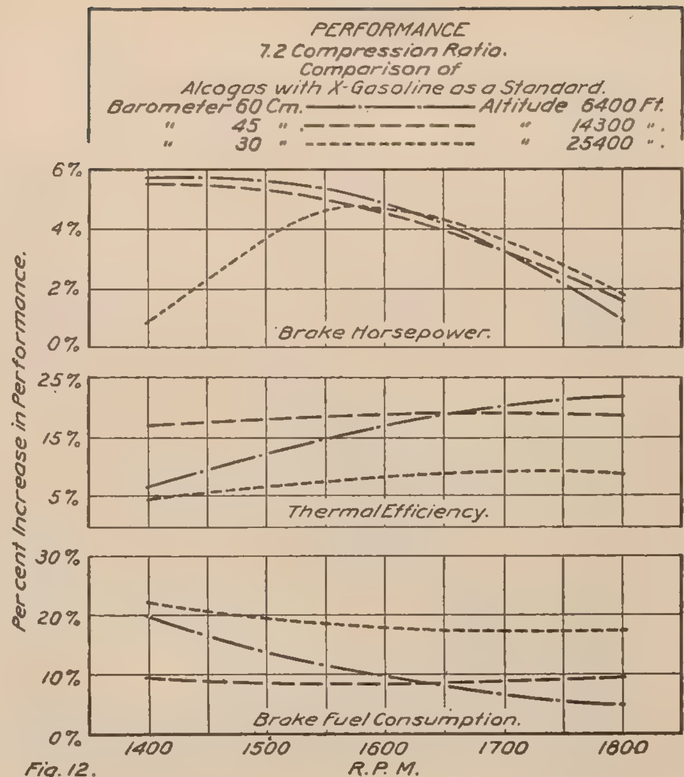


Fig. 12.

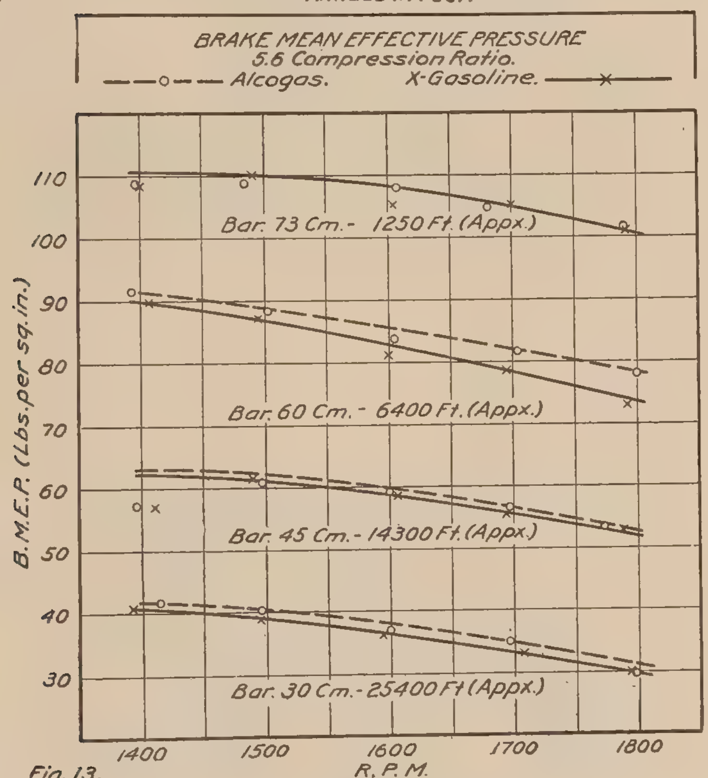


Fig. 13.

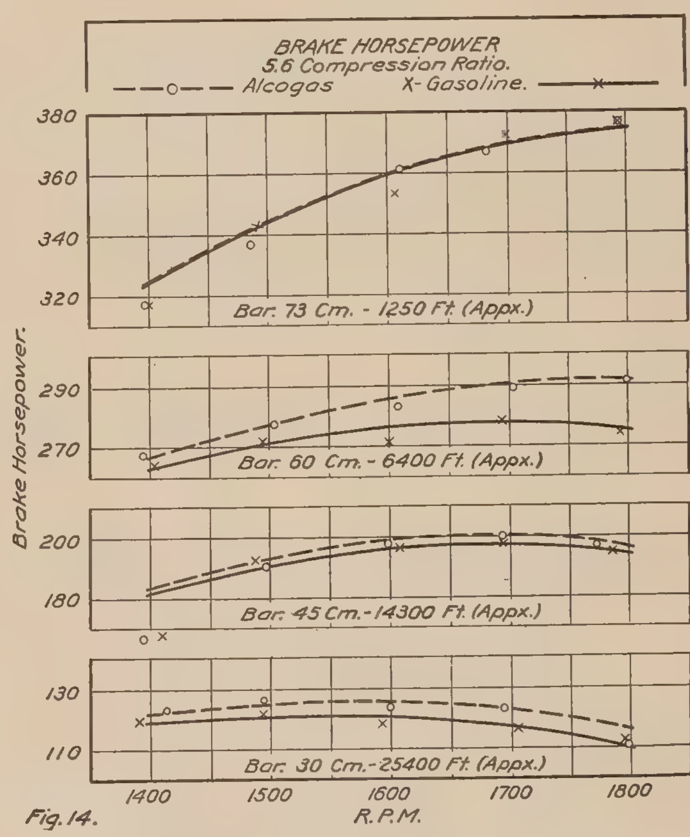


Fig. 14.

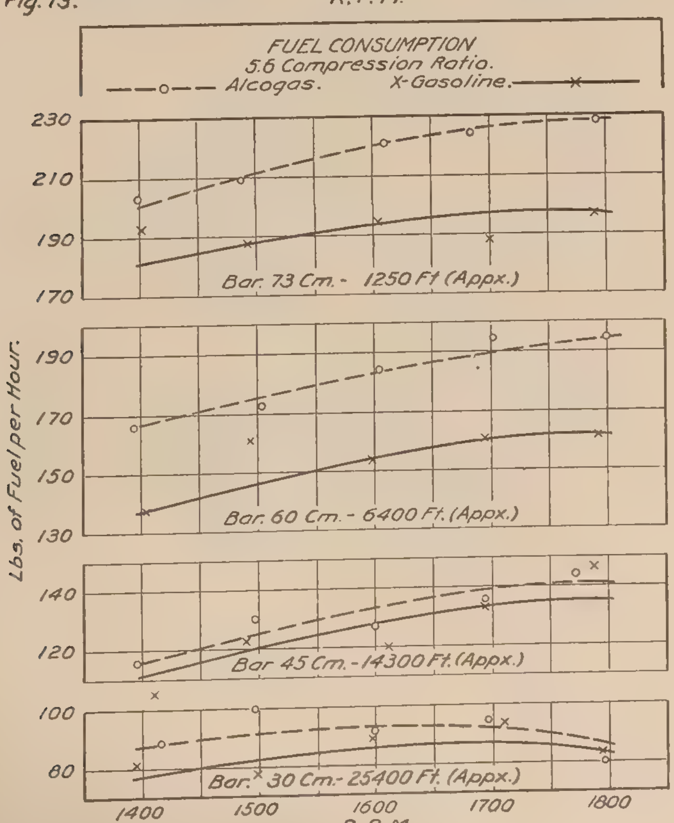


Fig. 15.

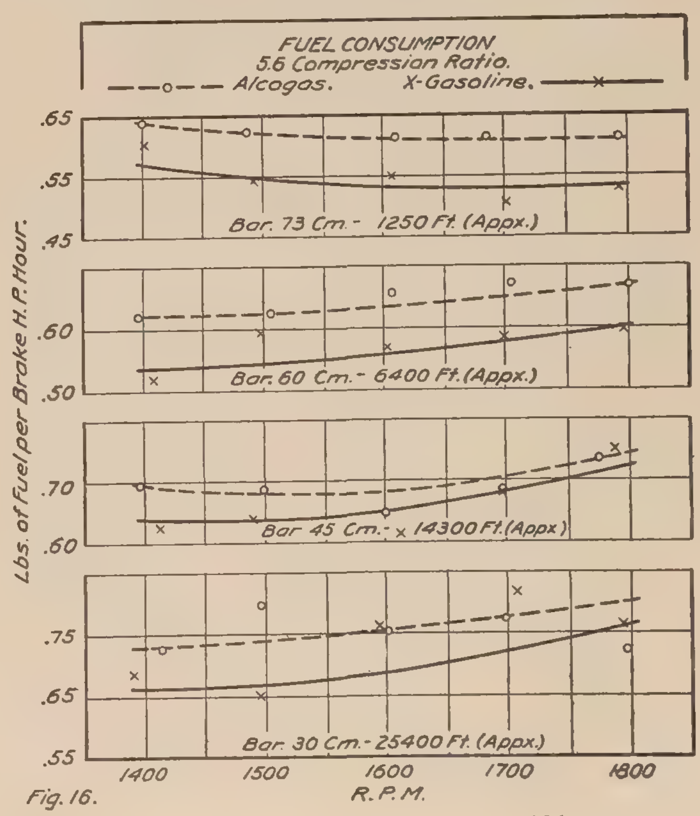


Fig. 16.

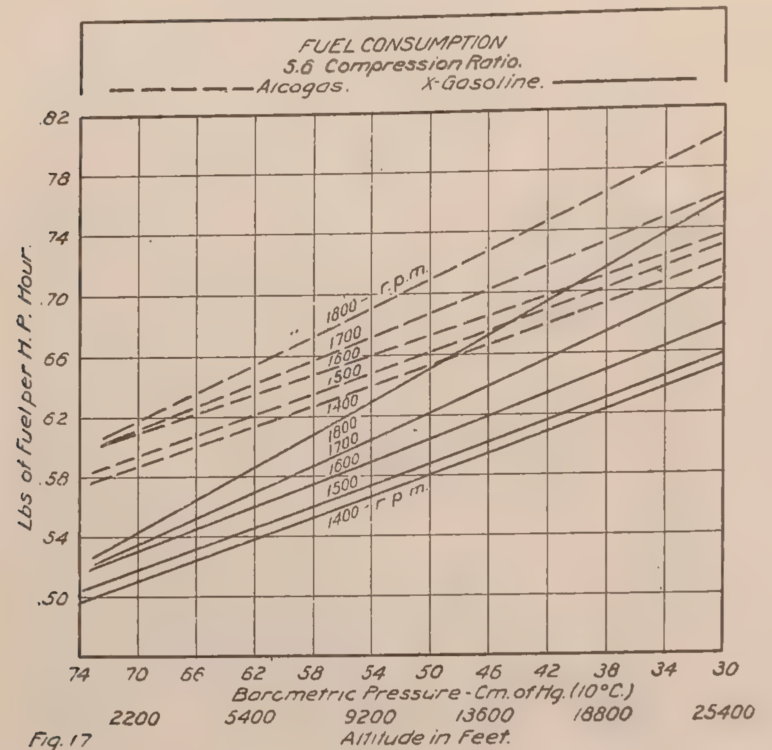


Fig. 17

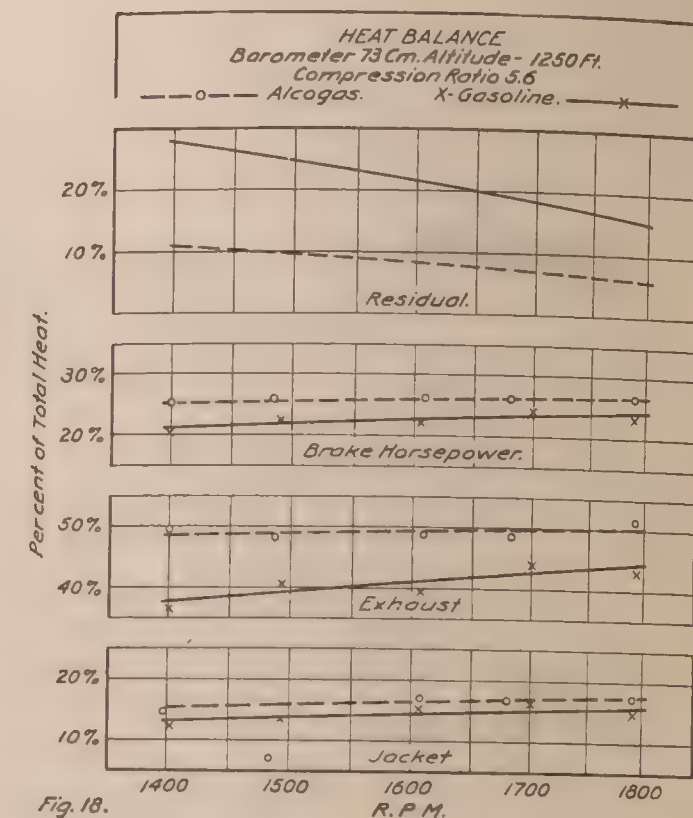


Fig. 18.

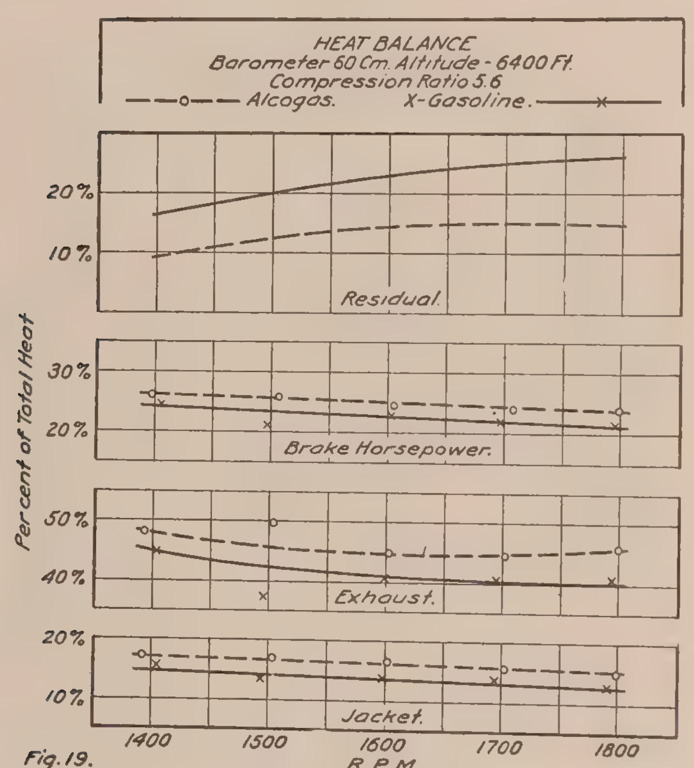


Fig. 19.

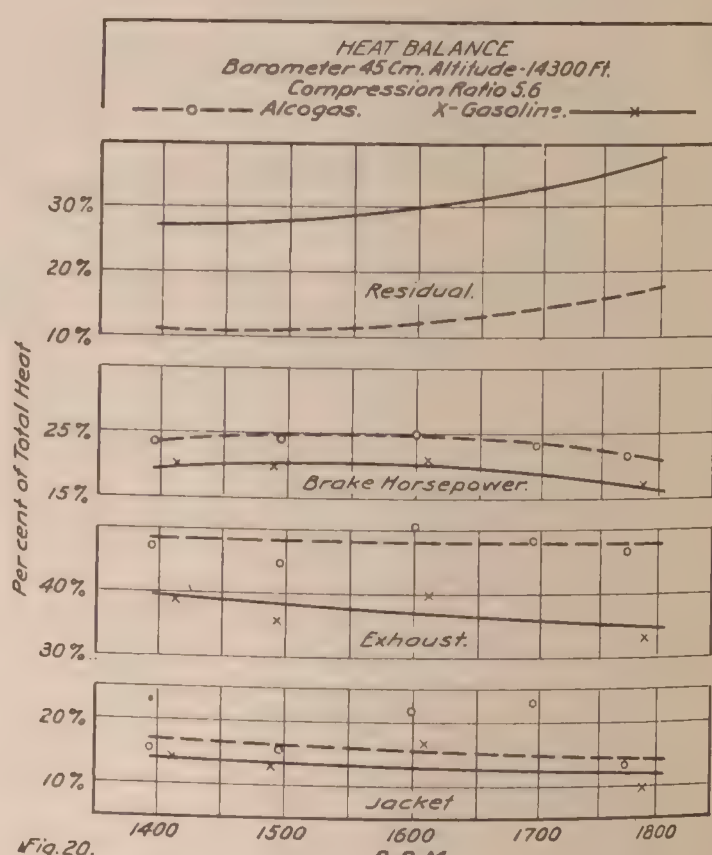


Fig. 20.

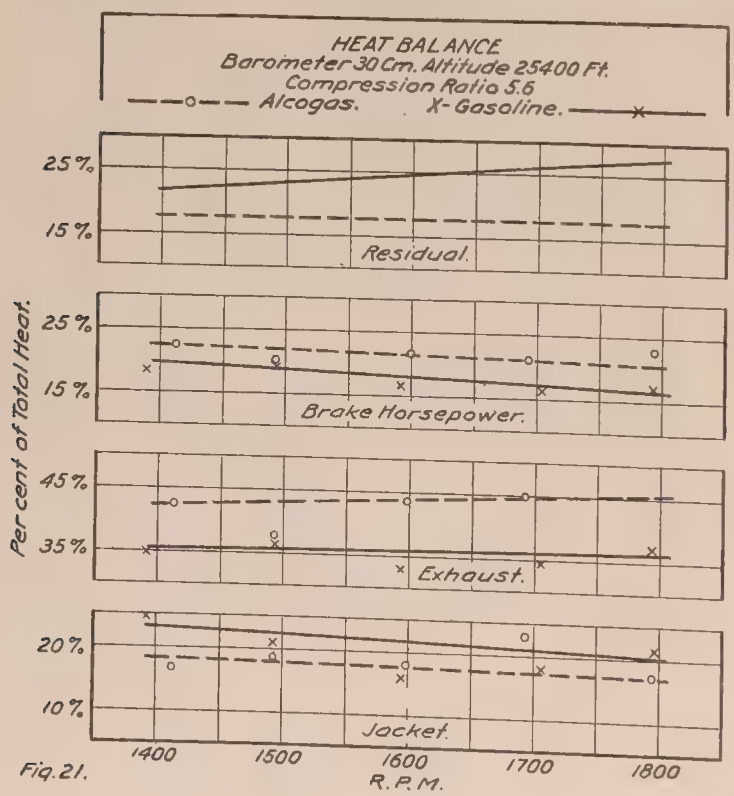


Fig. 21.

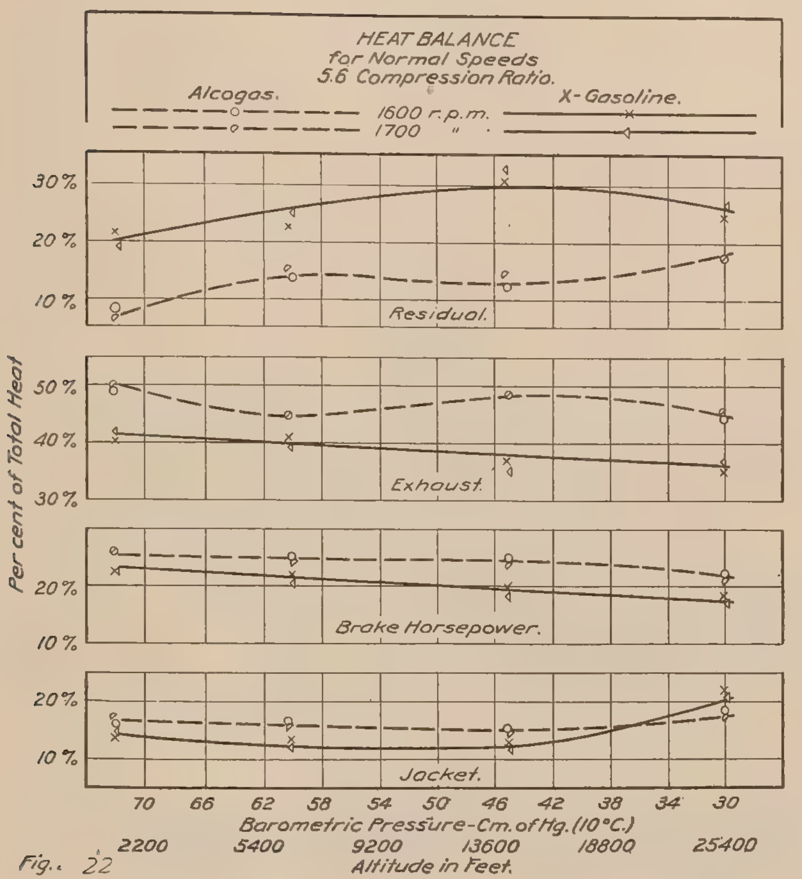


Fig. 22

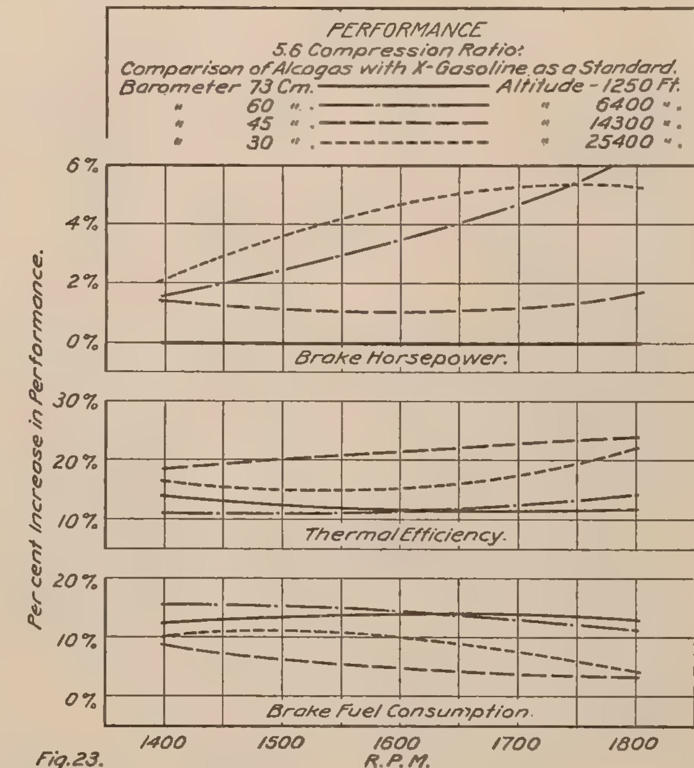


Fig. 23.

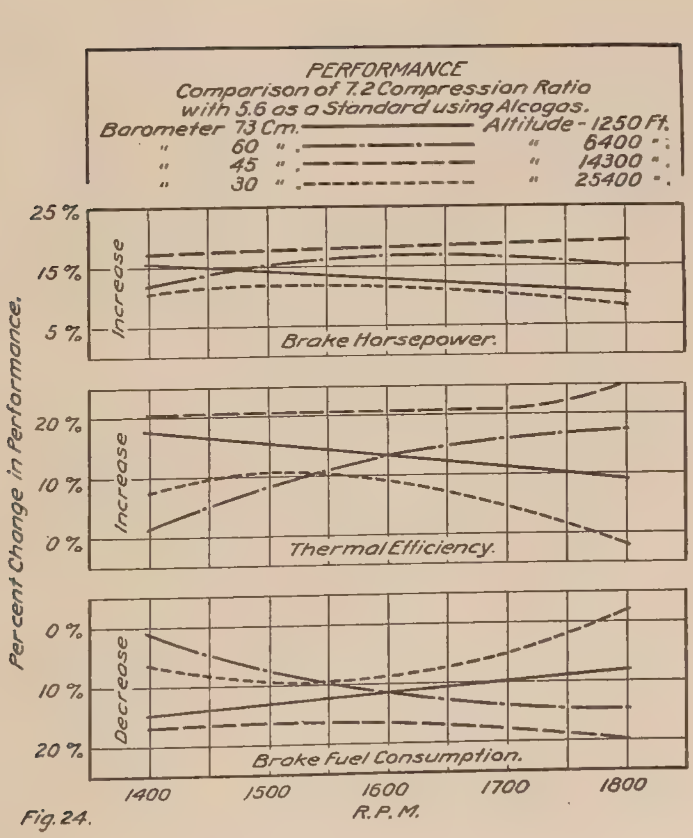


Fig. 24.

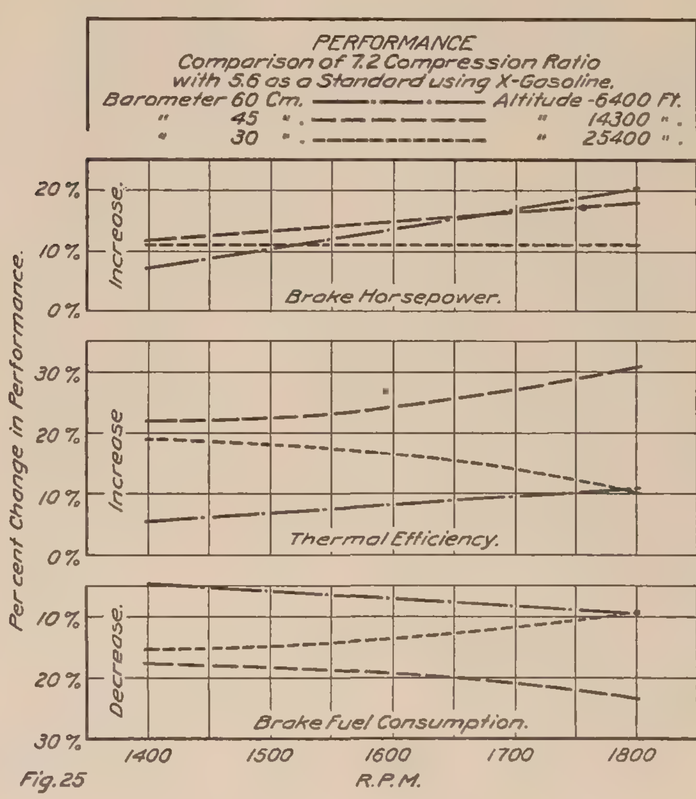


Fig. 25

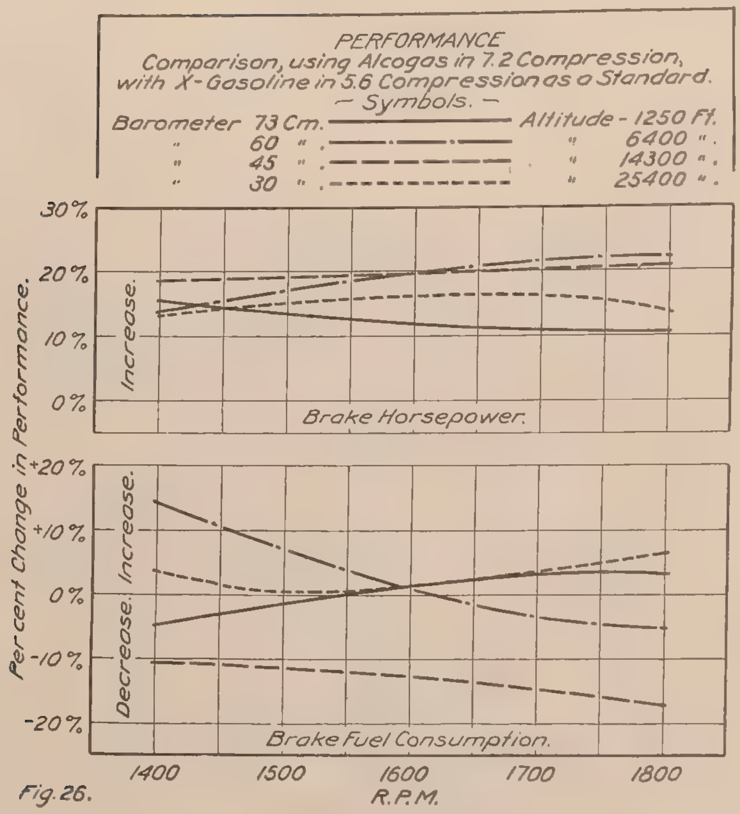

PERFORMANCE
Comparison, using Alcogas in 7.2 Compression, with X-Gasoline in 5.6 Compression as a Standard.
— Symbols. —
Barometer 73 Cm. Altitude - 1250 Ft.
" 60 " " 6400 "
" 45 " " 14300 "
" 30 " " 25400 "
30%
20%
10%
0%
Increase.
Brake Horsepower.
+20%
+10%
0%
-10%
-20%
Decrease. Increase.
Brake Fuel Consumption.
Percent Change in Performance.
1400
1500
1600
1700
1800
R.P.M.

Fig. 26.

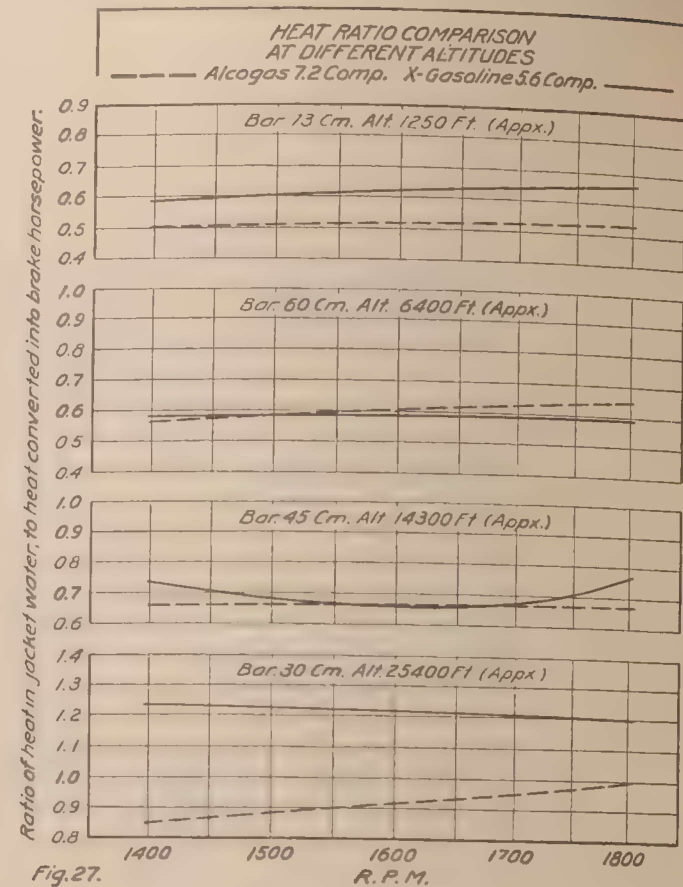

HEAT RATIO COMPARISON AT DIFFERENT ALTITUDES
Alcogas 7.2 Comp. X-Gasoline 5.6 Comp.
Bar. 73 Cm. Alt. 1250 Ft. (Appx.)
Bar. 60 Cm. Alt. 6400 Ft. (Appx.)
Bar. 45 Cm. Alt. 14300 Ft. (Appx.)
Bar. 30 Cm. Alt. 25400 Ft. (Appx.)
Ratio of heat in jacket water, to heat converted into brake horsepower.
1400
1500
1600
1700
1800
R.P.M.

Fig. 27.

REPORT No. 90

COMPARISON OF HECTER FUEL WITH EXPORT AVIATION GASOLINE

By H. C. DICKINSON, V. R. GAGE, AND S. W. SPARROW
Bureau of Standards

REPORT No. 90.

COMPARISON OF HECTER FUEL WITH EXPORT AVIATION GASOLINE.

By H. C. Dickinson, V. R. Gage, and S. W. Sparrow,

Bureau of Standards.

RÉSUMÉ.

This report was prepared for the National Advisory Committee for Aeronautics and describes an investigation conducted at the altitude laboratory at the Bureau of Standards.

Aviation engine developments for attaining higher power at altitude are following two principal lines, supercharging and increase in compression ratio. For the latter, fuels have been demanded which are capable of operating under compressions too high for gasoline. Among the fuels which will operate at compression ratios up to at least 8.0 without preignition or "pinking" is Hecter fuel, whence a careful determination of its performance is of importance.

A comprehensive investigation by the United States Bureau of Mines of fuels for internal-combustion engines included the development of cyclohexane mixtures to ascertain definitely whether they possessed the marked advantages attributed to them in rumors from abroad. The Bureau of Mines, cooperating with others, developed hydrogenation of benzol to cyclohexane, testing mixtures of the two in various types of engines at compression ratios ranging from 5.3 to 8.2. The cyclohexane benzol mixtures, the former constituent predominating, were designated "Hecter" fuels. They gave very promising results at high compression in the experimental engines at ground level, and their usability at altitude was tested by actual flight tests. Accordingly, data were desired regarding power development and economy at altitude, data not readily obtainable under the varying conditions of actual flight. The fuel was submitted to the Bureau of Standards for test in the altitude laboratory.

The Hecter fuel supplied by the Bureau of Mines for use in these tests was a mixture of 30 per cent benzol (C_6H_6) and 70 per cent cyclohexane (C_6H_{12}), having a low freezing point, and distilling from first drop to 90 per cent at nearly a constant temperature, about 20° C. below the average distillation temperature ("mean volatility") of the X gasoline.

This comparison of the performance of the two fuels in an aviation engine was made in the altitude chamber at the Bureau of Standards, duplicating altitude conditions up to about 25,000 feet, except that the temperature of the air entering the carburetor was maintained nearly constant at about 10° C. A Liberty 12-cylinder aviation engine was used, supplied with special pistons giving a compression ratio of 7.2 (the compression pressure measured by check valve gage was 170 pounds per square inch). Stromberg carburetors were used and were adjusted for each change of fuel, speed, load, and altitude so as to give the maximum possible power with the least fuel for this power. The tests covered a speed range of 1,400 to 1,800 r. p. m.

The results of these experiments show that the power developed by Hecter fuel is the same as that developed by Export aviation gasoline, at about 1,800 r. p. m. at all altitudes. At lower speeds differences in the power developed by the fuels become evident. At 1,400 r. p. m. and 25,000 feet Hecter gives a little less power than X gasoline, at 15,000 feet about the same, and at 6,000 feet perhaps 6 per cent more. Comparisons at ground level were omitted to avoid any possibility of damaging the engine by operating with open throttle on gasoline at so high a compression. The fuel consumption per unit power based on weight, not volume, averaged more than 10 per cent greater with Hecter than with X gasoline, considering all conditions. The thermal efficiency of the engine when using Hecter is less than when using gasoline, particularly at the higher speeds, a generalization of the difference for all altitudes and speeds being

8 per cent. The general deduction from these facts is that more Hecter is exhausted unburnt. Undoubtedly Hecter can withstand high compression pressures and temperatures without preignition. This characteristic was proved by operating the engine (compression ratio 7.2) with full throttle at 1,500 r. p. m. on the ground, carburetor air temperature 42° C. (107.6° F.) and jacket-water temperature, leaving engine, at 90° C. (194° F.). No signs of preignition or "pinking" were noted.

The engine was not operated for a sufficient period to compare the compression ratios or the fuels as regards effects upon engine deterioration.

It is of interest to compare the engine performance using X gasoline in a 5.6 compression engine and Hecter in a 7.2 compression engine of the same type. Previous tests with a similar engine, using one fuel, show that a change of compression ratio from 5.6 to 7.2 results in about 10 per cent increase in power. This indicates that Hecter in a 7.2 compression would produce 10 per cent more power for the same weight fuel consumption per unit power than would X gasoline in a 5.6 compression.

OBJECT OF TESTS.

The object of the tests and the subject of this report is the comparison of Hecter fuel with X gasoline, with regard to the relative power-producing qualities and fuel consumptions of the two fuels when used in an extremely high compression aviation engine (7.2 compression ratio).

DESCRIPTION OF THE FUELS.

The gasoline used in these tests was the standard reference fuel of this laboratory (known as "X" gasoline), prepared for the Bureau of Standards by the Atlantic Refining Co. from Pennsylvania crude oil. It complies with the specification No. 3512 of the Bureau of Aircraft Production, for export aviation gasoline for the use of the A. E. F., 1918. The heating value (higher) is 11,300 calories per gram (20,340 B. t. u. per pound). The Hecter fuel supplied through the Bureau of Mines for these tests was a mixture of approximately 30 per cent benzol, 70 per cent cyclohexane by volume. The freezing point was about −32° C. (−25° F.) and the heat of combustion (higher) was 10,800 calories per gram (19,440 B. t. u. per pound), about 4.5 per cent less than that of X gasoline.

The distillation curves and other properties of the two fuels are given on figure 1. The values for the distillation curves are also given in Table I.

TABLE I.—*Distillation.*

	Hecter fuel.	X gasoline.
	°*C.*	°*C.*
Initial boiling point	76	59
10 per cent	77	72
20 per cent	77	77
30 per cent	77	82
40 per cent	78	87
50 per cent	78	92
60 per cent	78	97
70 per cent	78	103
80 per cent	78	111
90 per cent	78	127
95 per cent	79	150
Dry point	[1] 82	[2] 153

[1] 98 per cent. [2] 96 per cent.

DESCRIPTION OF APPARATUS.

A Liberty 12-cylinder airplane engine was used for these tests, manufacturer's No. 586, Aircraft Production No. 30641. Mobile "B" oil was used for lubrication. The equipment was standard except for the high-compression pistons and the Stromberg carbureters. The clearance volumes were measured by filling them with oil, and were found to give a compression ratio of 7.2. The compression pressure as measured by gage and check valve was 170 pounds per square inch. The carbureters were equipped with a manual adjustment of float chamber pressure so that the mixture ratio could be changed as desired. This adjustment was ample at all altitudes and speeds, so that it was always possible to make the mixture too lean.

The engine was mounted in the altitude chamber of the Bureau of Standards automotive power plants laboratory. The air from the chamber can be exhausted so that the pressure will correspond to that at any altitude up to 30,000 feet. By means of refrigerating coils and heaters, the temperature of the air in the chamber and of that supplied to the carbureter can be changed through a very wide range. All controls, adjustments, and measuring instruments (including dynamometer) are outside the chamber. A complete description of this equipment can be found in Report No. 44 of the National Advisory Committee for Aeronautics.

TEST PROCEDURE.

A complete description of the standard method of test procedure for this laboratory is in preparation. Accordingly a brief treatment of the subject will suffice here. In examining curves and the tables of data attached to this report the reader should bear in mind that many of the measurements made in connection with these tests are for use in further analyses not connected with the fuel comparison, which is the subject of this report. Only the features which have direct bearing on the fuel comparison need be considered here.

Two fuel tanks were used, each mounted on a balance; one containing the X gasoline, the other Hecter. The engine was started on X gasoline and the desired conditions of speed and altitude reached with a comparatively rich mixture. The maximum dynamometer (engine) torque having been attained, observations of torque were continued while the rate of gasoline supply was gradually reduced. The leaning of the mixture was continued until the torque fell off considerably, then the mixture was very gradually enriched again, but only enough to secure a torque equal to the maximum which had previously been noted. Readings were then taken of the various temperatures, pressures, torque, rates of flow, speed, etc. The fuel supply from the X tank was cut off, and Hecter was supplied to the carbureter. The carbureter adjustment was again made for maximum torque at the least possible expenditure of fuel, as described for X, and readings of test data again were made. By changing from one fuel to another in this manner, it is possible to eliminate, to a great extent, the relative effect upon the comparison of the fuels of any changes in the condition of the engine. By adjusting the carbureter for each fuel at each change of load, speed, or altitude, it is possible to obtain the engine characteristics, independent of the carbureter characteristics, and also to obtain information as to what the desired carbureter action should be. This knowledge of how the carbureter should perform is highly essential as generally the engine is hampered by poor carbureter characteristics.

DISCUSSION OF CURVES.

METHODS OF COMPUTATION, CURVE DRAWING, AND OF REDUCING TO STANDARD CONDITIONS.

The dynamometer torque as observed was reduced to brake mean effective pressure by means of a multiplication constant. These values were plotted versus r. p. m., figure 2. On the ground run it will be noted that brake power and mean effective pressure have both been corrected for exhaust pressure. The corrected points are those marked by triangles. Normally the exhaust pressure is kept near enough to carbureter air pressure so that no correction is required. Many considerations aid in determining the relative value of the actual points from the data. These are to be found in the notes on the original data sheets regarding steadiness of conditions during the run, difficulties in determining correct settings, apparent ignition troubles, etc. Also the various measurements of pressure and temperatures throughout the induction system, not bearing directly upon the fuel comparison, are of great value in determining the probable location of the curves. The curve for 1,250 feet on figure 6 may be cited as an illustration. The points which have been neglected in locating the curve were those where the manifold suction was found to be abnormal.

The curves of brake horsepower versus speed (fig. 3) are drawn through values computed from the faired curves of figure 2, because on the curves as drawn the mean effective pressure (torque) values give more nearly a straight line relation than does the brake horsepower. However, the points shown on figure 3 are computed directly from the test data. A detailed exposition of the analyses of the test data and notes would be required to make more clear the reasons

for locating the curves of figures 2 and 3 as drawn instead of passing them more nearly through the apparent average of the points. As a check on the faired curves of figures 2 and 3 horsepower values were read, from the curves for constant speeds, and then were plotted against the third variable air pressure (figs. 4 and 5.) This relation should be nearly linear. It appears that the slope of the H. P. barometer curves is greater with increased speed. This tendency may be attributed to the effect of increased friction H. P. at higher speeds, but there are so many factors entering into the friction losses that it is well to defer discussion until analysis may be made of many tests.

FUEL CONSUMPTION.

The test results of weight of fuel consumed per hour are plotted versus r. p. m. on figure 6, a curve for each barometric pressure. These are used to assist in judging the results of the tests. On figure 7 are plotted pounds of fuel per brake horsepower per hour, versus r. p. m. On figure 8, the pounds fuel per brake horsepower per hour versus barometric pressure are obtained from the faired curves of figure 7. Even under the most favorable conditions, considerable change in mixture is possible for a slight change in power, so that very high accuracy is impossible in duplicating the condition of maximum power with minimum fuel consumption. This is the reason for the scattering of the points on figures 6 and 7, rather than lack of precision in fuel measurements. Had the carbureter setting been left unchanged for the two fuels, the results would have been more consistent, but of no value as a measure of their power-producing ability. It is probable that different fuels require different air to fuel ratios and it is by no means certain that the same carbureter setting will give the same air to fuel ratio with two different fuels. Likewise, it would have been possible to secure more consistent results if a definite and fixed carbureter setting had been used for each fuel. But, by doing this, the carbureter characteristics would have been superposed upon the engine characteristics. In these tests it was desired to know what was the best the engine could do, independent of the kind of carbureter used, and also to find what carbureter characteristics gave the best performance with each fuel.

The relative "pulverization" of fuels is dependent partly on surface tension, or cohesion, partly on the form of carbureting device, and partly on the temperatures, pressures and time available for vaporization. These factors and others are to be considered in studying figure 8. Here the fuel consumption (per unit power) of X gasoline seems to reach a minimum at about 50 centimeters barometer (13,000 feet). This tendency has been noted at other times with other set-ups, and it remains to be studied.

The heat distribution is presented in the form of curves in figures 9, 10, and 11, percentage of heat supplied versus r. p. m. The points shown are the original test results of per cent heat appearing in brake horsepower, exhaust, and jacket. The curves of "residual" heat are the differences between 100 per cent and the sum of the above three. The residual heat as computed here, therefore, includes the unburnt fuel in the exhaust and the so-called radiation losses, less the heat supplied by combustion of lubricating oil. The heat supplied is computed from the total or higher heating value of the fuel, and the exhaust heat is measured by "exhaust calorimeter" methods. The residual heat when using Hecter is always more than when using X gasoline, the exhaust and jacket losses, and the brake thermal efficiency are always less. The interpretation is that less of the heat energy of Hecter is liberated in the cylinder and more of the fuel is exhausted unburnt. These curves should not be construed as showing the exact quantative effect of speed changes alone upon heat distribution, being considerably influenced by the carburetor adjustment.

The computed values of heat distribution for two normal speeds (1,600 and 1,700 r. p. m.), are the points plotted on figure 12, as per cent of heat supplied by the fuel versus barometric pressure. Curves were drawn through these points, and the per cent residual heat was derived from the other curves. Again, this plot should be interpreted more as heat distribution at various altitudes than as the exact quantative effect of altitude on the distribution. The reverse curvature of the exhaust and residual lines, indicating a more complete burning of

X gasoline at 12,000 feet, is a tendency noted on other tests, and which will require further study.

Figure 13 is derived from the preceding curves, and presents the net results of the first comparison in graphical form.

For the sake of clearness the scale of per cent is made open, so that differences of little magnitude (2 per cent) may give an impression of a gain or loss which is in reality a probable equality.

CONCLUSIONS.

(1) For flight at low altitudes Hecter fuel showed slight advantages in comparison with gasoline by affording a small increase of power over and above that necessary to offset the disadvantage of increased fuel consumption. The usual ratio of fuel weight to plane weight is of the order of 1 to 7 so that for full throttle flying an increased fuel consumption of 7 per cent balances an increase of 1 per cent in power developed. The test at 6,500 feet altitude howed that Hecter fuel developed slightly more power than X gasoline, the maximum advantage being 7 per cent and the average for all speeds 4 per cent, whereas the increase in suel consumption averaged 5 per cent or 6 per cent. Since at 14,000 feet and 25,000 feet no appreciable difference in power was obtained, whereas the fuel consumption of Hecter was greater to the extent of 15 per cent by weight, the advantage lies with X gasoline.

(2) The large difference in densities of Hecter fuel and X gasoline makes the fuel comparisons by weight and by volume read quite differently, and care must be exercised to distinguish them. Upon reducing pounds per brake horsepower hour to pints per brake horsepower hour it is found that Hecter consumption is the less by volume at ground, and about equal to that of X gasoline at 25,000 feet.

(3) One gallon of Hecter contains nearly 9 per cent more heat units than a gallon of X gasoline, and the brake thermal efficiency of this engine using Hecter is about the same per cent less than when using X gasoline. Thus the same tank full of either fuel would supply a plane with about the same available energy. Any part of a flight at very low altitude might be accomplished at slightly higher plane speed with the Hecter than with gasoline, as a consequence of the power characteristics described above.

(4) It has been claimed that a high-compression engine has a greater factor of safety when operated with Hecter fuel than with gasoline. The engine was not operated for a sufficient period of time to ascertain whether engine deterioration was more rapid with the 7.2 compression ratio than would be expected from experience with the 5.6 compression ratio. Consequently no comparison can be made of the effect of compression or fuels upon engine deterioration.

(5) However, since it is not generally considered advisable to operate an engine of this type with gasoline at a higher compression ratio than 5.6, it is of interest to compare the performance of a Liberty 12-cylinder aviation engine of 5.6 compression ratio using gasoline with the performance of the same type of engine with 7.2 compression ratio using Hecter. Previous tests with this type of engine have shown that this change in compression produces about 10 per cent increase in power with about the same percentage decrease in weight of fuel consumed per unit power. This change would be expected from a comparison of the "air standard" efficiencies. From these data it is concluded that Hecter in a 7.2 compression ratio engine would produce about 10 per cent more power than would X gasoline in a 5.6 compression ratio, while using the same weight of fuel per unit power as for X gasoline in the lower compression.

JANUARY 29, 1920.

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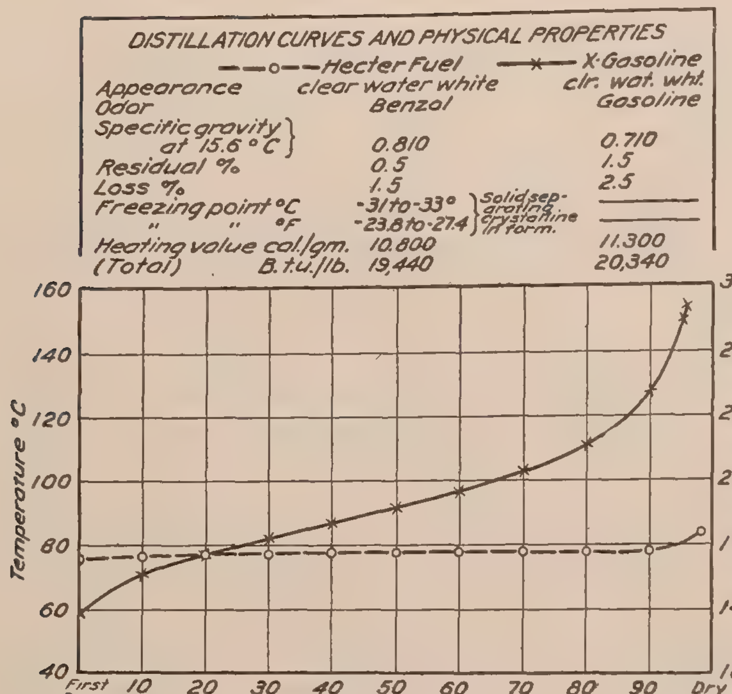


Fig. 1.

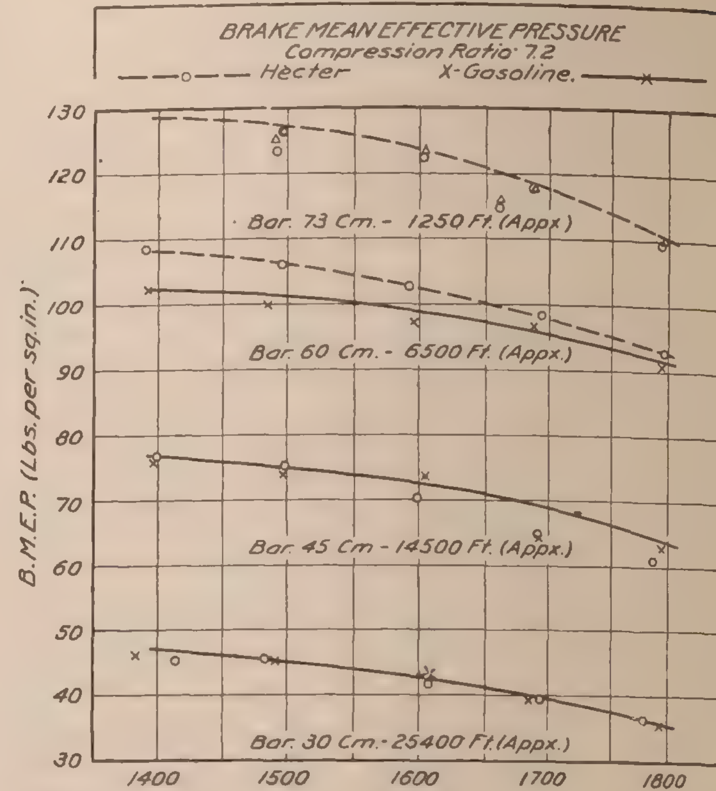
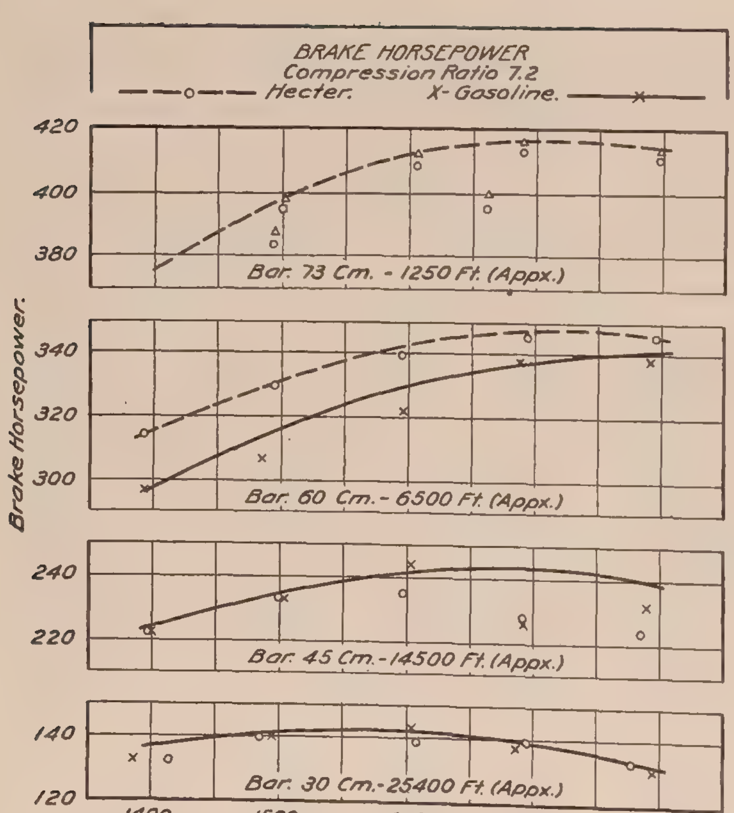


Fig. 2.



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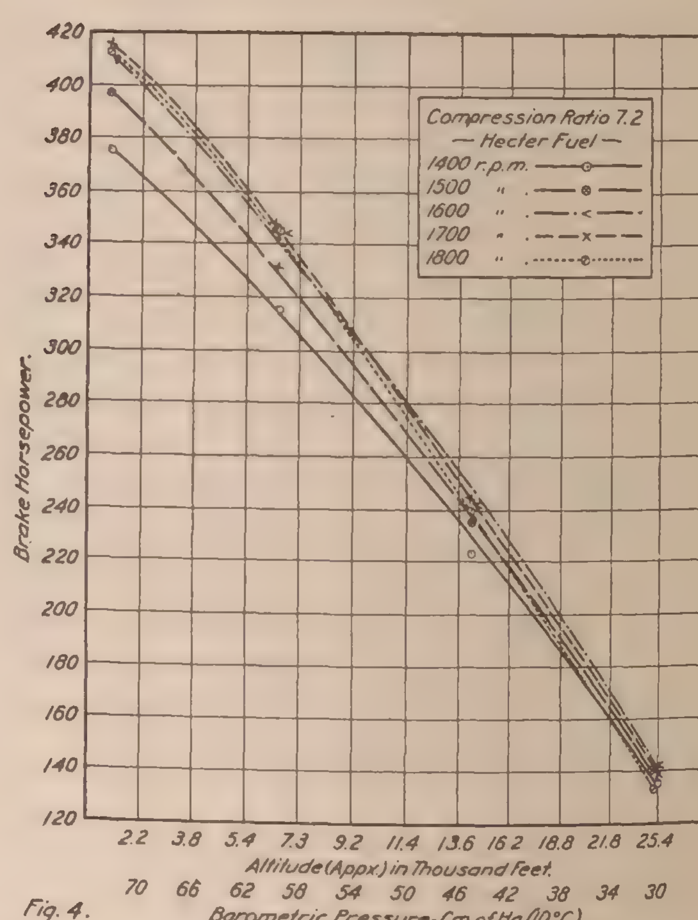

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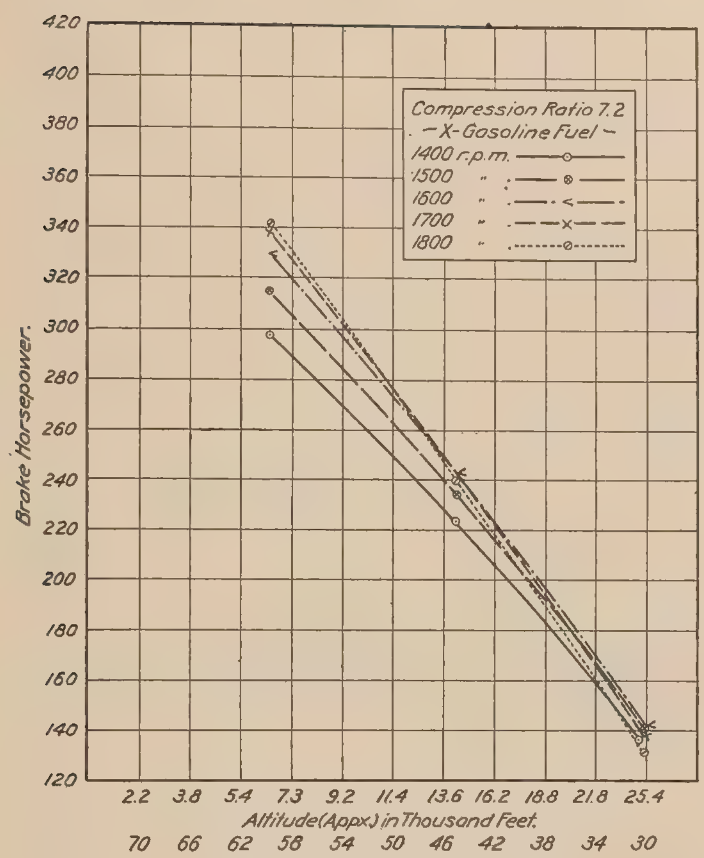


Fig. 5.

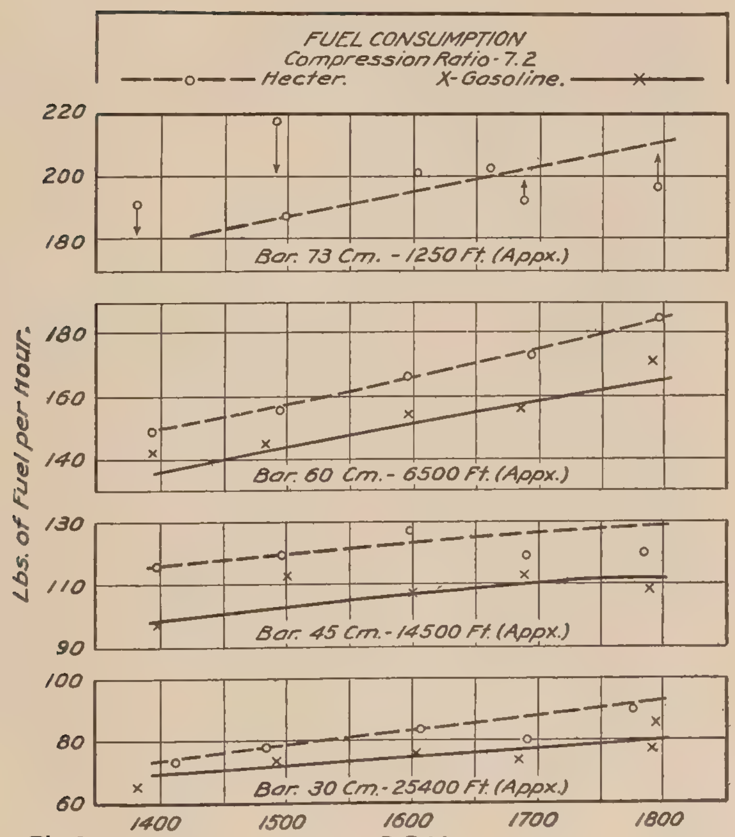


Fig. 6.

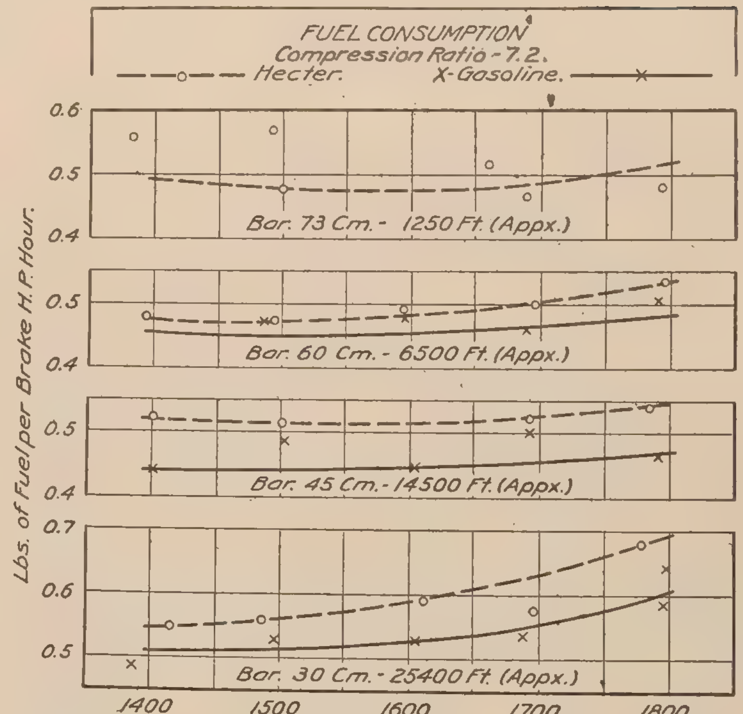


Fig. 7.

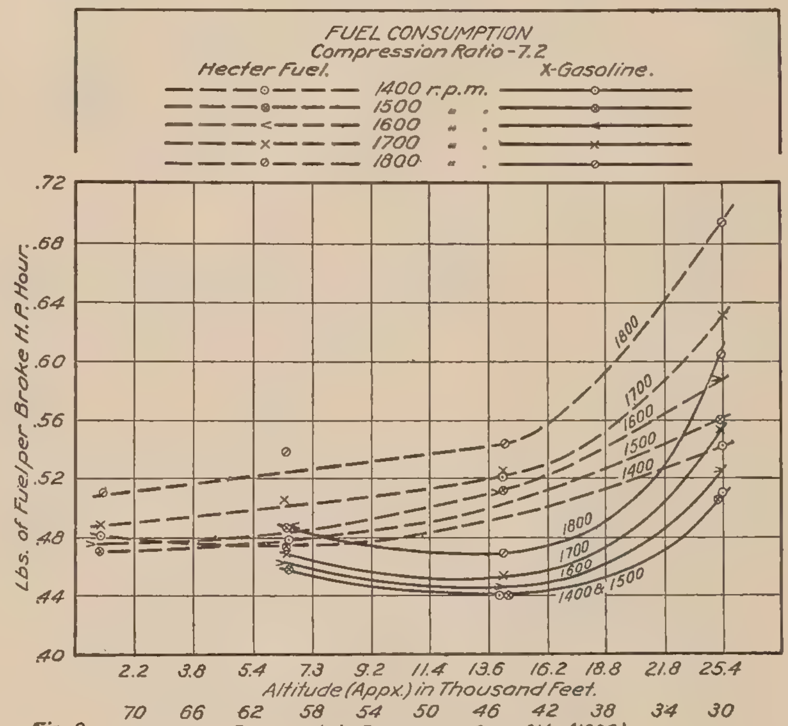


Fig. 8.

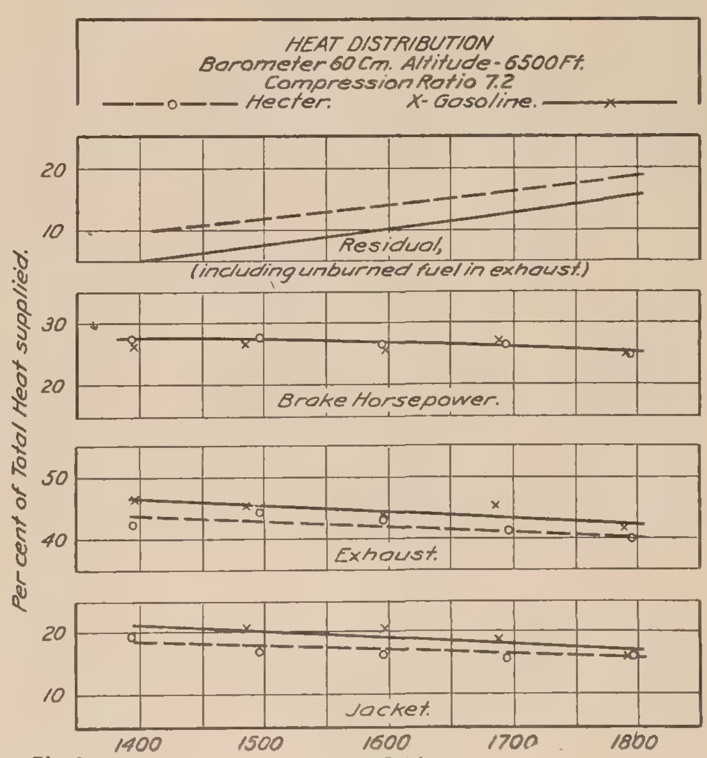


Fig. 9.

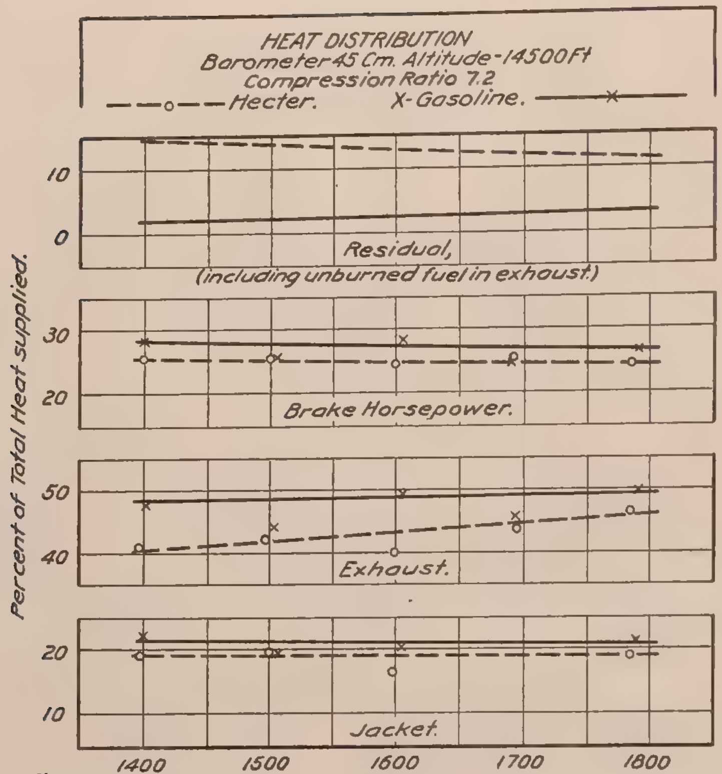
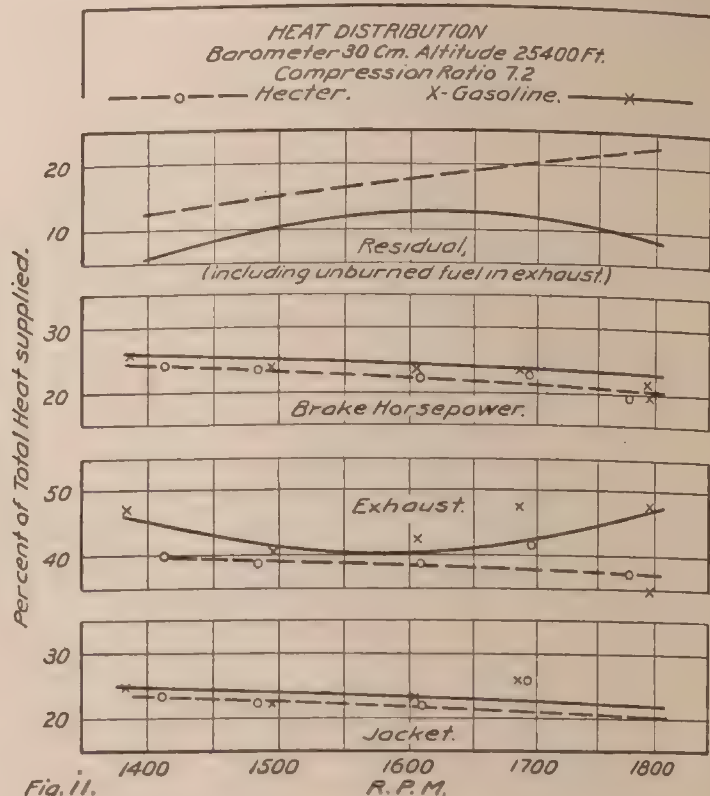


Fig. 10.



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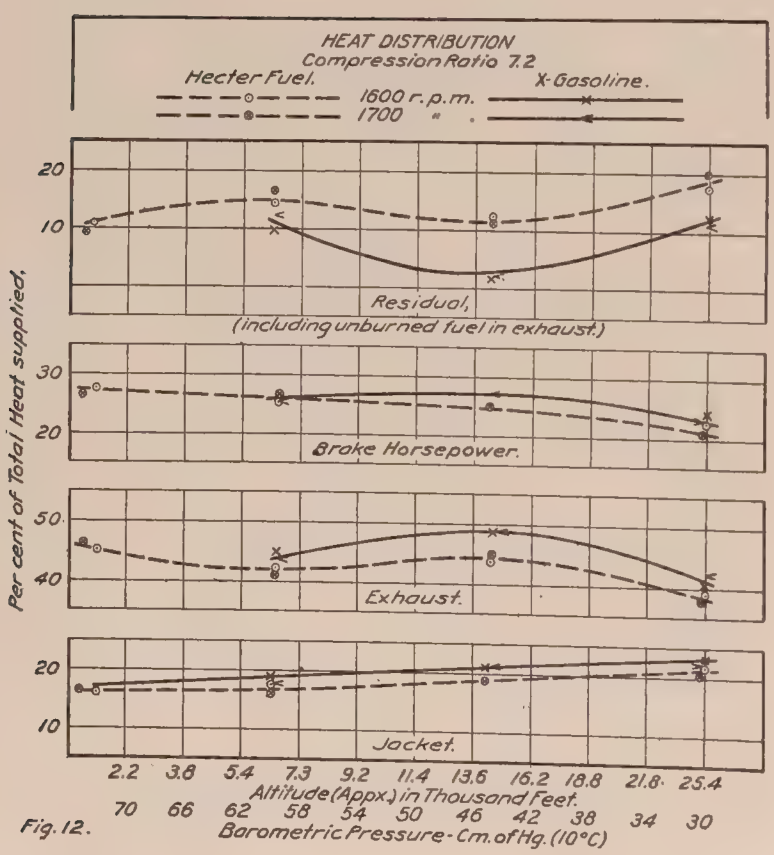

Fig. 12.

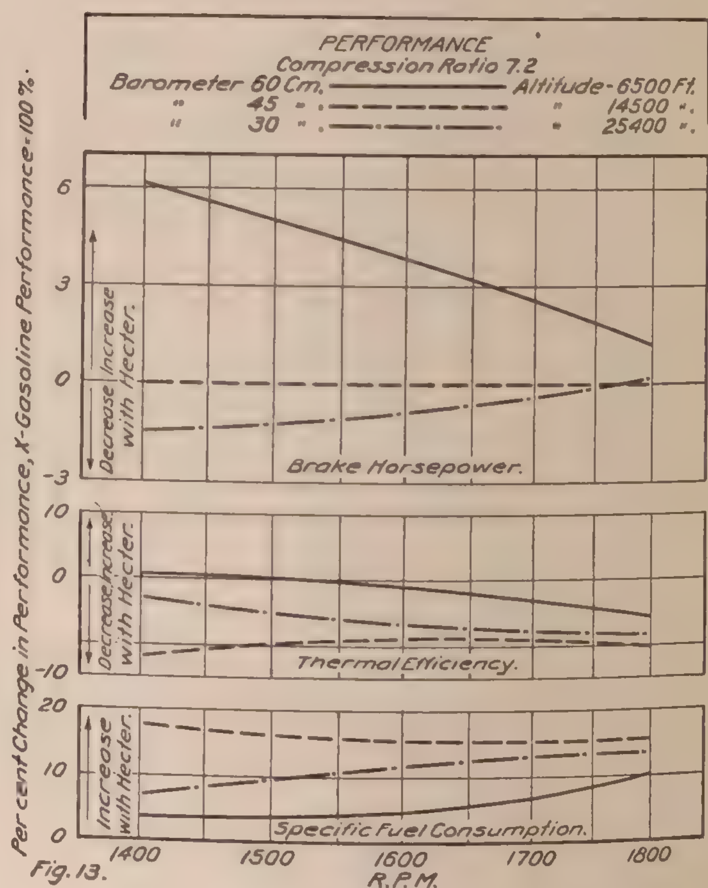

Fig. 13.

REPORT No. 91

NOMENCLATURE FOR AERONAUTICS

IN THREE PARTS

BY NATIONAL ADVISORY COMMITTEE
FOR AERONAUTICS

INTRODUCTION.

The following nomenclature and list of symbols were approved by the Executive Committee of the National Advisory Committee for Aeronautics, for publication as a technical report on April 1, 1920, on recommendation of the Subcommittee on Aerodynamics.

The purpose of the committee in the preparation and publication of this report is to secure uniformity in the official documents of the Government and, as far as possible, in technical and other commercial publications. This report supersedes all previous publications of the committee on this subject.

The Subcommittee on Aerodynamics had charge of the preparation of the report. It was materially assisted by an Interdepartmental Conference on Aeronautical Nomenclature and Symbols, organized by the Executive Committee, with the approval of the War and Navy Departments, for the purpose of giving adequate representation to the divisions of the Army Air Service and to the bureaus of the Navy Department most concerned. The first meeting of the interdepartmental conference was held on October 23, 1919, and the second meeting on January 15, 1920, at which meeting this report was unanimously approved and recommended to the Subcommittee on Aerodynamics, with the reservation that stability terms and power-plant terms be given further and special consideration.

The stability terms were accordingly referred for special consideration to Messrs. E. B. Wilson, J. C. Hunsaker, A. F. Zahm, E. P. Warner, and H. Bateman, and the power-plant terms were referred to the Subcommittee on Power plants for Aircraft. The complete report was adopted by the Subcommittee on Aerodynamics on March 8, 1920, and recommended to the Executive Committee for approval and publication. The personnel of the two organizations primarily concerned with the preparation of this report follows.

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Col. T. H. Bane, United States Army.
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Dr. A. F. Zahm.
Dr. L. J. Briggs.
Dr. M. D. Hersey.

INTERDEPARTMENTAL CONFERENCE ON AERONAUTICAL NOMENCLATURE AND SYMBOLS.

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS.

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Dr. M. D. Hersey.

REPORT No. 91.

NOMENCLATURE FOR AERONAUTICS.

By the NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS.

PART I.

NOMENCLATURE BY DIVISIONS.

AIRCRAFT.

A. TYPES OF AIRCRAFT.

AEROSTAT.—An aircraft which embodies a container filled with a gas lighter than air and which is sustained by the buoyancy of this gas; e. g., airship, balloon.

AIRCRAFT.—Any form of craft designed for the navigation of the air—airplanes, airships, balloons, helicopters, kites, kite balloons, ornithopters, gliders, etc.

AIRPLANE.—A form of aircraft heavier than air which obtains support by the dynamic reaction of the air against the wings and which is driven through the air by a screw propeller. This term is commonly used in a more restricted sense to refer to airplanes fitted with landing gear suited to operation from the land. If the landing gear is suited to operation from the water, the term "seaplane" is used. (See definition.)

AIRSHIP.—A form of aerostat provided with a propelling system and with means of controlling the direction of movement.

BALLOON.—A form of aerostat deriving its support in the air from the bouyancy of the air displaced by an envelope, the form of which is maintained by the pressure of a contained gas lighter than air, and having no power plant or means of controlling the direction of flight in the horizontal plane.

GLIDER.—A form of aircraft similar to an airplane, but without any power plant. Gliders are used chiefly for sport.

HELICOPTER.—A form of aircraft whose support in the air is derived from the, vertical thrust of propellers.

KITE.—A form of aircraft without other propelling means than the towline pull, whose support is derived from the force of the wind moving past its surface.

ORNITHOPTER.—A form of aircraft deriving its support and propelling force from flapping wings.

PARACHUTE.—An apparatus used to retard the descent of a falling body by offering resistance to motion through the air; usually made of light fabric with no rigid parts.

AEROSTATS.

B. TYPES OF AEROSTATS.

AIRSHIP:

NONRIGID.—An airship whose form is maintained by the pressure of the contained gas.

RIGID.—An airship whose form is maintained by a rigid structure contained within the envelope.

SEMIRIGID.—An airship whose form is maintained by means of a rigid or jointed keel and by gas pressure.

BALLOON:

BARRAGE.—A small captive balloon, raised as a protection against attacks by airplanes.

CAPTIVE.—A balloon restrained from free flight by means of a cable attaching it to the earth.

B. TYPES OF AREOSTATS—Continued.

BALLOON—Continued.

KITE.—An elongated form of captive balloon, fitted with tail appendages to keep it headed into the wind, and usually deriving increased lift due to its axis being inclined to the wind. A Caquot balloon is of this type. (Fig. 8.)

NURSE.—A small balloon made of heavy fabric, employed as a portable means for storing gas. Sometimes one is so connected as to automatically allow for the expansion or contraction of the gas in an aerostat when on the ground.

PILOT.—A small balloon sent up to show the direction of the wind by observations of its flight with theodolites.

SOUNDING.—A small balloon sent aloft without passengers but with registering meteorological and other instruments.

AIRPLANES.

C. TYPES OF AIRPLANES.

AIRPLANE:

PUSHER.—A term commonly applied to a single-engined airplane with the propeller in the rear of the main supporting surfaces. (Fig. 3.)

TRACTOR.—A term commonly applied to a single-engined airplane with the propeller forward of the main supporting surfaces. (Fig. 4.)

BIPLANES.—A form of airplane whose main supporting surface is divided into two parts, superimposed.

MONOPLANE.—A form of airplane which has but one main supporting surface extending equally on each side of the body.

MULTIPLANE.—A form of airplane whose main supporting surface is divided into more than four parts.

QUADRUPLANE.—A form of airplane whose main supporting surface is divided into four parts, superimposed.

SEAPLANE.—A particular form of airplane designed to rise from and land on the water.

(a) BOAT SEAPLANE (or flying boat).—A form of seaplane having for its central portion a boat which provides flotation. It is often provided with auxiliary floats or pontoons. (Fig. 14.)

(b) FLOAT SEAPLANE.—A form of seaplane in which the landing gear consists of one or more floats or pontoons. (Fig. 15.)

TANDEM.—An airplane with two or more sets of wings of substantially the same area (not including the empennage) placed one in front of the other and on about the same level.

TRIPLANE.—A form of airplane whose main supporting surface is divided into three parts, superimposed.

GENERAL AERODYNAMICS.

D. MISCELLANEOUS TERMS.

AEROFOIL.—A winglike structure, flat or curved, designed to obtain reaction upon its surfaces from the air through which it moves.

AEROFOIL SECTION.—A section of an aerofoil made by a plane parallel to the plane of symmetry of the aerofoil and to the normal direction of motion.

ASPECT RATIO.—The ratio of span to mean chord of an aerofoil.

ATTACK, ANGLE OF.—The acute angle between the direction of the relative wind and the chord of an aerofoil; i. e., the angle between the chord of an aerofoil and its motion relative to the air. (This definition may be extended to any body having an axis.)

D. MISCELLANEOUS TERMS—Continued.

CAMBER.—The convexity or rise of the curve of an aerofoil from its chord, usually expressed as the ratio of the maximum departure of the curve from the chord to the length of the chord. "Top camber" refers to the top surface of an aerofoil, and "bottom camber" to the bottom surface; "mean camber" is the mean of these two.

CENTER OF PRESSURE OF AN AEROFOIL SECTION.—The point in the chord of an aerofoil section, prolonged if necessary, through which at any given attitude the line of action of the resultant air force passes.

CHORD (of an Aerofoil Section).—The line of a straightedge brought into contact with the lower surface of the section at two points. In the case of an aerofoil having double convex camber, the straight line joining the leading and trailing edges. (These edges may be defined, for this purpose, as the two points in the section which are farthest apart.) (Fig. 9.)

Length.—The length of the projection of the aerofoil section on its chord.

CRITICAL ANGLE.—The angle of attack at which the flow about an aerofoil changes abruptly, with corresponding abrupt changes in the lift and drag coefficients. An aerofoil may have two or more critical angles, one of which almost always corresponds to the angle of maximum lift.

LEADING EDGE.—The foremost edge of an aerofoil or propeller blade.

SKIN FRICTION.—The tangential component of the fluid force at a point on a surface. It depends on the viscosity and density of the fluid, the total surface area, and the roughness of the surface of the object.

STABILIZER, MECHANICAL.—A mechanical device to stabilize the motion of an aircraft. Includes gyroscopic stabilizers, pendulum stabilizers, inertia stabilizers, etc.

STREAMLINE.—The path of a small portion of a fluid, supposed continuous, commonly taken relative to a solid body with respect to which the fluid is moving. The term is commonly used only of such paths as are not eddying, but the distinction should be made clear by the context.

STREAMLINE FLOW.—The condition of continuous flow of a fluid, as distinguished from eddying flow.

STREAMLINE FORM.—A fair form intended to avoid eddying and to preserve streamline flow.

SURFACE.—An aerofoil used for sustentation or control or to increase stability. Applies to the whole member, and not to one side only.

Balanced.—A surface, such as a rudder, aileron, etc., part of which is in front of its pivot.

TRAILING EDGE.—The rearmost edge of an aerofoil or propeller blade.

WIND TUNNEL.—An elongated inclosed chamber, including means for the production of a substantially steady air current through the chamber. Models of aircraft or other objects are supported in the center of the airstream and their resistance and other characteristics when exposed to an air current of known velocity are determined. The term includes those laboratories in which, as in the Eiffel type, there is an experimental chamber of much larger cross section than the air current.

ZERO LIFT ANGLE.—The angle between the chord and the relative wind when the lift is zero.

ZERO LIFT LINE.—The position in the plane of an aerofoil section of the line of action of the resultant air force when the position of the section is such that the lift is zero.

E. FORCES, MOMENTS, ANGLES, AND AXES.

ATTITUDE.—The attitude of an aircraft is determined by the inclination of its axes to a "frame of reference", fixed to the earth; i. e., the attitude depends entirely on the position of the aircraft as seen by an observer on the ground.

AXES OF AN AIRCRAFT.—Three fixed lines of reference; usually centroidal and mutually rectangular. (Fig. 7.)

The principal longitudinal axis in the plane of symmetry, usually parallel to the axis of the propeller, is called the longitudinal axis; the axis perpendicular to this in the plane of symmetry is called the normal axis; and the third axis, perpendicular to the other two, is called the lateral axis. In mathematical discussions the first of these axes, drawn from front to rear, is called the X axis; the second, drawn upward, the Z axis; and the third, running from right to left, the Y axis.

CROSS-WIND FORCE.—The component perpendicular to the lift and to the drag of the total force on an aircraft due to the air through which it moves.

DRAG.—The component parallel to the relative wind of the total force on an aerofoil or aircraft due to the air through which it moves.

In the case of an airplane, that part of the drag due to the wings is called "wing resistance"; that due to the rest of the airplane is called "structural," or "parasite resistance."

LIFT.—The component of the total air force which is perpendicular to the relative wind and in the plane of symmetry. It must be specified whether this applies to a complete aircraft or parts thereof. (In the case of an airship this is often called "dynamic lift.")

PITCH, ANGLE OF.—The angle between two planes defined as follows: One plane includes the lateral axis of the aircraft and the direction of the relative wind; the other plane includes the lateral axis and the longitudinal axis. (In normal flight the angle of pitch is, then, the angle between the longitudinal axis and the direction of the relative wind.) This angle is denoted by θ, and is positive when the nose of the aircraft rises.

ROLL, ANGLE OF, or BANK, ANGLE OF.—The angle through which an aircraft must be rotated about its longitudinal axis in order to bring its lateral axis into a horizontal plane. This angle is denoted by ϕ.

YAW:

ANGLE OF.—The angle between the direction of the relative wind and the plane of symmetry of an aircraft. This angle is denoted by ψ, and is positive when the aircraft turns to the right.

YAWING.—Angular motion about the normal axis.

F. PERFORMANCE AND CONDITIONS OF FLIGHT.

CEILING:

ABSOLUTE.—The maximum height above sea level which a given aircraft can approach asymptotically, assuming standard air conditions.

SERVICE.—The height above sea level at which a given aircraft ceases to rise at a rate higher than a small specified one (100 feet per minute in United States Air Service). This specified rate may be different in the services of different countries.

DRIFT.—The angular deviation from a set course over the earth, due to cross currents of wind; hence, "drift meter."

DYNAMIC FACTOR.—The ratio between the load carried by any part of an aircraft when accelerating or when otherwise subjected to abnormal conditions and the load carried in normal flight.

FACTOR OF SAFETY.—The ratio of the ultimate strength of a member to the maximum possible load occurring under conditions specified.

F. PERFORMANCE AND CONDITIONS OF FLIGHT—Continued.

FLIGHT PATH.—The path of the center of gravity of an aircraft with reference to the earth.

FREE-FLIGHT TESTING.—The conduct of special flight tests of a scientific nature, as contrasted with performance testing (q. v.).

LOAD:

DEAD.—The structure, power plant, and essential accessories of an aircraft. Included in this are the water in the radiator, tachometer, thermometer, gauges, air-speed indicators, levels, altimeter, compass, watch and hand starter, and also, in the case of an aerostat, the amount of ballast which must be carried to assist in making a safe landing.

FULL.—The total weight of an aircraft when loaded to the maximum authorized loading of that particular type.

USEFUL.—The excess of the full load over the dead load of the aircraft itself. Therefore useful load includes the crew and passengers, oil and fuel, ballast, electric-light installation, chart board, detachable gun mounts, bomb storage and releasing gear, wireless apparatus, etc.

LOAD FACTOR.—The ratio of the ultimate strength of a member to the load under horizontal steady rectilinear flight conditions.

MARGIN OF POWER.—The difference between the power available at any given speed and in air of given density and the power required for level flight under the same conditions. The best rate of climb at any altitude depends on the maximum margin of power.

MINIMUM SPEED.—The lowest speed which can be maintained in level flight, with any throttle setting whatever.

PERFORMANCE.—The maximum and minimum speeds and rate of climb at various altitudes, the time to climb to these altitudes, and the ceiling constitute the performance characteristics of an airplane. *Performance testing* is the process of determining these quantities.

POWER LOADING.—The weight per horsepower, computed on a basis of full load and of power in air of standard density, unless otherwise stated.

RATE OF CLIMB.—The vertical component of the air speed of an aircraft; i. e., its vertical velocity with reference to the air.

RELATIVE WIND.—The motion of the air with reference to a moving body. Its direction and velocity, therefore, are found by adding two vectors, one being the velocity of the air with reference to the earth, the other being equal and opposite to the velocity of the body with reference to the earth.

RUDDER TORQUE.—The twisting effect exerted by the rudder on the fuselage, due to the relative displacement of the center of pressure of the rudder. The product of the rudder area by the distance from its center of area to the center line of the fuselage may be used as a relative measure of rudder torque.

SPEED:

AIR.—The speed of an aircraft relative to the air.

GROUND.—The horizontal component of the velocity of an aircraft relative to the earth.

G. MATERIALS.

CORD.—A species of wire made up of several strands (usually 7) twisted together as in a rope, each of the strands, in turn, being made up of several (usually 19) individual wires.

DOPE, AIRPLANE.—A general term applied to the material used in treating the cloth surface of airplane members to increase strength, produce tautness, and act as a filler to maintain air-tightness.

G. MATERIALS—Continued.

LAMINATED WOOD.—Wooden parts made up by gluing or otherwise fastening together individual wood planks or laminations with the grain substantially parallel.

PLYWOOD.—A product formed by gluing together two or more layers of wood veneer.

STRAND.—A species of wire made up of several individual wires twisted together. (There are usually 19 wires; a single wire as core, and inner layer of 6 wires, and an outer layer of 12.)

VENEER.—Thin sheets or strips of wood.

WIRE.—In aeronautics refers specifically to hard-drawn solid wire.

H. GENERAL CONSTRUCTIONAL TERMS.

FAIRING.—A member whose primary function is to produce a smooth outline and to reduce head resistance or drag.

FITTING.—A generic term for any small metal part used in the structure of an airplane.

INSPECTION WINDOW.—A small transparent window in the envelope of a balloon or in the wing of an airplane to allow inspection of the interior.

SPLICE (of a wooden member).—A joint of two or more pieces of wood in which one piece overlaps the other in such a manner as to maintain the strength.

STAY.—A wire or other tension member. For example, the stays of the wing and body trussing.

STRUT.—A member of a truss frame designed to carry compressive loads. For instance, the vertical members of the wing truss of a biplane (interplane struts) and the short vertical and horizontal member separating the longerons (q. v.) in the fuselage. (Figs. 1 and 12.)

I. CONTROLS.

AILERON.—A hinged or pivoted movable auxiliary surface of an airplane, usually part of the trailing edge of a wing, the primary function of which is to impress a rolling moment on the airplane. (Fig. 1.)

CONTROLS.—A general term applying to the means provided to enable the pilot to control the speed, direction of flight, attitude and power of an aircraft.

CONTROL COLUMN, or YOKE.—A control lever with a rotatable wheel mounted at its upper end. (*See* Control stick.) Pitching is controlled by fore-and-aft movement of the column; rolling by rotation of the wheel. "Wheel control" is that type of control in which such a column or yoke is used.

CONTROL STICK.—The vertical lever by means of which certain of the principal controls of an airplane are operated. Pitching is controlled by a fore-and-aft movement of the stick, rolling by a side-to-side movement. "Stick control" is that type of control in which such a stick is used.

ELEVATOR.—A movable auxiliary surface of an airplane, usually attached to the tail plane, the function of which is to impress a pitching moment on the aircraft. (Fig. 10.)

HORN.—The operating lever of a control surface of an aircraft; e. g., aileron horn, rudder horn, elevator horn.

RUDDER.—A hinged or pivoted surface used for the purpose of impressing yawing moments on an aircraft; i. e., for controlling its direction of flight. (Fig. 10.)

RUDDER BAR.—The foot bar by means of which the rudder is operated.

J. PROPELLER TERMS.

ASPECT RATIO OF PROPELLERS.—The ratio of propeller diameter to maximum blade width.

BLADE BACK.—The markedly convex surface of a propeller blade which corresponds to the upper surface of an aerofoil.

J. PROPELLER TERMS—Continued.

BLADE FACE.—The surface of a propeller blade, flat or slightly cambered near the tips, which corresponds to the lower surface of an aerofoil.

BLADE SETTING, ANGLE OF.—The angle which the chord of a propeller section makes with a plane perpendicular to the axis of the propeller. This angle varies along the blade, increasing as the boss is approached.

BLADE WIDTH RATIO.—The ratio of the width of a propeller blade at any point to the circumference of the circle along which that point travels when the propeller is rotating and the airplane is held stationary. When used without qualifying terms, it refers to the ratio of the maximum blade width to the circumference of the circle swept by the propeller.

TOTAL WIDTH RATIO.—The product of blade width ratio by number of blades.

BOSS.—The central portion of an air screw. The portion in which the hub is mounted.

CAMBER RATIO.—The ratio of the maximum ordinate of a propeller section to its chord.

DISK AREA.—The total area swept by a propeller; i. e., the area of a circle having a diameter equal to the propeller diameter.

INDRAFT.—The drawing in of air from in front of a propeller by the action of the rotating blades. The indraught velocity relative to the propeller is somewhat higher than that of the undisturbed air at most points of the propeller disk.

PITCH:

(*a*) PITCH, AERODYNAMIC.—The distance a propeller would have to advance in one revolution in order that the torque might be zero.

(*b*) PITCH, EFFECTIVE.—The distance an aircraft advances along its flight path for one revolution of the propeller.

(*c*) PITCH, GEOMETRICAL.—The distance an element of a propeller would advance in one revolution if it were turning in a solid nut; i. e., if it were moving along a helix of slope equal to the angle between the chord of the element and a plane perpendicular to the propeller axis. The mean geometrical pitch of a propeller, which is a quantity commonly used in specifications, is the mean of the geometrical pitches of the several elements.

(*d*) PITCH, STANDARD.—The "pitch of a propeller" is usually stated as the geometrical pitch taken at two-thirds of the radius.

(*e*) PITCH, VIRTUAL.—The distance a propeller would have to advance in one revolution in order that there might be no thrust.

(*f*) PITCH SPEED.—The product of the mean geometrical pitch by the number of revolutions of the propeller in unit time; i. e., the speed the aircraft would make if there were no slip.

(*g*) SLIP.—The difference between the effective pitch and the mean geometrical pitch. Slip is usually expressed as a percentage of the mean geometrica pitch.

PUSHER PROPELLER.—A propeller which is placed at the rear end of its shaft and pushes against the thrust bearing.

RACE ROTATION.—The rotation of the air influenced by a propeller. This rotation is much more marked in the slip-stream than in front of the propeller.

RAKE, BLADE.—The angle which the line joining the centroids of the sections of a propeller blade makes with a plane perpedicular to the propeller shaft. The rake is positive when the blades are thrown forward.

SLIP, OF PROPELLER. [*See* Pitch (*g*).]

SLIP STREAM.—The stream of air behind a propeller.

STATIC THRUST.—The thrust developed by a propeller when the aircraft is held stationary on the ground.

TRACTOR PROPELLER.—A propeller which is placed at the forward end of its shaft and pulls on the thrust bearing.

J. PROPELLER TERMS—Continued.

WINDMILL.—A small air-driven turbine with blades similar to those of a propeller exposed on an aircraft, usually in the slip-stream, and used to drive such auxiliary apparatus as gasoline pumps and radio generators.

K. INSTRUMENTS.

AIR-SPEED INDICATOR.—An anemometer mounted on an aircraft for the purpose of indicating the speed of the aircraft.

True air-speed indicator.—An instrument, usually working on the principle of the Biram or Robinson anemometers, which gives the true air speed, independent of density.

Apparent air-speed indicator.—An instrument, usually dependent on pressure measurements, the readings of which vary with the density of the air.

ALTIMETER.—An aneroid barometer, mounted on an aircraft, whose dial is marked in feet, yards, or meters.

ANEMOMETER.—Any instrument for measuring the velocity or force of the wind.

BAROGRAPH.—An instrument used to make a permanent record of variations in barometric pressure. In aeronautics the charts on which the records are made sometimes indicate altitudes directly instead of barometric pressures.

DRIFT METER.—An instrument for the measurement of the angular deviation of an aircraft from a set course, due to cross winds.

INCLINOMETER:

Relative.—An instrument giving the attitude of an aircraft with reference to apparent gravity. Such instruments are sometimes incorrectly referred to as banking indicators.

Absolute.—An instrument giving the attitude of an aircraft with reference to true gravity.

PITOT TUBE.—A tube with an end open square to a fluid stream. It is exposed with the open end pointing upstream to detect an impact pressure. It is usually associated with a coaxial tube surrounding it, having perforations normal to the axis for indicating static pressure; or there is such a tube placed near it and parallel to it, with a closed conical end and having perforations in its side. The velocity of the fluid can be determined from the difference between the impact pressure and the static pressure, as read by a suitable gauge. This instrument is often used to determine the velocity of an aircraft through the air. (Fig. 13.)

PRESSURE NOZZLE.—The apparatus which, in combination with a gauge, is used to measure the pressure due to speed through the air. Includes both Pitot and Venturi tubes. Pressure nozzles of various types are also used in yawmeters and other instruments.

RATE-OF-CLIMB INDICATOR.—An instrument indicating the vertical component of velocity of an aircraft. Most rate-of-climb meters depend on the rate of change of the atmospheric pressure.

STATOSCOPE.—An instrument to detect the existence of minute changes of atmospheric pressure, and so of small vertical motions of an aircraft.

TURN INDICATOR.—An instrument showing when the direction of the line of flight, or the direction of the projection of that line on a horizontal plane, is altering, and, in its more refined forms, giving the rate of turn, in terms either of the angular velocity or of the radius of curvature.

VENTURI TUBE.—A short tube with flaring ends and a constriction between them, so that, when fluid flows through it, there will be a suction produced in a side tube opening into the constricted throat. This tube, when combined with a Pitot tube or with one giving static pressure, forms a pressure nozzle, which may be used as an instrument to determine the speed of an aircraft through the air. (Fig. 21.)

YAWMETER.—An instrument giving by direct reading the angle of yaw.

AEROSTATICS.

L. AEROSTATIC TERMS.

AERONAUT.—The pilot of an aerostat.

AEROSTATICS.—The science which relates to the buoyancy and behavior of lighter-than-air craft.

AEROSTATION.—The operation of balloons and airships. Corresponds to aviation (q. v.), but refers to lighter-than-air craft.

APPARENT PRESSURE.—The excess of pressure inside the envelope of an aerostat over the atmospheric pressure. In the case of an airship, the excess of pressure is measured at the bottom of the envelope unless otherwise specified.

BUOYANCY.—The upward force exerted on a lighter-than-air craft due to the air which it displaces.

Center of.—The center of volume of the gas container (envelope) or the center of gravity of a gas of a balloon or airship.

Positive and negative.—The positive or negative difference between the buoyancy and the weight of a balloon or airship. The unbalanced force which causes ascent or descent.

CAPACITY.—The cubic contents or volume of an aerostat.

DISCHARGEABLE WEIGHT.—The excess of the gross buoyancy over the dead load, the crew, and such items of equipment as are essential to enable an airship to fly and land safely.

GROSS BUOYANCY.—The total upward force on an aerostat at rest; the total volume multiplied by the difference of density of the air and the contained gas.

PERMEABILITY.—The measure of the rate of diffusion of gas through intact balloon fabric; usually expressed in cubic meters per square meter per 24 hours.

PURITY OF A GAS.—The percentage, by number of molecules, of the light gas used for inflation, such as hydrogen, to all the gases within the container.

TAIL DROOP.—A deformation of the airship in which the axis bends downward at the after end.

M. PARTS OF AEROSTATS.

AIR SCOOP.—A projecting cowl, which, by using the dynamic pressure of the relative wind or slipstream, serves to maintain air pressure in the interior of the ballonet of an aerostat. (Fig. 2.)

APPENDIX.—The tube at the bottom of a balloon, used for inflation. In the case of a spherical balloon it also serves to increase the "head" of gas, and so to build up an internal pressure sufficient to keep the envelope from being pulled out of shape by the weight of the basket. (Fig. 6.)

AUTOMATIC VALVE.—An automatic escape and safety valve for the purpose of regulating internal pressure in an aerostat.

BALLONET.—A small balloon within the interior of a balloon or airship for the purpose of controlling the ascent or descent and for maintaining pressure on the outer envelope so as to prevent deformation. The ballonet is kept inflated with air at the required pressure, under the control of valves, by a blower or by the action of the wind caught in an air scoop.

BALLOON BED.—A mooring place on the ground for a captive balloon.

BALLOON FABRIC.—The finished material, usually rubberized, of which balloon or airship envelopes are made.

Biased.—Plied fabric in which the threads of the plies are at an angle to each other.

Parallel.—Plied fabric in which the threads of the plies are parallel to each other.

BASKET.—The car suspended beneath a balloon, for passengers, ballast, etc.

M. PARTS OF AEROSTATS—Continued.

BONNET.—The appliance, having the form of a parasol, which protects the valve of a spherical balloon against rain.

BOW STIFFENERS.—Rigid members attached to the bow of a nonrigid or semirigid envelope to reinforce it against the pressure caused by the motion of the airship. (Sometimes called nose stiffeners.)

BRIDLE.—A sling of cordage which has its ends attached to the envelope of a balloon or airship and a rope or cable running from an intermediate point.

CAR.—The nacelle of an airship.

CONCENTRATION RING:

(*a*) Free balloon.—A hoop to which are attached the ropes suspending the basket and to which the net is also secured.

(*b*) Parachute.—A hoop to which the rigging of the parachute is attached and also the line sustaining the passenger.

(*c*) Airship.—A metal ring to which several rigging lines are brought from the envelope and from which one or more lines also lead to the car.

CROW'S-FOOT.—A system of diverging short ropes for distributing the pull of a single rope.

DRAG ROPE.—The rope dropped by an airship in order to allow it to be secured by a landing party.

DRIP FLAP.—A strip of fabric attached by one edge to the envelope of an aerostat so that rain runs off its free edge instead of dripping into the basket or car. The drip flap assists also to keep the suspension ropes dry and nonconducting.

ENVELOPE.—The outer covering of a rigid airship; or, in the case of a balloon or a nonrigid airship, the bag which contains the gas.

EQUATOR.—The largest horizontal circle of a spherical balloon.

FINS (KITE BALLOON).—The air-inflated lobes intended to keep the balloon headed into the wind.

GORE.—The portion of the envelope of a balloon or airship included between two adjacent meridian seams.

GROUND CLOTH.—Canvas placed on the ground to protect a balloon.

HOG (AIRSHIP).—A distortion of the envelope in which the axis becomes convex upward or both ends droop.

HULL.—The main structure of a rigid airship, consisting of a covered elongated framework which incloses the gas bags and which supports the cars and equipment.

JACKSTAY.—A longitudinal rigging provided to maintain the correct distance between the heads of various riggings on an airship.

KEEL.—A member or assembly of members which provides longitudinal strength to an airship of rigid or semirigid type. In the case of a rigid airship the keel is usually an elaborately trussed girder and may be inclosed within the envelope or may project beyond (usually below) the regular cross-sectional form of the envelope.

Articulated.—A keel made up of a series of members hinged together at their ends.

LOBES.—Inflated bags at the stern of an elongated balloon, designed to give it directional stability. Also used to denote the sections into which the envelope is sometimes (e. g., in the Astra-Torres) divided by the tension of the internal rigging.

MOORING HARNESS.—The system of bands of tape over the top of a balloon to which are attached the mooring ropes.

NET.—A rigging made of ropes and twine on spherical balloons which supports the weight of the basket, etc., distributing the load over the entire upper surface of the envelope.

NOSE CAP.—A cap used to reinforce the bow stiffeners of an airship.

PANEL.—The unit piece of fabric of which the envelope of an aerostat is made.

M. PARTS OF AEROSTATS—Continued.

PATCH.—A strengthened or reinforced flap of fabric, of variable form, according to the maker, which is cemented to the envelope and forms an anchor by which some portion of the machine is attached to the envelope. (Fig. 2.)

PROOFING.—Material applied to the fabric of an aerostat at the time of manufacture, to protect it against weather or to prevent the passage of gas.

RIP CORD.—The rope running from the rip panel of a balloon or nonrigid airship to the basket, the pulling of which tears off the rip panel and causes immediate deflation.

RIP PANEL.—A strip in the upper part of a balloon or nonrigid airship which is torn off when immediate deflation is desired.

SAFETY LOOP.—A loop formed immediately outside the conical reversing bag through which the valve rope emerges from the bottom of an aerostat. Before the automatic valve can be opened by the aid of the valve rope the fastening of this safety loop is torn off by a strong pull on the valve rope from the nacelle.

SERPENT.—A short, heavy trail rope.

SUSPENSION BAND.—The band around a balloon or airship to which are attached the main bridle suspensions of the basket or car.

SUSPENSION BAR.—The bar used for the concentration of basket suspension ropes in captive balloons.

TAIL CUPS.—A steadying device attached by lines at the rear of certain types of elongated captive balloons. Somewhat similar to a sea anchor. (Fig. 17.)

TRAIL ROPE.—The long trailing rope attached to a spherical balloon to serve as a brake and as a variable ballast.

TOGGLE.—A short crossbar of wood or metal, having a shouldered groove, which is fitted at the end of a rope at right angles to it. It is used for obtaining a quickly detachable connection with an eye at the end of another rope. (Fig. 18.)

TRAJECTORY BAND.—A band of webbing carried in a curve over the top of the envelope of an airship to distribute the stresses due to the suspension. The use of trajectory bands was introduced in the Parseval airships. (Fig. 19.)

TERMS LIMITED TO AIRPLANES.

N. WING PARTS.

ANTIDRAG WIRES.—Wires designed primarily to resist forces acting parallel to the planes of the wings of an airplane and in the same direction as the direction of flight.

ANTILIFT WIRES.—Wires in an airplane intended mainly to resist forces in the opposite direction to the lift, and to oppose the lift wires and prevent distortion of the structure by overtightening of those members.

BAY.—The cubic section of a truss included between two transversely adjacent sets of struts of an airplane. The first bay is the one closest to the plane of symmetry.

CABANE.—A pyramidal or prismoidal framework to which wire or cable stays are secured.

CELL.—The entire structure of the wings and wing trussing on one side of the fuselage of an airplane, or between fuselages or nacelles, where there are more than one.

DRAG STRUT.—A compression member of the internal bracing system of an aerofoil.

DRAG WIRES.—All wires designed primarily to resist forces acting parallel to the planes of the wings of an airplane and opposite to the direction of flight.

INTERNAL DRAG WIRES are concealed inside the wings.

EXTERNAL DRAG WIRES run from the wing cell to the nose of the fuselage or some other part of the machine.

KING POST.—The main compression member of a trussing system applied to a member subject to bending. (Fig. 4.)

N. WING PARTS—Continued.

LIFT WIRES.—The wires which transmit the lift on the outer portion of the wings of an airplane in toward the fuselage or nacelle. These wires usually run from the top of an interplane strut to the bottom of the strut next nearer the fuselage.

MAIN SUPPORTING SURFACE.—A pair of wings, extending on the same level from tip to tip of an airplane; e. g., a triplane has three main supporting surfaces. The main supporting surfaces do not include any surfaces intended primarily for control or stabilizing purposes.

PANEL.—A portion of a wing of an airplane which is constructed entirely separate from the rest of the wing, and which is attached to the remainder by bolts and fittings.

PHILLIPS' ENTRY.—A reversal of curvature of the lower surface of an aerofoil near the leading edge. The result is to decrease the drag and provide more depth for the front spar. (Fig. 9.)

STAGGER WIRES.—Wires connecting the upper and lower main supporting surfaces of an airplane, and lying in planes substantially parallel to the plane of symmetry.

WING.—The portion of a main supporting surface of an airplane on one side of the plane of symmetry; e. g., a biplane has four wings.

WING RIB.—A fore-and-aft member of the wing structure of an airplane, used to give the wing section its form and to transmit the load from the fabric to the spars. (Fig. 20.)

Rib, compression.—A heavy rib designed to have the above functions and also to act as a strut opposing the pull of the wires in the internal drag truss. (Fig. 20.)

Rib, form.—An incomplete rib, frequently consisting only of a strip of wood extending from the leading edge to the front spar, which is used to assist in maintaining the form of the wing where the curvature of the aerofoil section is sharpest. (Fig. 20.)

WING SPARS.—The principal transverse structural elements of the wing assembly of an airplane. The load is transmitted from the ribs to the spars, and thence to the lift and drag trusses. (Fig. 20.)

WING TRUSS.—The framing by which the wing loads of an airplane are transmitted to the fuselage; comprises struts, wires, or tie-rods, and spars.

O. FUSELAGE AND NACELLE PARTS.

BULKHEAD.—A transverse structural member of a fuselage or nacelle, continuous around the periphery.

COCKPIT.—The open space in which the pilot and passengers are accommodated. A cockpit is never completely housed in.

COWLING.—The metal covering which houses the engine, and sometimes a portion of the fuselage or nacelle as well.

FIRE WALL.—A metal plane, so set as to isolate from the engine the other parts of the airplane structure, and so to reduce the risk from a backfire.

FUSELAGE.—The elongated structure, of approximately streamline form, to which are attached the wings and tail unit of an airplane. In general it is designed to hold the passengers.

LONGERON.—A fore-and-aft member of the framing of an airplane fuselage or nacelle, usually continuous across a number of points of support. (Fig. 12.)

MONOCOQUE.—A type of fuselage which is constructed by wrapping strips of veneer around formers, and in which the veneer is primarily depended on to carry stresses arising in the fuselage.

NACELLE.—The inclosed shelter for passengers or for a power plant. A nacelle is usually shorter than a fuselage and does not carry the tail unit.

O. FUSELAGE AND NACELLE PARTS—Continued.

SHUTTERS.—The adjustable blinds or vanes which are used to control the amount of air flowing through the radiator and so to regulate the temperature of the cooling water.

SPINNER.—A fairing, usually made of sheet metal and roughly conical or paraboloid in form, which is attached to the propeller boss and revolves with it.

STATION.—A term used to denote the location of framing attachment in a fuselage or nacelle (strut points in a trussed fuselage, bulkhead points in a veneer fuselage)

P. LANDING GEAR PARTS.

FLOAT.—A completely inclosed water-tight structure attached to an aircraft in order to furnish it buoyancy when in contact with the surface of the water. In float seaplanes the crew is carried in a fuselage or nacelle separate from the float.

FLOTATION GEAR.—An emergency landing gear attached to an airplane, which will permit of safe landing on the water and provide buoyancy when resting on the surface of the water.

HULL.—The portion of a boat seaplane which furnishes buoyancy when in contact with the surface of the water, to which the main supporting surfaces and other parts are attached, and which contains accommodations for the crew.

LANDING GEAR.—The understructure of an aircraft designed to carry the load when in contact with the land or water.

SHOCK ABSORBER.—A spring or elastic member, designed to prevent the imposition of large accelerations on the fuselage, wings, and other heavy concentrated weights. Shock absorbers are usually interposed between the wheels, floats, or tail skid, and the remainder of the airplane to secure resiliency in landing and taxi-ing.

SHOCK-ABSORBER HYSTERESIS.—The ratio of the work absorbed in the shock absorber during one complete cycle to the total energy transmitted to the shock absorber during the first half of the cycle.

SKIDS.—Runners used as members of the landing gear and designed to aid the aircraft in landing or taxi-ing.

Tail skid.—A skid used to support the tail when in contact with the ground

Wing skid.—A skid placed near the wing tip and designed to protect the wing from contact with the ground.

Q. MISCELLANEOUS PARTS.

FINS.—Small stationary surfaces, substantially vertical, attached to different parts of aircraft, in order to promote stability; for example, tail fins, skid fins, etc. Fins are sometimes adjustable (Fig. 10.)

Skid fins.—Fore-and-aft vertical surfaces, usually placed well out toward the tips of the upper plane, designed to provide the vertical keel surface required for stability.

STEP.—A break in the form of the bottom of a float or hull, designed to assist in securing a dynamic reaction from the water.

TAIL BOOM.—A spar or outrigger connecting the tail surfaces and main supporting surfaces. Usually used on pushers. (Fig. 3.)

TAIL PLANE.—A stationary horizontal, or nearly horizontal, tail surface, used to stabilize the pitching motion. Often called "stabilizer." (Fig. 10.)

TAIL UNIT.—The tail surfaces of an aircraft.

R. LANDING FIELDS, ETC.

AIRDROME.—A field providing facilities for aircraft to land and take-off, and equipped with hangars, shops, and a supply depot for the storage, maintenance, and repair of aircraft.

R. LANDING FIELDS, ETC.—Continued.

HANDLING TRUCK.—A truck, mounted on wheels or sliding on ways, on which airplanes or seaplanes may be placed to facilitate moving them about and carrying them to and from their hangars.

HANGAR.—A shelter for housing aircraft.

LANDING FIELD.—A field of such a nature as to permit of airplanes landing or takingoff.

S. OPERATION AND MANEUVERS.

AVIATOR.—The operator or pilot of heavier-than-air craft. This term is applied regardless of the sex of the operator.

BANK.—To incline an airplane laterally. Right bank is to incline the airplane with the right wing down. Also used as a noun to describe the position of an airplane when its lateral axis is inclined to the horizontal.

BARREL ROLL.—An aerial maneuver in which a complete revolution about the longitudinal axis is made, the direction of flight being approximately maintained.

DIVE.—A steep glide.

GLIDE, TO.—To descend at a normal angle of attack without engine power sufficient for level flight, the propeller thrust being replaced by a component of gravity along the line of flight.

LOOP.—An aerial maneuver in which the airplane describes an approximately circular path in the plane of the longitudinal and normal axes, the lateral axis remaining horizontal, and the upper side of the airplane remaining on the inside of the circle.

NOSE HEAVY.—The condition of an aircraft in which, in any given condition of normal flight, the nose tends to drop if the longitudinal control is released; i. e., the condition in which the pilot has to exert a pull on the control stick or column to maintain the given condition.

PANCAKE, TO.—To "level off" an airplane higher than for a normal landing, causing it to stall and descend with the wings at a very large angle of attack and approximately without bank, on a steeply inclined path.

REVERSE TURN.—A rapid maneuver to reverse the direction of flight of an airplane, made by a half loop and half roll in either sequence.

RIGGER.—One who is employed in assembling and aligning aircraft.

RIGGING.—The assembling and aligning of an aircraft.

SIDE SLIPPING.—Sliding with a component of velocity along the lateral axis which is inclined and in the direction of the lower end of that axis. When it occurs in connection with a turn it is the opposite of skidding (q. v.).

SKIDDING.—Sliding sidewise away from the center of curvature when turning. It is usually caused by banking insufficiently and is the opposite of side slipping (q. v.).

SOAR, TO.—To fly without engine power and without loss of altitude. Lightly loaded gliders will soar in rising currents of air.

SPIN.—An aerial maneuver consisting of a combination of roll and yaw, with the longitudinal axis of the airplane inclined steeply downward. The airplane descends in a helix of large pitch and very small radius, the upper side of the airplane being on the inside of the helix, and the angle of attack on the inner wing being maintained at an extremely large value.

STALLING.—A term describing the condition of an airplane which from any cause has lost the relative air speed necessary for control.

TAIL HEAVY.—The condition of an aircraft in which, in any given condition of normal flight, the nose tends to rise if the longitudinal control is released; i. e., the condition in which the pilot has to exert a push on the control stick or column to maintain the given condition.

S. OPERATION AND MANEUVERS—Continued.

TAIL SLIDE.—The rearward motion which certain airplanes may be made to take after having been brought into a stalling position.

TAXI, TO.—To run an airplane over the ground, or a seaplane on the surface of water, under its own power.

WARP, TO.—To change the form of a wing by twisting it. Warping is sometimes used to maintain the lateral equilibrium of an airplane.

ZOOM, TO.—To climb for a short time at an angle greater than that which can be maintained in steady flight, the machine being carried upward at the expense of its stored kinetic energy. This term is sometimes used by pilots to denote any sudden increase in the upward slope of the flight path.

T. DIMENSIONS AND CHARACTERISTICS.

ANGLE, GLIDING.—The acute angle which the flight path makes with the horizontal when descending in still air under the influence of gravity alone; i. e., without power from the engine.

ANGLE, LANDING.—The angle of attack of the main supporting surfaces of an airplane at the instant of touching the ground in a three-point landing; i. e., the angle between the wing chord and the horizontal when the machine is resting on the ground in its normal position.

ANGLE OF INCIDENCE (in directions for rigging).—In the process of rigging an airplane some arbitrary definite line in the airplane is kept horizontal; the angle of incidence of a wing, or of any aerofoil, is the angle between its chord and this horizontal line, which may be the line of the upper longerons of the fuselage or nacelle or the thrust line.

ANGLE OF TAIL SETTING.—The acute angle between the chord of the wings of an airplane and the chord of the tail plane. Denoted by the symbol β.

DIHEDRAL ANGLE.—The main supporting surfaces of an airplane are said to have a dihedral angle when both right and left wings are upwardly or downwardly inclined to a horizontal transverse line. The angle is measured by the inclination of each wing to the horizontal. If the inclination is upward, the angle is said to be positive; if downward, negative. The several main supporting surfaces of an airplane may have different amounts of dihedral. (Fig. 5.)

DOWNWASH ANGLE.—The acute angle through which the air stream relative to the airplane is deflected by an aerofoil. It is measured in a plane parallel to the plane of symmetry, and is denoted by the symbol ϵ.

GAP.—The shortest distance between the planes of the chords of the upper and lower wings of a biplane, measured along a line perpendicular to the chord of the upper wing at any designated point of its entering edge. (Fig. 11.)

LENGTH OF FUSELAGE.—The distance from the nose of the fuselage (including the engine bed and radiator, if present) to the after end of the fuselage, not including the control and stabilizing surfaces.

MEAN CHORD OF A WING.—The quotient obtained by dividing the wing area by the extreme dimension of the wing projection at right angles to the chord.

MEAN CHORD OF A COMBINATION OF WINGS.—If c be the mean chord of the combination; c_1, c_2, c_3, etc., the mean chords of each wing corresponding to areas S_1, S_2, S_3, etc., then

$$c=\frac{c_1S_1+c_2S_2+c_3S_3+\quad\cdot\quad\cdot\quad\cdot\quad\cdot}{S_1+S_2+S_3+\quad\cdot\quad\cdot\quad\cdot\quad\cdot}$$

T. DIMENSIONS AND CHARACTERISTICS—Continued.

MEAN SPAN OF A COMBINATION OF WINGS.—If s be the mean span of the combination; s_1, s_2, and s_3; etc., the spans of each pair of wings separately corresponding to areas S_1, S_2, S_3, etc., then

$$s=\frac{s_1S_1+s_2S_2+s_3S_3+\quad . \quad . \quad . \quad .}{S_1+S_2+S_3+\quad . \quad . \quad . \quad .}$$

OVER-ALL LENGTH.—The distance from the extreme front to the extreme rear of an aircraft, including the propeller and the tail unit.

OVERHANG.—One-half the difference in the span of any two main supporting surfaces of an airplane. The overhang is positive when the upper of the two main supporting surfaces has the larger span. (Fig. 5.)

RAKE.—The cutting away of the wing tip at an angle so that the main supporting surfaces, seen from above, will appear of trapezoidal form. The amount of rake is measured by the angle between the straight portion of the wing-tip outline and the plane of symmetry. The rake is positive when the trailing edge is longer than the leading edge.

SPAN (or SPREAD).—The maximum distance laterally from tip to tip of an airplane inclusive of ailerons, or the lateral dimension of an aerofoil.

STAGGER.—The amount of advance of the entering edge of an upper wing of a biplane, triplane, or multiplane over that of a lower, expressed as percentage of gap; it is considered positive when the upper wing is forward, and is measured from the entering edge of the upper wing along its chord to the point of intersection of this chord with a line drawn perpendicular to the chord of the upper wing at the entering edge of the lower wing, all lines being drawn in a plane parallel to the plane of symmetry. (Fig. 11.)

In directions for rigging: The horizontal distance between the entering edge of the upper plane and that of the lower when the airplane is in the standard position; i. e., when the arbitrary line of reference in the airplane is horizontal. (This line is usually the axis of the propeller shaft.)

SWEEP BACK.—The angle, measured in a plane parallel to the lateral axis and to the chord of the main planes, between the lateral axis of an airplane and the entering edge of the main planes. (Fig. 16.)

WASH.—The disturbance in the air produced by the passage of an aerofoil.

WASHIN.—A permanent increase in the angle of attack near the tip of the wing.

WASHOUT.—A permanent decrease in the angle of attack near the tip of the wing.

WING LOADING.—The weight carried per unit area of supporting surface. The area used in computing the wing loading should include the ailerons, but not the tail-plane or elevators.

U. STABILITY THEORY.

DAMPING FACTOR.—The percentage of damping in one period; i. e., $1-e^{-\lambda T}$, where λT is the logarithmic decrement (q. v.).

DIVERGENCE.—A disturbance which increases without oscillation.

LOGARITHMIC DECREMENT.—The natural logarithm of the ratio of two successive amplitudes of an oscillation; i. e., at an interval of one period. The general equation of an oscillation may be written

$$s=Ae^{-\lambda t}\sin(pt-\propto),$$

in which A, λ, p, and $\propto$ are constants. The amplitude of oscillation is $Ae^{-\lambda t}$. The phase of the vibration is $pt-\propto$. The period is $2\pi/p$, and may be written T.

It follows that the logarithmic decrement is λT. If λ is a positive number, the vibration is said to be "damped." (In an unstable oscillation (q. v.), the quantity λ is a negative number.)

PERIOD.—The time taken for a complete oscillation.

U. STABILITY THEORY—Continued.

PHUGOID OSCILLATION.—A long period oscillation characteristic of the disturbed longitudinal motion of an airplane.

RESISTANCE DERIVATIVES.—Quantities expressing the variation of the forces and moments on aircraft due to disturbance of steady motion. They form the experimental basis of the theory of stability, and from them the periods and damping factors of aircraft can be calculated. In the general case there are 18 translatory and 18 rotary derivatives.

Rotary.—Resistance derivatives expressing the variation of moments and forces due to small increases in the rotational velocities of the aircraft.

Translatory.—Resistance derivatives expressing the variation of moments and forces due to small increases in the translatory velocities of the aircraft.

RIGHTING MOMENT.—A moment which tends to restore an aircraft to its previous attitude after any small rotational displacement.

SPIRAL INSTABILITY.—The instability on account of which an airplane tends to depart from straight flight, by a combination of side slipping and banking, the latter being always too great for the turn.

STABILITY:

(a) Static stability.—A machine is statically stable if, when slightly displaced by rotation about its center of gravity (as in wind tunnel experimentations), moments come into play which tend to return the machine to its normal attitude.

(b) Dynamical stability.—A machine is dynamically stable if, when displaced from steady motion in flight, it tends to return to that steady state of motion.

In a general way, the difference between static stability and dynamical stability is that the former depends on restoring moments and the latter on damping factors.

Automatic.—Stability dependent upon movable control surfaces. The term "automatic stability" is usually applied to those cases in which the control surfaces are automatically operated by mechanical means.

Directional.—Stability with reference to rotations about the normal axis; i. e., a machine possessing directional stability in its simplest form is one for which Nv is negative. Owing to symmetry, directional stability is closely associated with lateral stability.

Inherent.—Stability of an aircraft due solely to the disposition and arrangement of its fixed parts; i. e., that property which causes it when disturbed to return to its normal attitude of flight without the use of the controls or the interposition of any mechanical device.

Lateral.—Stability with reference to disturbances involving rolling, yawing, or side slipping; i. e., disturbances in which the position of the plane of symmetry of the aircraft is affected.

Longitudinal.—Stability with reference to disturbances in the plane of symmetry; i. e., disturbances involving pitching and variations of the longitudinal and normal velocities.

STABLE OSCILLATION.—An oscillation which tends to die out.

UNSTABLE OSCILLATION.—An oscillation of which the amplitude tends to increase.

ENGINE TERMS.

V. ENGINE TERMS.

CONSUMPTION PER B. H. P. HOUR.—The quantity of fuel or oil consumed per hour by an engine running at ground level divided by the brake-horsepower developed, unless specifically stated otherwise.

DRY WEIGHT.—The weight of an engine including carburetors, propeller hub assembly, and ignition system complete, but excluding exhaust manifolds, oil, and water.

V. ENGINE TERMS—Continued.

HORSEPOWER OF AN ENGINE, MAXIMUM.—The maximum horsepower which can be safely maintained for periods not less than five minutes.

HORSEPOWER OF AN ENGINE, NORMAL.—The highest horsepower which can be safely maintained for long periods.

REVOLUTIONS, MAXIMUM.—The maximum number of revolutions per minute that may be maintained for periods not less than five minutes.

REVOLUTIONS, NORMAL.—The highest number of revolutions per minute that may be maintained for long periods.

RIGHT-HAND ENGINE.—An engine the final power delivery shaft of which rotates clockwise when viewed by an observer looking along the engine toward the power delivery end.

WEIGHT PER HORSEPOWER.—The dry weight of an engine divided by the normal horsepower developed at ground level.

REPORT No. 91

NOMENCLATURE FOR AERONAUTICS

By the National Advisory Committee for Aeronautics

PART II.

ALPHABETICAL NOMENCLATURE

AERODYNAMIC PITCH.—(*See* Pitch.)

AEROFOIL.—A winglike structure, flat or curved, designed to obtain reaction upon its surfaces from the air through which it moves.

AEROFOIL SECTION.—A section of an aerofoil made by a plane parallel to the plane of symmetry of the aerofoil and to the normal direction of motion.

AERONAUT.—The pilot of an aerostat.

AEROSTAT.—An aircraft which embodies a container filled with a gas lighter than air and which is sustained by the buoyancy of this gas; e. g., airship, balloon.

AEROSTATICS.—The science which relates to the buoyancy and behavior of lighter-than-air craft.

AEROSTATION.—The operation of balloons and airships. Corresponds to aviation (q. v.). but refers to lighter-than-air craft.

AILERON.—A hinged or pivoted movable auxiliary surface of an airplane, usually part of the trailing edge of a wing, the primary function of which is to impress a rolling moment on the airplane. (Fig. 1.)

AIR SCOOP.—A projecting cowl, which, by using the dynamic pressure of the relative wind or slip-stream, serves to maintain air pressure in the interior of the ballonet of an aerostat. (Fig. 2.)

AIRCRAFT.—Any form of craft designed for the navigation of the air—airplanes, airships, balloons, helicopters, kites, kite balloons, ornithopters, gliders, etc.

AIRDROME.—A field providing facilities for aircraft to land and take off and equipped with hangars, shops, and a supply depot for the storage, maintenance, and repair of aircraft.

AIRPLANE.—A form of aircraft heavier than air which obtains support by the dynamic reaction of the air against the wings and which is driven through the air by a screw propeller, This term is commonly used in a more restricted sense to refer to airplanes fitted with landing gear suited to operation from the land. If the landing gear is suited to operation from the water, the term "seaplane" is used. (See definition.)

Pusher.—A term commonly applied to a single-engine airplane with the propeller in the rear of the main supporting surfaces. (Fig. 3.)

Tandem.—An airplane with two or more sets of wings of substantially the same area (not including the tail unit) placed one in front of the other and on about the same level.

Tractor.—A term commonly applied to a single-engined airplane with the propeller forward of the main supporting surfaces. (Fig. 4.)

AIRSHIP.—A form of aerostat provided with a propelling system and with means of controlling the direction of movement.

Nonrigid.—An airship whose form is maintained by the pressure of the contained gas.

Rigid.—An airship whose form is maintained by a rigid structure contained within the envelope.

Semirigid.—An airship whose form is maintained by means of a rigid or jointed keel and by gas pressure.

AIR SPEED.—(*See* Speed.)

AIR-SPEED INDICATOR.—(*See* Indicator.)

ALTIMETER.—An aneroid barometer, mounted on an aircraft, whose dial is marked in feet, yards, or meters.

ANEMOMETER.—Any instrument for measuring the velocity or force of the wind.

ANGLE, CRITICAL.—The angle of attack at which the flow about an aerofoil changes abruptly, with corresponding abrupt changes in the lift and drag coefficients. An aerofoil may have two or more critical angles, one of which almost always corresponds to the angle of maximum lift.

ANGLE, DIHEDRAL.—The main supporting surfaces of an airplane are said to have a dihedral angle when both right and left wings are upwardly or downwardly inclined to a horizontal transverse line. The angle is measured by the inclination of each wing to the horizontal. If the inclination is upward, the angle is said to be positive; if downward, negative. The several main supporting surfaces of an airplane may have different amounts of dihedral. (Fig. 5.)

ANGLE, DOWNWASH.—The acute angle through which the air stream relative to the airplane is deflected by an aerofoil. It is measured in a plane parallel to the plane of symmetry, and is denoted by the symbol ϵ.

ANGLE, GLIDING.—The acute angle which the flight path makes with the horizontal when descending in still air under the influence of gravity alone; i. e., without power from the engine.

ANGLE, LANDING.—The angle of attack of the main supporting surfaces of an airplane at the instant of touching the ground in a three-point landing; i. e., the angle between the wing chord and the horizontal when the machine is resting on the ground in its normal position.

ANGLE OF ATTACK.—The acute angle between the direction of the relative wind and the chord of an aerofoil; i. e., the angle between the chord of an aerofoil and its motion relative to the air. (This definition may be extended to any body having an axis.)

ANGLE OF INCIDENCE (in directions for rigging).—In the process of rigging an airplane some arbitrary definite line in the airplane is kept horizontal; the angle of incidence of a wing, or of any aerofoil, is the angle between its chord and this horizontal line, which may be the line of the upper longerons of the fuselage or nacelle or the thrust line.

ANGLE OF PITCH.—The angle between two planes defined as follows: One plane includes the lateral axis of the aircraft and the direction of the relative wind; the other plane includes the lateral axis and the longitudinal axis. (In normal flight the angle of pitch is, then, the angle between the longitudinal axis and the direction of the relative wind.) This angle is denoted by θ, and is positive when the nose of the aircraft rises.

ANGLE OF PROPELLER BLADE SETTING.—The angle which the chord of a propeller section makes with a plane perpendicular to the axis of the propeller. This angle varies along the blade, increasing as the boss is approached.

ANGLE OF ROLL, or ANGLE OF BANK.—The angle through which an aircraft must be rotated about its longitudinal axis in order to bring its lateral axis into a horizontal plane. This angle is denoted by ϕ.

ANGLE OF TAIL SETTING.—The acute angle between the chord of the wings of an airplane and the chord of the tail plane. Denoted by the symbol β.

ANGLE OF YAW.—The angle between the direction of the relative wind and the plane of symmetry of an aircraft. This angle is denoted by ψ, and is positive when the aircraft turns to the right.

ANGLE OF ZERO LIFT.—(*See* Zero lift angle.)

ANTIDRAG WIRES.—(*See* Wires.)

ANTILIFT WIRES.—(*See* Wires.)

APPARENT PRESSURE.—The excess of pressure inside the envelope of an aerostat over the atmospheric pressure. In the case of an airship, the excess of pressure is measured at the bottom of the envelope unless otherwise specified.

APPENDIX.—The tube at the bottom of a balloon, used for inflation. In the case of a spherical balloon it also serves to increase the "head" of gas, and so to build up an internal pressure sufficient to keep the envelope from being pulled out of shape by the weight of the basket. (Fig. 6.)

ASPECT RATIO.—The ratio of span to mean chord of an aerofoil.

ASPECT RATIO OF PROPELLERS.—The ratio of propeller diameter to maximum blade width.

ATTACK, ANGLE OF.—(*See* Angle.)

ATTITUDE.—The attitude of an aircraft is determined by the inclination of its axes to a "frame of reference" fixed to the earth; i. e., the attitude depends entirely on the position of the aircraft as seen by an observer on the ground.

AUTOMATIC VALVE.—An automatic escape and safety valve for the purpose of regulating internal pressure in an aerostat.

AVIATOR.—The operator or pilot of heavier-than-air craft. This term is applied regardless of the sex of the operator.

AXES OF AN AIRCRAFT.—Three fixed lines of reference; usually centroidal and mutually rectangular. (Fig. 7.)

The principal longitudinal axis in the plane of symmetry, usually parallel to the axis of the propeller, is called the longitudinal axis; the axis perpendicular to this in the plane of symmetry is called the normal axis; and the third axis, perpendicular to the other two, is called the lateral axis. In mathematical discussions the first of these axes, drawn from front to rear, is called the X axis; the second, drawn upward, the Z axis; and the third, running from right to left, the Y axis.

BALANCED SURFACE.—(*See* Surface.)

BALLONET.—A small balloon within the interior of a balloon or airship for the purpose of controlling the ascent or descent and for maintaining pressure on the outer envelope so as to prevent deformation.

BALLOON.—A form of aerostat deriving its support in the air from the buoyancy of the air displaced by an envelope the form of which is maintained by the pressure of a contained gas lighter than air, and having no power plant or means of controlling the direction of flight in the horizontal plane.

BARRAGE.—A small captive balloon, raised as a protection against attacks by airplanes.

CAPTIVE.—A balloon restrained from free flight by means of a cable attaching it to the earth.

KITE.—An elongated form of captive balloon, fitted with tail appendages to keep it headed into the wind, and usually deriving increased lift due to its axis being inclined to the wind. A Caquot balloon is of this type. (Fig. 8.)

NURSE.—A small balloon made of heavy fabric, employed as a portable means for storing gas. Sometimes one is so connected as to automatically allow for the expansion or contraction of the gas in an aerostat when on the ground.

PILOT.—A small balloon sent up to show the direction of the wind by observations of its flight with theodolites.

SOUNDING.—A small balloon sent aloft without passengers but with registering meteorological and other instruments.

BALLOON BED.—A mooring place on the ground for a captive balloon.

BALLOON FABRIC.—(*See* Fabric.)

BANK.—To incline an airplane laterally. Right bank is to incline the airplane with the right wing down. Also used as a noun to describe the position of an airplane when its lateral axis is inclined to the horizontal.

BANK, ANGLE OF.—(*See* Angle of roll.)

BAROGRAPH.—An instrument used to make a permanent record of variations in barometric pressure. In aeronautics the charts on which the records are made sometimes indicate altitudes directly instead of barometric pressures.

BARRAGE BALLOON.—(*See* Balloon.)

BARREL ROLL.—An aerial maneuver in which a complete revolution about the longitudinal axis is made, the direction of flight being approximately maintained.

BASKET.—The car suspended beneath a balloon, for passengers, ballast, etc.

BAY.—The cubic section of a truss included between two transversely adjacent sets of struts of an airplane. The first bay is the one closest to the plane of symmetry.

BIPLANE.—A form of airplane whose main supporting surface is divided into two parts, superimposed.

BLADE BACK.—The markedly convex surface of a propeller blade which corresponds to the upper surface of an aerofoil.

BLADE FACE.—The surface of a propeller blade, flat or slightly cambered near the tips, which corresponds to the lower surface of an aerofoil.

BLADE SETTING, ANGLE OF.—(*See* Angle.)

BLADE WIDTH RATIO.—The ratio of the width of a propeller blade at any point to the circumference of the circle along which that point travels when the propeller is rotating and the airplane is held stationary. When used without qualifying terms, it refers to the ratio of the maximum blade width to the circumference of the circle swept by the propeller.

BOAT SEAPLANE.—(*See* Seaplane.)

BONNET.—The appliance, having the form of a parasol, which protects the valve of a spherical balloon against rain.

BOSS.—The central portion of an airscrew. The portion in which the hub is mounted.

BOW STIFFENERS.—Rigid members attached to the bow of a nonrigid or semirigid envelope to reinforce it against the pressure caused by the motion of the airship. (Sometimes called nose stiffeners.)

BRIDLE.—A sling of cordage which has its ends attached to the envelope of a balloon or airship and a rope or cable running from an intermediate point.

BULKHEAD.—A transverse structural member of a fuselage or nacelle, continuous around the periphery.

BUOYANCY.—The upward force exerted on a lighter-than-air craft due to the air which it displaces.

Center of.—The center of volume of the gas container or the center of gravity of the gas (envelope) of a balloon or airship.

Gross.—The total upward force on an aerostat at rest: the total volume multiplied by the difference of density of the air and the contained gas.

Positive and negative.—The positive or negative difference between the buoyancy and the weight of a balloon or airship. The unbalanced force which causes ascent or descent.

CABANE.—A pyramidal or prismoidal framework to which wire or cable stays are secured.

CAMBER.—The convexity or rise of the curve of an aerofoil from its chord, usually expressed as the ratio of the maximum departure of the curve from the chord to the length of the chord. "Top camber" refers to the top surface of an aerofoil and "bottom camber" to the bottom surface; "mean camber" is the mean of these two.

CAMBER RATIO.—The ratio of the maximum ordinate of a propeller section to its chord.

CAPACITY.—The cubic contents or volume of an aerostat.

CAPTIVE BALLOON.—(*See* Balloon.)

CAQUOT BALLOON.—(*See* Balloon, kite.)

CAR.—The nacelle of an airship.

CEILING:

Absolute.—The maximum height above sea level which a given aircraft can approach asymptotically, assuming standard air conditions.

Service.—The height above sea level at which a given aircraft ceases to rise at a rate higher than a small specified one (100 feet per minute in United States Air Service). This specified rate may be different in the services of different countries.

CELL.—The entire structure of the wings and wing trussing on one side of the fuselage of an airplane, or between fuselages or nacelles, where there are more than one.

CENTER OF PRESSURE OF AN AEROFOIL SECTION.—The point in the chord of an aerofoil section, prolonged if necessary, through which at any given attitude the line of action of the resultant air force passes.

CHORD:

Of an aerofoil section.—The line of a straightedge brought into contact with the lower surface of the section at two points. In the case of an aerofoil having double convex camber the straight line joining the leading and trailing edges. (These edges may be defined, for this purpose, as the two points in the section which are farthest apart.) (Fig. 9.)

Length.—The length of the projection of the aerofoil section on its chord.

CHORD, MEAN, OF A WING.—The quotient obtained by dividing the wing area by the extreme dimension of the wing projection at right angles to the chord.

CHORD, MEAN, OF A COMBINATION OF WINGS.—If c be the mean chord of the combination c_1, c_2, c_3, etc., the mean chords of each wing corresponding to areas S_1, S_2, S_3, etc., then

$$c=\frac{c_1 S_1+c_2 S_2+c_3 S_3+ \quad . \quad . \quad . \quad .}{S_1+S_2+S_3+ \quad . \quad . \quad . \quad .}$$

CLIMB, RATE OF.—The vertical component of the air speed of an aircraft; i. e., its vertical velocity with reference to the air.

COCKPIT.—The open spaces in which the pilot and passengers are accommodated. A cockpit is never completely housed in.

CONCENTRATION RING:

Airship.—A metal ring to which several rigging lines are brought from the envelope and from which one or more lines also lead to the car.

Free balloon.—A hoop to which are attached the ropes suspending the basket and to which the net is also secured.

Parachute.—A hoop to which the rigging of the parachute is attached and also the line sustaining the passenger.

CONSUMPTION PER B. H. P. HOUR.—The quantity of fuel or oil consumed per hour by an engine running at ground level divided by the brake horsepower developed, unless specifically stated otherwise.

CONTROL COLUMN OR YOKE.—A control lever with a rotatable wheel mounted at its upper end. (*See* Control stick.) Pitching is controlled by fore-and-aft movement of the column; rolling, by rotation of the wheel. "Wheel control" is that type of control in which such a column or yoke is used.

CONTROL STICK.—The vertical lever by means of which certain of the principal controls of an airplane are operated. Pitching is controlled by a fore-and-aft movement of the stick, rolling by a side-to-side movement. "Stick control" is that type of control in which such a stick is used.

CONTROLS.—A general term applying to the means provided to enable the pilot to control the speed, direction of flight, attitude, and power of an aircraft.

CORD.—A species of wire made up of several strands (usually 7) twisted together as in a rope, each of the strands, in turn, being made up of several (usually 19) individual wires.

COWLING.—The metal covering which houses the engine and sometimes a portion of the fuselage or nacelle as well.

CRITICAL ANGLE.—(*See* Angle.)

CROSS-WIND FORCE.—The component perpendicular to the lift and to the drag of the total force on an aircraft due to the air through which it moves.

CROW'S-FOOT.—A system of diverging short ropes for distributing the pull of a single rope.

DAMPING FACTOR.—The percentage of damping in one period, i. e., $1=e-\lambda T$, where λT is the logarithmic decrement (q. v.).

DEAD LOAD.—(*See* Load.)

DIHEDRAL ANGLE.—(*See* Angle.)

DISK AREA.—The total area swept by a propeller, i. e., the area of a circle having a diameter equal to the propeller diameter.

DISCHARGEABLE WEIGHT.—The excess of the gross buoyancy over the dead load, the crew and such items of equipment as are essential to enable an airship to fly and land safely.

DIVE.—A steep glide.

DIVERGENCE.—A disturbance which increases without oscillation.

DOPE, AIRPLANE.—A general term applied to the material used in treating the cloth surface of airplane members to increase strength, produce tautness, and act as a filler to maintain air-tightness.

DOWNWASH ANGLE.—(*See* Angle.)

DRAG.—The component parallel to the relative wind of the total force on an aerofoil or aircraft due to the air through which it moves.

In the case of an airplane, that part of the drag due to the wings is called "wing resistance;" that due to the rest of the airplane is called "structural," or "parasite resistance."

DRAG ROPE.—The rope dropped by an airship in order to allow it to be secured by a landing party.

DRAG STRUT.—A compression member of the internal bracing system of an aerofoil.

DRAG WIRES.—(*See* Wires.)

DRIFT.—The angular deviation from a set course over the earth, due to cross currents of wind; hence, "drift meter."

DRIFT METER.—An instrument for the measurement of the angular deviation of an aircraft from a set course, due to cross winds.

DRIP FLAP.—A strip of fabric attached by one edge to the envelope of an aerostat so that rain runs off its free edge instead of dripping into the basket or car. The drip flap assists also to keep the suspension ropes dry and nonconducting.

DRY WEIGHT.—The weight of an engine, including carburetors, propeller-hub assembly, and ignition system, complete, but excluding exhaust manifolds.

DYNAMIC FACTOR.—The ratio between the load carried by any part of an aircraft when accelerating or when otherwise subjected to abnormal conditions and the load carried in normal flight.

DYNAMIC LIFT.—(*See* Lift.)

EFFECTIVE PITCH.—(*See* Pitch.)

ELEVATOR.—A movable auxiliary surface of an airplane, usually attached to the tail plane, the function of which is to impress a pitching moment on the aircraft. (Fig. 10.)

EMPENNAGE.—Same as Tail unit (q. v.).

ENVELOPE.—The outer covering of a rigid airship; or, in the case of a balloon or a nonrigid airship, the bag which contains the gas.

EQUATOR.—The largest horizontal circle of a spherical balloon.

FABRIC, BALLOON.—The finished material, usually rubberized, of which balloon or airship envelopes are made.

BIASED.—Plied fabric in which the threads of the plies are at an angle to each other.

PARALLEL.—Plied fabric in which the threads of the plies are parallel to each other.

FACTOR, DYNAMIC.—(*See* Dynamic factor.)

FACTOR OF SAFETY.—The ratio of the ultimate strength of a member to the maximum possible load occurring under conditions specified.

FAIRING.—A member whose primary function is to produce a smooth outline and to reduce head resistance or drag.

FINS.—Small stationary surfaces, substantially vertical, attached to different parts of aircraft, in order to promote stability; for example, tail fins, skid fins, etc. Fins are sometimes adjustable. (Fig. 10.)

SKID FINS.—Fore and aft vertical surfaces, usually placed well out toward the tips of the upper plane, designed to provide the vertical keel-surface required for stability.

FINS, KITE BALLOON.—The air-inflated lobes intended to keep the balloon headed into the wind.

FIRE WALL.—A metal plate, so set as to isolate from the engine the other parts of the airplane structure, and thus to reduce the risk from a backfire.

FITTING.—A generic term for any small metal part used in the structure of an airplane.

FLIGHT PATH.—The path of the center of gravity of an aircraft with reference to the earth.

FLOAT.—A completely inclosed water-tight structure attached to an aircraft in order to furnish it buoyancy when in contact with the surface of the water. In float seaplanes the crew is carried in a fuselage or nacelle separate from the float.

FLOAT SEAPLANE.—(*See* Seaplane.)

FLOTATION GEAR.—An emergency landing gear attached to an airplane, which will permit of safe landing on the water and provide buoyancy when resting on the surface of the water.

FLYING BOAT.—(*See* Seaplane.)

FREE-FLIGHT TESTING.—The conduct of special flight tests of a scientific nature, as contrasted with performance testing (q. v.).

FULL LOAD.—(*See* Load.)

FUSELAGE.—The elongated structure, of approximately streamline form, to which are attached the wings and tail unit of an airplane. In general it is designed to hold the passengers.

FUSELAGE, LENGTH OF.—The distance from the nose of the fuselage (including the engine bed and radiator, if present) to the after end of the fuselage, not including the control and stabilizing surfaces.

GAP.—The shortest distance between the planes of the chords of the upper and lower wings of a biplane, measured along a line perpendicular to the chord of the upper wing at any designated point of its entering edge. (Fig. 11.)

GEOMETRICAL PITCH.—(*See* Pitch.)

GLIDE, TO.—To descend at a normal angle of attack without engine power sufficient for level flight, the propeller thrust being replaced by a component of gravity along the line of flight.

GLIDER.—A form of aircraft similar to an airplane, but without any power plant. Gliders are used chiefly for sport.

GLIDING ANGLE.—(*See* Angle.)

GORE.—The portion of the envelope of a balloon or airship included between two adjacent meridian seams.

GROSS BUOYANCY.—(*See* Buoyancy.)

GROUND CLOTH.—Canvas placed on the ground to protect a balloon.

GROUND SPEED.—(*See* Speed.)

HANDLING TRUCK.—A truck, mounted on wheels or sliding on ways, on which airplanes or seaplanes may be placed to facilitate moving them about and carrying them to and from their hangars.

HANGAR.—A shelter for housing aircraft.

HELICOPTER.—A form of aircraft whose support in the air is derived from the vertical thrust of propellers.

HOG (AIRSHIP).—A distortion of the envelope in which the axis becomes convex upward or both ends droop.

HORN.—The operating lever of a control surface of an aircraft, e. g., aileron horn, rudder horn, elevator horn.

HORSEPOWER OF AN ENGINE, MAXIMUM.—The maximum horsepower which can be safely maintained for periods not less than five minutes.

HORSEPOWER OF AN ENGINE, NORMAL.—The highest horsepower which can be safely maintained for long periods.

HULL (AIRSHIP).—The main structure of a rigid airship, consisting of a covered elongated framework which incloses the gas bags and which supports the cars and equipment.

HULL (SEAPLANE).—The portion of a boat seaplane which furnishes buoyancy when in contact with the surface of the water, to which the main supporting surfaces and other parts are attached, and which contains accommodations for the crew.

INCIDENCE, ANGLE OF.—(*See* Angle.)

INCLINOMETER:

Absolute.—An instrument giving the attitude of an aircraft with reference to true gravity.

Relative.—An instrument giving the attitude of an aircraft with reference to apparent gravity. Such instruments are sometimes incorrectly referred to as banking indicators

INDICATOR, AIR-SPEED.—An anemometer mounted on an aircraft for the purpose of indicating the speed of the aircraft.

True air-speed indicator.—An instrument, usually working on the principle of the Biram or Robinson anemometers, which gives the true air speed, independent of density.

Apparent air-speed indicator.—An instrument, usually dependent on pressure measurements, the readings of which vary with the density of the air.

INDRAFT.—The drawing in of air from in front of a propeller by the action of the rotating blades. The indraft velocity relative to the propeller is somewhat higher than that of the undisturbed air at most points of the propeller disk.

INSPECTION WINDOW.—A small transparent window in the envelope of a balloon or in the wing of an airplane to allow inspection of the interior.

JACKSTAY.—A longitudinal rigging provided to maintain the correct distance between the heads of various riggings on an airship.

KEEL.—A member or assembly of members which provides longitidunal strength to an airship of rigid or semirigid type. In the case of a rigid airship the keel is usually an elaborately trussed girder and may be inclosed within the envelope or may project beyond (usually below) the regular cross-sectional form of the envelope.

Articulated.—A keel made up of a series of members hinged together at their ends.

KING POST.—The main compression member of a trussing system applied to a member subject to bending. (Fig. 4.)

KITE.—A form of aircraft without other propelling means then the towline pull, whose support is derived from the force of the wind moving past its surface.

KITE BALLOON.—(*See* Balloon.)

LAMINATED WOOD.—Wooden parts made up by gluing or otherwise fastening together individual wood planks or laminations with the grain substantially parallel.

LANDING ANGLE.—(*See* Angle.)

LANDING FIELD.—A field of such a nature as to permit of airplanes landing or taking off.

LANDING GEAR.—The understructure of an aircraft designed to carry the load when in contact with the land or water.

LEADING EDGE.—The foremost edge of an aerofoil or propeller blade.

LENGTH, CHORD.—(*See* Chord.)

LENGTH, FUSELAGE.—(*See* Fuselage.)

LENGTH, OVER-ALL.—(*See* Over-all.)

LIFT.—The component of the total air force which is perpendicular to the relative wind and in the plane of symmetry. It must be specified whether this applies to a complete aircraft or parts thereof. (In the case of an airship this is often called "dynamic lift.")

LIFT WIRES.—(*See* Wires.)

LOAD:

Dead.—The structure, power plant, and essential accessories of an aircraft. Included in this are the water in the radiator, tachometer, thermometer, gauges, air-speed indicators, levels, altimeter, compass, watch and hand starter, and also, in the case of an aerostat, the amount of ballast which must be carried to assist in making a safe landing.

Full.—The total weight of an aircraft when loaded to the maximum authorized loading of that particular type.

Useful.—The excess of the full load over the dead load of the aircraft itself. Therefore useful load includes the crew and passengers, oil and fuel, ballast, electric-light installation, chart board, detachable gun mounts, bomb storage and releasing gear, wireless apparatus, etc.

LOAD FACTOR.—The ratio of the ultimate strength of a member to the load under horizontal steady rectilinear flight conditions.

LOBES.—Inflated bags at the stern of an elongated balloon, designed to give it directional stability. Also used to denote the sections into which the envelope is sometimes (e. g., in the Astra-Torres) divided by the tension of the internal rigging.

LOGARITHMIC DECREMENT.—The natural logarithm of the ratio of two successive amplitudes of an oscillation; i. e., at an interval of one period. The general equation of an oscillation may be written

$$s = Ae^{-\lambda t} \sin (pt - \alpha),$$

in which A, λ, p, and α are constants. The amplitude of oscillation is $Ae^{-\lambda t}$. The phase of the vibration is $pt - \alpha$. The period is $2\pi/p$, and may be written T. It follows that the logarithmic decrement is λT. If λ is a positive number, the vibration is said to be "damped." [In an unstable oscillation (q. v.), the quantity λ is a negative number.]

LONGERON.—A fore-and-aft member of the framing of an airplane fuselage or nacelle, usually continuous across a number of points of support. (Fig. 12.)

LOOP.—An aerial maneuver in which the airplane describes an approximately circular path in the plane of the longitudinal and normal axes, the lateral axis remaining horizontal, and the upper side of the airplane remaining on the inside of the circle.

MAIN SUPPORTING SURFACE.—(*See* Surface.)

MARGIN OF POWER.—(*See* Power.)

MEAN CHORD OF A WING.—(*See* Chord.)

MEAN CHORD OF A COMBINATION OF WINGS.—(*See* Chord.)

MEAN SPAN.—(*See* Span, mean.)

MINIMUM SPEED.—(*See* Speed.)

MONOCOQUE.—A type of fuselage which is constructed by wrapping strips of veneer around formers, and in which the veneer is primarily depended on to carry stresses arising in the fuselage.

MONOPLANE.—A form of airplane which has but one main supporting surface extending equally on each side of the body.

MOORING HARNESS.—The system of bands of tape over the top of a balloon to which are attached the mooring ropes.

MULTIPLANE.—A form of airplane whose main supporting surface is divided into four parts, superimposed.

NACELLE.—The inclosed shelter for passengers or for a power plant. A nacelle is usually shorter than a fuselage, and does not carry the tail unit.

NET.—A rigging made of ropes and twine on spherical balloons which supports the weight of the basket, etc., distributing the load over the entire upper surface of the envelope.

NONRIGID AIRSHIP.—(*See* Airship.)

NOSE CAP.—A cap used to reinforce the bow stiffeners of an airship.

NOSE HEAVY.—The condition of an aircraft in which, in any given condition of normal flight, the nose tends to drop if the longitudinal control is released; i. e., the condition in which the pilot has to exert a pull on the control stick or column to maintain the given condition.

NURSE BALLOON.—(*See* Balloon.)

ORNITHOPTER.—A form of aircraft deriving its support and propelling force from flapping wings.

OSCILLATION, PHUGOID.—A long period oscillation characteristic of the disturbed longitudinal motion of an airplane.

OSCILLATION, STABLE.—An oscillation which tends to die out.

OSCILLATION, UNSTABLE.—An oscillation of which the amplitude tends to increase.

OVER-ALL LENGTH.—The distance from the extreme front to the extreme rear of an aircraft, including the propeller and the tail unit.

OVERHANG.—One-half the difference in the span of any two main supporting surfaces of an airplane. The overhang is positive when the upper of the two main supporting surfaces has the larger span. (Fig. 5.)

PANCAKE, TO.—To "level off" an airplane higher than for a normal landing, causing it to stall and descend with the wings at a very large angle of attack and approximately without bank, on a steeply inclined path.

PANEL AEROSTAT.—The unit piece of fabric of which the envelope of an aerostat is made.

PANEL AIRPLANE.—A portion of a wing of an airplane which is constructed entirely separately from the rest of the wing, and which is attached to the remainder by bolts and fittings.

PARACHUTE.—An apparatus used to retard the descent of a falling body by offering resistance to motion through the air; usually made of light fabric with no rigid parts.

PARASITE RESISTANCE.—(*See* Drag.)

PATCH, AIRSHIP.—A strengthened or reinforced flap of fabric, of variable form according to the maker, which is cemented to the envelope and forms an anchor by which some portion of the machine is attached to the envelope. (Fig. 2.)

PERFORMANCE.—The maximum and minimum speeds and rate of climb at various altitudes, the time to climb to these altitudes, and the ceiling constitute the performance characteristics of an airplane.

PERFORMANCE TESTING.—The process of determining the performance characteristics of an airplane.

PERIOD.—The time taken for a complete oscillation.

PERMEABILITY.—The measure of the rate of diffusion of gas through intact balloon fabric; usually expressed in cubic meters per square meter per 24 hours.

PHILLIPS' ENTRY.—A reversal of curvature of the lower surface of an aerofoil near the leading edge. The result is to decrease the drag and provide more depth for the front spar. (Fig. 9.)

PHUGOID OSCILLATION.—(*See* Oscillation.)

PILOT BALLOON.—(*See* Balloon.)

PITCH OF A PROPELLER:

Pitch, aerodynamic.—The distance a propeller would have to advance in one revolution in order that the torque might be zero.

Pitch, effective.—The distance an aircraft advances along its flight path for one revolution of the propeller.

Pitch, geometrical.—The distance an element of a propeller would advance in one revolution if it were turning in a solid nut; i. e., if it were moving along a helix of slope equal to the angle between the chord of the element and a plane perpendicular to the propeller axis. The mean geometrical pitch of a propeller, which is a quantity commonly used in specifications, is the mean of the geometrical pitches of the several elements.

PITCH OF A PROPELLER—Continued.

PITCH, STANDARD.—The "pitch of a propeller" is usually stated as the geometrical pitch taken at two-thirds of the radius.

PITCH, VIRTUAL.—The distance a propeller would have to advance in one revolution in order that there might be no thrust.

PITCH, ANGLE OF.—(*See* Angle.)

PITCH SLIP.—(*See* Slip.)

PITCH SPEED.—(*See* Speed.)

PITOT TUBE.—A tube with an end open square to a fluid stream. It is exposed with the open end pointing upstream to detect an impact pressure. It is usually associated with a coaxial tube surrounding it, having perforations normal to the axis for indicating static pressure; or there is such a tube placed near it and parallel to it, with a closed conical end and having perforations in its side. The velocity of the fluid can be determined from the difference between the impact pressure and the static pressure, as read by a suitable gauge. This instrument is often used to determine the velocity of an aircraft through the air. (Fig. 13.)

PLYWOOD.—A product formed by gluing together two or more layers of wood veneer.

POWER, MARGIN OF.—The difference between the power available at any given speed and in air of given density and the power required for level flight under the same conditions. The best rate of climb at any altitude depends on the maximum margin of power.

POWER LOADING.—The weight per horsepower, computed on a basis of full load and of power in air of standard density unless otherwise stated.

PRESSURE NOZZLE.—The apparatus which, in combination with a gauge, is used to measure the pressure due to speed through the air. Includes both Pitot and Venturi tubes. Pressure nozzles of various types are also used in yawmeters and other instruments.

PROOFING.—Material applied to the fabric of an aerostat at the time of manufacture to protect it against weather or to prevent the passage of gas.

PROPELLER, PUSHER.—A propeller which is placed at the rear end of its shaft and pushes against the thrust bearing.

PROPELLER, TRACTOR.—A propeller which is placed at the forward end of its shaft and pulls on the thrust bearing.

PURITY OF A GAS.—The percentage, by number of molecules, of the light gas used for inflation, such as hydrogen, to all the gases within the container.

PUSHER AIRPLANE.—(*See* Airplane.)

PUSHER PROPELLER.—(*See* Propeller.)

QUADRUPLANE.—A form of airplane whose main supporting surface is divided into four parts, superimposed.

RACE ROTATION.—The rotation of the air influenced by a propeller. This rotation is much more marked in the slip stream than in front of the propeller.

RAKE.—The cutting away of the wing tip at an angle so that the main supporting surfaces seen from above will appear of trapezoidal form. The amount of rake is measured by the angle between the straight portion of the wing-tip outline and the plane of symmetry. The rake is positive when the trailing edge is longer than the leading edge.

RAKE, BLADE.—The angle which the line joining the centroids of the sections of a propeller blade makes with a plane perpendicular to the propeller shaft. The rake is positive when the blades are thrown forward.

RATE OF CLIMB.—The vertical component of the air speed of an aircraft; i. e., its vertical velocity with reference to the air.

RATE-OF-CLIMB INDICATOR.—An instrument indicating the vertical component of the velocity of an aircraft. Most rate-of-climb meters depend on the rate of change of the atmospheric pressure.

RELATIVE WIND.—The motion of the air with reference to a moving body. Its direction and velocity, therefore, are found by adding two vectors, one being the velocity of the air with reference to the earth, the other being equal and opposite to the velocity of the body with reference to the earth.

RESISTANCE DERIVATIVES.—Quantities expressing the variation of the forces and moments on aircraft due to disturbance of steady motion. They form the experimental basis of the theory of stability, and from them the periods and damping factors of aircraft can be calculated. In the general case there are 18 translatory and 18 rotary derivatives.

Rotary.—Resistance derivatives expressing the variation of moments and forces due to small increases in the rotational velocities of the aircraft.

Translatory.—Resistance derivatives expressing the variation of moments and forces due to small increases in the translatory velocities of the aircraft.

REVERSE TURN.—A rapid maneuver to reverse the direction of flight of an airplane, made by a half loop and half roll in either sequence.

REVOLUTIONS, MAXIMUM.—The maximum number of revolutions per minute that may be maintained for periods not less than 5 minutes.

REVOLUTIONS, NORMAL.—The highest number of revolutions per minute that may be maintained for long periods.

RIB.—(*See* Wing rib.)

RIGGER.—One who is employed in assembling and aligning aircraft.

RIGGING.—The assembling and aligning of an aircraft.

RIGHT-HAND ENGINE.—An engine the final power delivery shaft of which rotates clockwise when viewed by an observer looking along the engine toward the power delivery end.

RIGHTING MOMENT.—A moment which tends to restore an aircraft to its previous attitude after any small rotational displacement.

RIGID AIRSHIP.—(*See* Airship.)

RIP CORD.—The rope running from the rip panel of a balloon or nonrigid airship to the basket, the pulling of which tears off the rip panel and causes immediate deflation.

RIP PANEL.—A strip in the upper part of a balloon or nonrigid airship which is torn off when immediate deflation is desired.

ROLL, ANGLE OF.—(*See* Angle.)

RUDDER.—A hinged or pivoted surface used for the purpose of impressing yawing moments on an aircraft; i. e., for controlling its direction of flight. (Fig. 10.)

RUDDER BAR.—The foot bar by means of which the rudder is operated.

RUDDER TORQUE.—The twisting effect exerted by the rudder on the fuselage, due to the relative displacement of the center of pressure of the rudder. The product of the rudder area by the distance from its center of area to the center line of the fuselage may be used as a relative measure of rudder torque.

SAFETY, FACTOR OF.—(*See* Factor of Safety.)

SAFETY LOOP.—A loop formed immediately outside the conical reversing bag through which the valve rope emerges from the bottom of an aerostat. Before the automatic valve can be opened by the aid of the valve rope the fastening of this safety loop is torn off by a strong pull on the valve rope from the nacelle.

SEAPLANE.—A particular form of airplane designed to rise from and land on the water.

Boat seaplane, or flying boat.—A form of seaplane having for its central portion a boat which provides flotation. It is often provided with auxiliary floats or pontoons. (Fig. 14.)

Float seaplane.—A form of seaplane in which the landing gear consists of one or more floats or pontoons. (Fig. 15.)

SEMIRIGID AIRSHIP.—(*See* Airship.)

SERPENT.—A short, heavy trail rope.

SHOCK ABSORBER.—A spring or elastic member, designed to prevent the imposition of large accelerations on the fuselage, wings, and other heavy concentrated weights. Shock absorbers are usually interposed between the wheels, floats, or tail skid, and the remainder of the airplane to secure resiliency in landing and taxi-ing.

SHOCK-ABSORBER HYSTERESIS.—The ratio of the work absorbed in the shock absorber during one complete cycle to the total energy transmitted to the shock absorber during the first half of the cycle.

SHUTTERS.—The adjustable blinds or vanes which are used to control the amount of air flowing through the radiator and so to regulate the temperature of the cooling water.

SIDE SLIPPING.—Sliding with a component of velocity along the lateral axis which is inclined and in the direction of the lower end of that axis. When it occurs in connection with a turn it is the opposite of skidding (q. v.).

SKID FINS.—(*See* Fins.)

SKIDDING.—Sliding sidewise away from the center of curvature when turning. It is usually caused by banking insufficiently and is the opposite of side slipping (q. v.).

SKIDS.—Runners used as members of the landing gear and designed to aid the aircraft in landing or taxi-ing.

TAIL SKID.—A skid used to support the tail when in contact with the ground.

WING SKID.—A skid placed near the wing-tip and designed to protect the wing from contact with the ground.

SKIN FRICTION.—The tangential component of the fluid force at a point on a surface. It depends on the viscosity and density of the fluid, the total surface area and the roughness of the surface of the object.

SLIP.—The difference between the effective pitch and the mean geometrical pitch. Slip is usually expressed as a percentage of the mean geometrical pitch.

SLIP STREAM.—The stream of air behind a propeller.

SOAR, TO.—To fly without engine power and without loss of altitude. Lightly loaded gliders will soar in rising currents of air.

SOUNDING BALLOON.—(*See* Balloon.)

SPAN, OR SPREAD.—The maximum distance laterally from tip to tip of an airplane inclusive of ailerons, or the lateral dimension of an aerofoil.

SPAN, MEAN, OF A COMBINATION OF WINGS.—If s be the mean span of the combination, s_1, s_2, and s_3, etc., the spans of each pair of wings separately corresponding to areas S_1, S_2, S_3, etc., then

$$s=\frac{s_1 S_1+s_2 S_2+s_3 S_3+ \; . \; . \; . \; .}{S_1+S_2+S_3+ \; . \; . \; . \; .}$$

SPEED:

AIR.—The speed of an aircraft relative to the air.

GROUND.—The horizontal component of the velocity of an aircraft relative to the earth.

SPEED, MINIMUM.—The lowest speed which can be maintained in level flight, with any throttle setting whatever.

SPEED, PITCH.—The product of the mean geometrical pitch by the number of revolutions of the propeller in unit time; i. e., the speed the aircraft would make if there were no slip.

SPIN.—An aerial maneuver consisting of a combination of roll and yaw, with the longitudinal axis of the airplane inclined steeply downward. The airplane descends in a helix of large pitch and very small radius, the upper side of the airplane being on the inside of the helix, and the angle of attack on the inner wing being maintained at an extremely large value.

SPINNER.—A fairing, usually made of sheet metal and roughly conical or paraboloid in form which is attached to the propeller boss and revolves with it.

SPIRAL INSTABILITY.—The instability on account of which an airplane tends to depart from straight flight, by a combination of side slipping and banking, the latter being always too great for the turn.

SPLICE (of a wooden member).—A joint of two or more pieces of wood in which one piece overlaps the other in such a manner as to maintain the strength.

SPREAD.—(*See* Span.)

STABILITY:

Static stability.—A machine is statically stable if, when slightly displaced by rotation about its center of gravity (as in wind tunnel experimentation), moments come into play which tend to return the machine to its normal attitude.

Dynamical stability.—A machine is dynamically stable if, when displaced from steady motion in flight, it tends to return to that steady state of motion.

In a general way, the difference between static stability and dynamical stability is that the former depends on restoring moments and the latter on damping factors.

Automatic.—Stability dependent upon movable control surfaces. The term "automatic stability" is usually applied to those cases in which the control surfaces are automatically operated by mechanical means.

Directional.—Stability with reference to rotations about the normal axis; i. e., a machine possessing directional stability in its simplest form is one for which N_v is negative. Owing to symmetry, directional stability is closely associated with lateral stability.

Inherent.—Stability of an aircraft due solely to the disposition and arrangement of its fixed parts; i. e., that property which causes it, when disturbed, to return to its normal attitude of flight without the use of the controls or the interposition of any mechanical device.

Lateral.—Stability with reference to disturbances involving rolling, yawing, or side-slipping; i. e., disturbances in which the position of the plane of symmetry of the aircraft is affected.

Longitudinal.—Stability with reference to disturbances in the plane of symmetry; i. e., disturbances involving pitching and variations of the longitudinal and normal velocities.

STABILIZER.—(*See* Tail plane.)

STABILIZER, MECHANICAL.—A mechanical device to stabilize the motion of an aircraft. Includes gyroscopic stabilizers, pendulum stabilizers, inertia stabilizers, etc.

STABLE OSCILLATION.—(*See* Oscillation.)

STAGGER.—The amount of advance of the entering edge of an upper wing of a biplane, triplane, or multiplane over that of a lower, expressed as percentage of gap. It is considered positive when the upper wing is forward and is measured from the entering edge of the upper wing along its chord to the point of intersection of this chord with a line drawn perpendicular to the chord of the upper wing at the entering edge of the lower wing, all lines being drawn in a plane parallel to the plane of symmetry. (Fig. 11.)

STAGGER WIRES.—(*See* Wires.)

STALLING.—A term describing the condition of an airplane which from any cause has lost the relative air speed necessary for control.

STANDARD PITCH.—(*See* Pitch.)

STATIC THRUST.—The thrust developed by a propeller when the aircraft is held stationary on the ground.

STATION.—A term used to denote the location of framing attachment in a fuselage or nacelle (strut points in a trussed fuselage, bulkhead points in a veneer fuselage).

STATOSCOPE.—An instrument to detect the existence of minute changes of atmospheric pressure, and so of small vertical motions of an aircraft.

STAY.—A wire or other tension member; for example, the stays of the wing and body trussing.

STEP.—A break in the form of the bottom of a float or hull designed to assist in securing a dynamic reaction from the water.

STICK CONTROL.—(*See* Control stick.)

STRAND.—A species of wire made up of several individual wires twisted together. (There are usually 19 wires—a single wire as core, an inner layer of 6 wires, and an outer layer of 12.)

STREAMLINE.—The path of a small portion of a fluid, supposed continuous, commonly taken relative to a solid body with respect to which the fluid is moving. The term is commonly used only of such paths as are not eddying, but the distinction should be made clear by the context.

STREAMLINE FLOW.—The condition of continuous flow of a fluid, as distinguished from eddying flow.

STREAMLINE FORM.—A fair form intended to avoid eddying and to preserve streamline flow.

STRUT.—A member of a truss frame designed to carry compressive loads. For instance, the vertical members of the wing truss of a biplane (interplane struts) and the short vertical and horizontal member separating the longerons (q. v.) in the fuselage. (Figs. 1 and 12.)

STRUT, DRAG.—(*See* Drag strut.)

SURFACE.—An aerofoil used for sustentation or control or to increase stability. Applies to the whole member, and not to one side only.

Balanced.—A surface, such as a rudder, aileron, etc., part of which is in front of its pivot.

SURFACE, MAIN SUPPORTING.—A pair of wings, extending on the same level from tip to tip of an airplane; i. e., a triplane has three main supporting surfaces. The main supporting surfaces do not include any surfaces intended primarily for control or stabilizing purposes.

SUSPENSION BAND.—The band around a balloon or airship to which are attached the main bridle suspensions of the basket or car.

SUSPENSION BAR.—The bar used for the concentration of basket suspension ropes in captive balloons.

SWEEP BACK.—The angle, measured in a plane parallel to the lateral axis and to the chord of the main planes, between the lateral axis of an airplane and the entering edge of the main planes. (Fig. 16.)

TAIL BOOM.—A spar or outrigger connecting the tail surfaces and main supporting surfaces Usually used on pushers. (Fig. 3.)

TAIL CUPS.—A steadying device attached by lines at the rear of certain types of elongated captive balloons. Somewhat similar to a sea anchor. (Fig. 17.)

TAIL DROOP.—A deformation of the airship in which the axis bends downward at the after end.

TAIL HEAVY.—The condition of an aircraft in which, in any given condition of normal flight the nose tends to rise if the longitudinal control is released; i. e., the condition in which the pilot has to exert a push on the control stick or column to maintain the given condition.

TAIL PLANE.—A stationary horizontal, or nearly horizontal, tail surface, used to stabilize the pitching motion. Often called "stabilizer." (Fig. 10.)

TAIL SETTING, ANGLE OF.—(*See* Angle.)

TAIL SKID.—(*See* Skids.)

TAIL SLIDE.—The rearward motion which certain airplanes may be made to take after having been brought into a stalling position.

TAIL UNIT.—The tail surfaces of an aircraft.

TANDEM AIRPLANE.—(*See* Airplane.)

TAXI, TO.—To run an airplane over the ground, or a seaplane on the surface of water, under its own power.

TOGGLE.—A short crossbar of wood or metal, having a shouldered groove, which is fitted at the end of a rope at right angles to it. It is used for obtaining a quickly detachable connection with an eye at the end of another rope. (Fig. 18.)

TRACTOR AIRPLANE.—(*See* Airplane.)

TRACTOR PROPELLER.—(*See* Propeller.)

TRAIL ROPE.—The long trailing rope attached to a spherical balloon, to serve as a brake and as a variable ballast.

TRAILING EDGE.—The rearmost edge of an aerofoil or propeller blade.

TRAJECTORY BAND.—A band of webbing carried in a curve over the top of the envelope of an airship to distribute the stresses due to the suspension. The use of trajectory bands was introduced in the Parseval airships. (Fig. 19.)

TRIPLANE.—A form of airplane whose main supporting surface is divided into three parts, superimposed.

TURN INDICATOR.—An instrument showing when the direction of the line of flight or the direction of the projection of that line on a horizontal plane is altering, and in its more refined forms, giving the rate of turn, in terms either of the angular velocity or of the radius of curvature.

UNSTABLE OSCILLATION.—(*See* Oscillation.)

USEFUL LOAD.—(*See* Load.)

VALVE, AUTOMATIC.—(*See* Automatic Valve.)

VENEER.—Thin sheets or strips of wood.

VENTURI TUBE.—A short tube with flaring ends and a constriction between them, so that, when fluid flows through it, there will be a suction produced in a side tube opening into the constricted throat. This tube, when combined with a Pitot tube or with one giving static pressure, forms a pressure nozzle, which may be used as an instrument to determine the speed of an aircraft through the air. (Fig. 21.)

VIRTUAL PITCH.—(*See* Pitch.)

WARP, TO.—To change the form of a wing by twisting it. Warping is sometimes used to maintain the lateral equilibrium of an airplane.

WASH.—The disturbance in the air produced by the passage of an aerofoil.

WASHIN.—A permanent increase in the angle of attack near the tip of the wing.

WASHOUT.—A permanent decrease in the angle of attack near the tip of the wing.

WEIGHT, DISCHARGEABLE.—(*See* Dischargeable Weight.)

WEIGHT, DRY.—(*See* Dry Weight.)

WEIGHT PER HORSEPOWER.—The dry weight of an engine divided by the normal horsepower developed at ground level.

WHEEL CONTROL.—(*See* Control Column.)

WIDTH RATIO, TOTAL (PROPELLER BLADE).—The product of blade width ratio by number of blades.

WIND, RELATIVE.—(*See* Relative Wind.)

WIND TUNNEL.—An elongated inclosed chamber, including means for the production of a substantially steady air current through the chamber. Models of aircraft or other objects are supported in the center of the airstream and their resistance and other characteristics when exposed to an air current of known velocity are determined. The term includes those laboratories in which, as in the Eiffel type, there is an experimental chamber of much larger cross-section than the air current.

WINDMILL.—A small air-driven turbine with blades similar to those of a propeller exposed on an aircraft, usually in the slip stream, and used to drive such auxiliary apparatus as gasoline pumps and radio generators.

WINDOW, INSPECTION.—(*See* Inspection window.)

WING.—The portion of a main supporting surface of an airplane on one side of the plane of symmetry; e. g., a biplane has four wings.

WING LOADING.—The weight carried per unit area of supporting surface. The area used in computing the wing loading should include the ailerons, but not the tail plane or elevators.

WING RESISTANCE. (*See* Drag.)

WING RIB.—A fore-and-aft member of the wing structure of an airplane, used to give the wing section its form and to transmit the load from the fabric to the spars. (Fig. 20.)

RIB, COMPRESSION.—A heavy rib designed to have the above functions and also to act as a strut opposing the pull of the wires in the internal drag truss. (Fig. 20.)

RIB, FORM.—An incomplete rib, frequently consisting only of a strip of wood extending from the leading edge to the front spar, which is used to assist in maintaining the form of the wing where the curvature of the aerofoil section is sharpest. (Fig. 20.)

WING SKID.—(*See* Skids.)

WING SPARS.—The principal transverse structural elements of the wing assembly of an airplane. The load is transmitted from the ribs to the spars, and thence to the lift and drag trusses. (Fig. 20.)

WING TRUSS.—The framing by which the wing loads of an airplane are transmitted to the fuselage; comprises struts, wires, or tie-rods, and spars.

WIRE.—In aeronautics refers specifically to hard-drawn solid wire.

WIRES, ANTIDRAG.—Wires designed primarily to resist forces acting parallel to the planes of the wings of an airplane and in the same direction as the direction of flight.

WIRES, ANTILIFT.—Wires in an airplane intended mainly to resist forces in the opposite direction to the lift, and to oppose the lift wires and prevent distortion of the structure by overtightening of those members.

WIRES, DRAG.—All wires designed primarily to resist forces acting parallel to the planes of the wings of an airplane and opposite to the direction of flight.

INTERNAL DRAG WIRES are concealed inside the wings.

EXTERNAL DRAG WIRES run from the wing cell to the nose of the fuselage or some other part of the machine.

WIRES, LIFT.—The wires which transmit the lift on the outer portion of the wings of an airplane in toward the fuselage or nacelle. These wires usually run from the top of an interplane strut to the bottom of the strut next nearer the fuselage.

WIRES, STAGGER.—Wires connecting the upper and lower surfaces of an airplane, and lying in planes substantially parallel to the plane of symmetry.

YAW, ANGLE OF.—(*See* Angle.)

YAWING.—Angular motion about the normal axis.

YAWMETER.—An instrument giving by direct reading the angle of yaw.

YOKE.—(*See* Control column.)

ZERO LIFT ANGLE.—The angle between the chord and the relative wind when the lift is zero.

ZERO LIFT LINE.—The position in the plane of an aerofoil section of the line of action of the resultant air force when the position of the section is such that the lift is zero.

ZOOM, TO.—To climb for a short time at an angle greater than that which can be maintained in steady flight, the machine being carried upward at the expense of its stored kinetic energy. This term is sometimes used by pilots to denote any sudden increase in the upward slope of the flight path.

REPORT No. 91.

NOMENCLATURE FOR AERONAUTICS.

By the National Advisory Committee for Aeronautics.

PART III.

AERONAUTICAL SYMBOLS.

	Symbol.
Height	h
Propeller diameter	D
Angle of attack	α
$\mathrm{Tan}^{-1}\ D/L$	γ
Dihedral	Γ
Angle of downwash	ϵ
Angle of tail setting	β
Propeller helix angle	ϕ
True air speed	V
Indicated air speed	V_i
Lift	L
Drag	D
Cross-wind force	C
Longitudinal force	X
Lateral force	Y
Normal force	Z
Rolling moment	L
Pitching moment	M
Yawing moment	N

Note.—In dealing with stability theory, X, Y, Z, L, M, and N are commonly referred to the forces and moments per unit mass.

Propeller thrust	T
Propeller torque	Q
Power	P
Mass of machine $=\dfrac{W}{g}$	m
Total weight	W
Propeller efficiency	π

	Symbol.
Air density (mass per unit volume)	ρ
Kinematic viscosity	ν
Area	S
Angle of yaw	Ψ
Angle of pitch	θ
Angle of roll	ϕ
Component of velocity parallel to the X-axis and relative to the undisturbed air	u
Component of velocity parallel to the Y-axis and relative to the undisturbed air	v
Component of velocity parallel to the Z-axis and relative to the undisturbed air	w
Angular velocity of roll	p
Angular velocity of pitch	q
Angular velocity of yaw	r
Moments of inertia about the X, Y, and Z axes, respectively	A, B, C
Products of inertia with respect to the Y and Z, X and Z, and X and Y axes, respectively	D, E, F
Radii of gyration about the X, Y, and Z axes, respectively	K_A, K_B, K_C
Logarithmic increment or decrement of amplitude. $(\theta=\theta_0 e^{-\lambda t})$	λ

COEFFICIENTS.

Coefficients of forces and moments ("absolute" coefficients to be used in all cases):

L_c, D_c, C_c, X_c, Y_c, Z_c, L_c, M_c, N_c.

Propeller thrust coefficients:

$$T_c=\frac{T}{\rho N^2 D^4},\quad T_c'=\frac{T}{\rho V^2 D^2},\quad T_c''=\frac{T}{\rho\,\frac{V^4}{N^2}};$$

where N = Revolutions per minute.

Propeller torque coefficients:

$$Q_c=\frac{Q}{\rho N^2 D^5},\quad Q_c'=\frac{Q}{\rho V^2 D^3},\quad Q_c''=\frac{Q}{\rho\,\frac{V^5}{N^3}}.$$

FIG. 2.

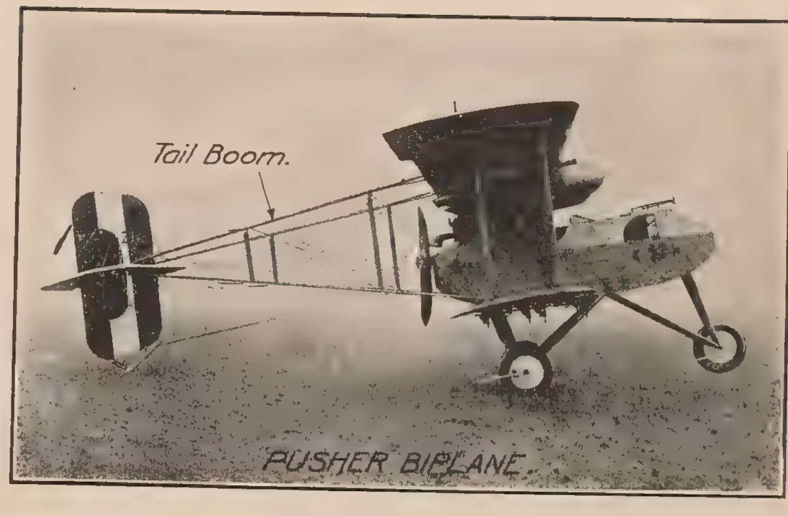

FIG. 3.

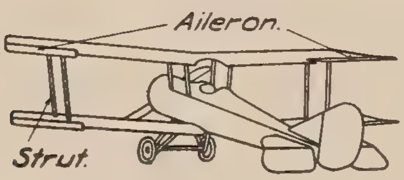

FIG. 1.

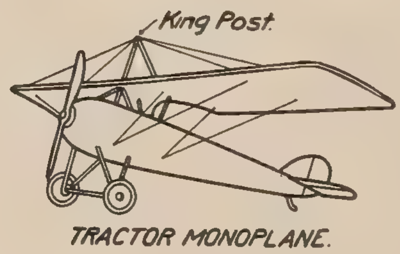

FIG. 4.

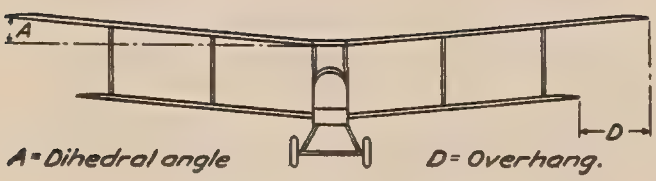

FIG. 5.

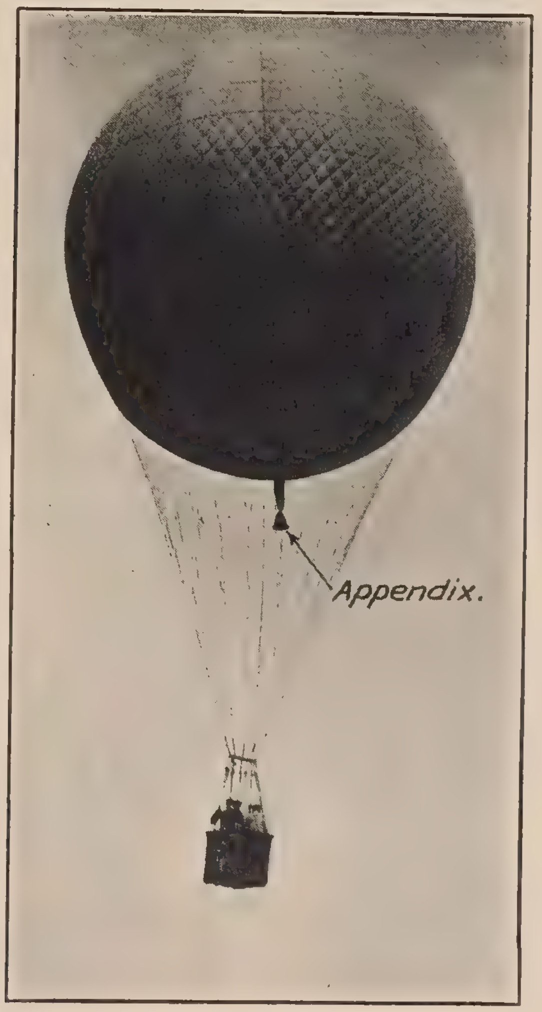

FIG. 6.

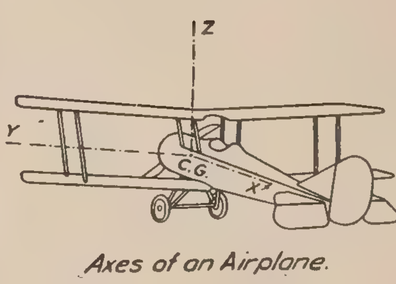

FIG. 7

FIG. 8.

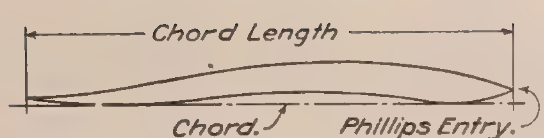

FIG. 9.

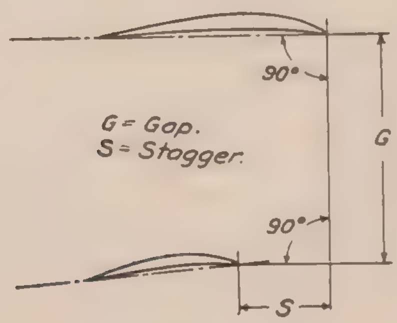

FIG. 11.

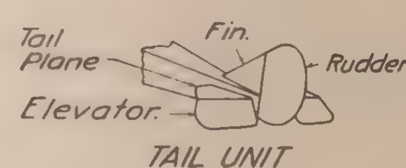

FIG. 10.

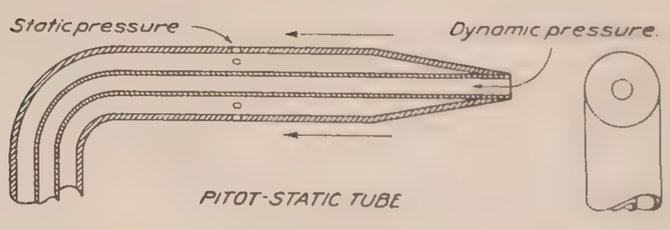

FIG. 13.

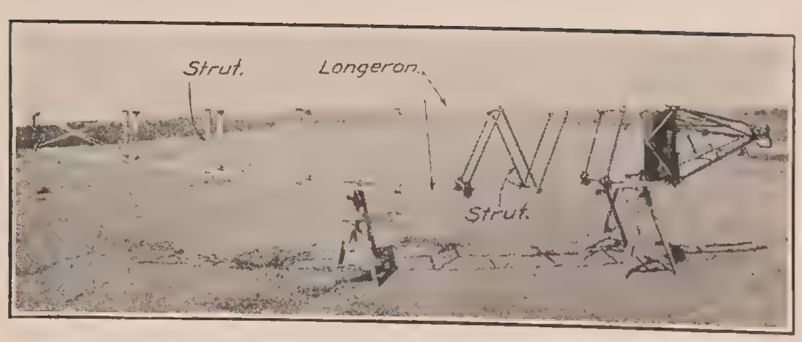

FIG. 12.

FIG. 14.

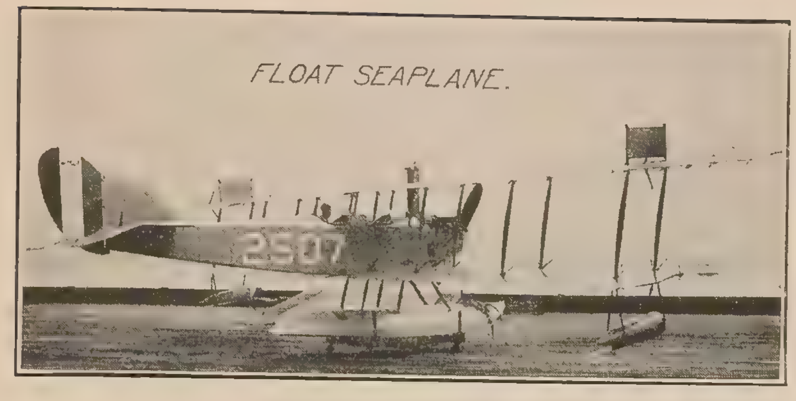

FIG. 15.

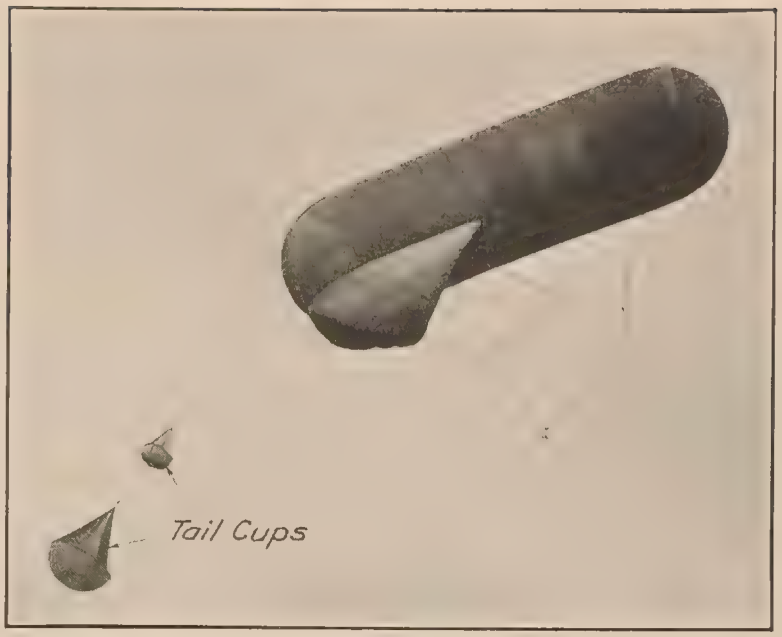

FIG. 17.

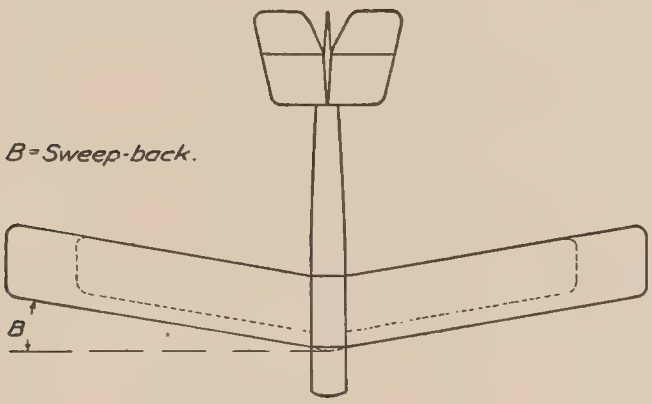

FIG. 16

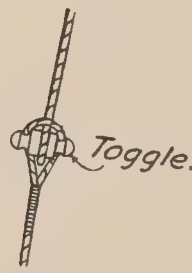

FIG. 18.

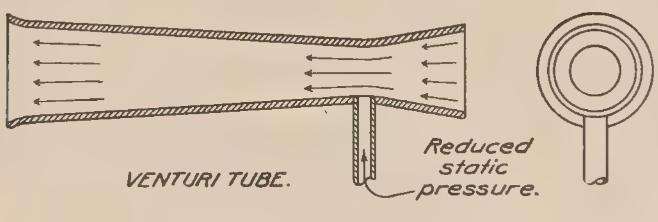

FIG. 21.

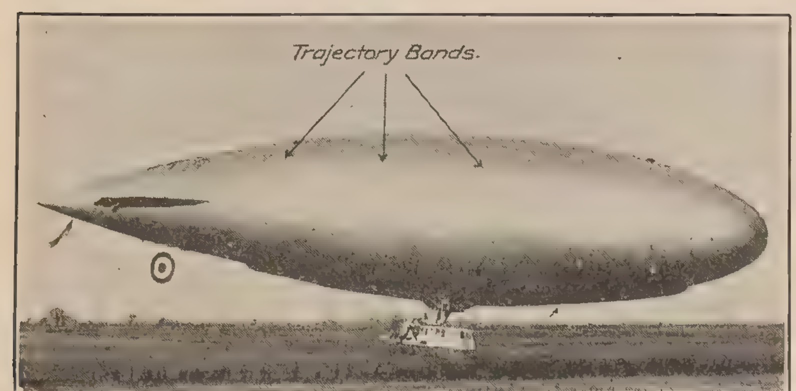

FIG. 19.

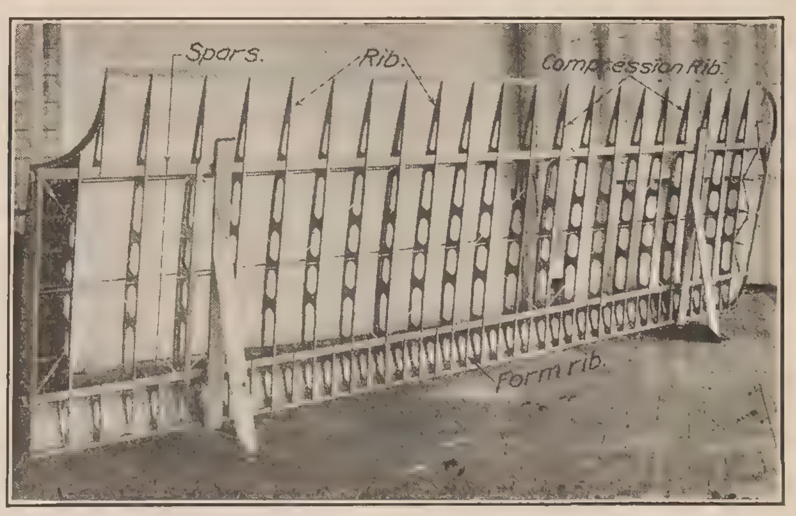

FIG. 20.

Propeller power coefficients:

$$P_c=\frac{P}{\rho N^3D^5},\ P_c'=\frac{P}{\rho V^3D^2},\ P_c''=\frac{P}{\rho\frac{V^5}{N^2}},\ P_c'''=\frac{P}{\rho VN^2D^4}.$$

DERIVATIVES.

$\left.\begin{matrix} X_u & Z_u & M_u \\ X_w & Z_w & M_w \\ X_q & Z_q & M_q \end{matrix}\right\}$ Symmetric resistance derivatives.

$\left.\begin{matrix} Y_v & L_v & N_v \\ Y_p & L_p & N_p \\ Y_r & L_r & N_r \end{matrix}\right\}$ Asymmetric resistance derivatives.

$\left.\begin{matrix} Y_u & L_u & N_u \\ Y_w & L_w & N_w \\ Y_q & L_q & N_q \end{matrix}\right\}$ Combined resistance derivatives which disappear in a symmetrical aircraft.

$\left.\begin{matrix} X_v & Z_v & M_v \\ X_p & Z_p & M_p \\ X_r & Z_r & M_r \end{matrix}\right\}$ Combined resistance derivatives which do not disappear but are generally neglected.

REPORT No. 92

ANALYSIS OF WING TRUSS STRESSES

INCLUDING

THE EFFECT OF REDUNDANCIES

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REPORT No. 92.

ANALYSIS OF WING TRUSS STRESSES.

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This report was prepared at the Langley Memorial Aeronautical Laboratory of the National Advisory Committee for Aeronautics under the direction of the Committee on Aerodynamics by Edward P. Warner and Roy G. Miller.

It has been the usual practice of airplane designers in making structural analyses to treat the airplane, not as a collected whole, but as an assemblage of separate units, and to carry through an analysis for each of these units in turn, ignoring members wherever necessary in order that the structure of each separate unit may be statically determinate. In wing truss analysis, for example, it is the invariable practice in making routine analyses to entirely ignore the effect of the stagger wires and the external drag wires, the forces acting on the truss being resolved into the planes of the lift bracing and the internal drag bracing and these bracing systems being designed strongly enough to carry the entire loads. When the stagger wires are taken into consideration at all, it is only on the assumption that the flying wire has been shot away and that the load must be carried from one lift truss to the other through the stagger wires. Obviously the members thus ignored will come into play under some conditions, and, in so doing, they will affect the stresses in the other members which are ordinarily taken into account. It is customary to fall back on the assertion that the ordinary method of analysis is on the safe side, but reliance on such a claim is always unscientific and unsatisfactory, and nowhere more so than in airplane design, where the loads acting are all dependent on the weight of the structure, and where it is therefore almost as undesirable to have one unit or group of members too strong and heavy relatively to the other members as to have one member too weak, since the excessive strength and weight of one increases the loads and stresses in all others. It is therefore eminently desirable that the analysis of the airplane structure should be carried through with the greatest possible refinement of detail, and that nothing should be left to guesswork or chance where it can be avoided.

To take one of the simplest cases as an illustration, it is evident that when an airplane is diving and the center of pressure is far to the rear of the rear spars the load on the rear truss will act upward and that on the front truss downward. If there were no restraint on the relative motion of the two systems of bracing the rear truss would therefore rise while the front one descended below its normal level, and the form would be distorted at each panel point, the truss being so warped as to decrease the angle of attack along the wing and to decrease the stagger near the tips of the wing. The physical reasoning on this point has been given at some length by Mr. John Case.[1] Other points at which there is uncertainty are the external drag wires, already alluded to, and the interaction between the fuselage and wings. The latter point was taken up in a recent report of the National Advisory Committee for Aeronautics,[2] but the analysis was not carried through in full and certain rough assumptions were made as to the tensions in the external drag wires.

The standard method of treating redundant members and statically indeterminate structures in general is furnished by the method of least work, originated by Castigliano. This method is commonly used in bridge design, and has found some application in other departments of engineering, but very little attempt has as yet been made to apply it to the needs of aeronautics.

The general means of application of the method of least work will be found discussed in any textbook on structures.[3] The application to airplanes has been briefly and simply discussed in

[1] The Importance of Incidence Wires in Strength Calculations, "Aeronautics," December 4, 1918.
[2] Fuselage Stress Analysis, by E. P. Warner and R. G. Miller, Report No. 76, National Advisory Committee for Aeronautics, Washington, 1920.
[3] The Theory of Structures, by C. M. Spofford, Chapter XVI, New York, 1915. Mechanics of Internal Work, by Church, New York, 1910.

Pippard and Pritchard's recent work on airplane structures (London, 1919), and Mr. Case,[4] in an extension of the article mentioned above, has treated mathematically the theory of the effect of incidence wires by this method, but a great deal of work on the subject remains to be done. The method pursued in this report is somewhat similar to that followed in the previous report on fuselage stresses. A representative airplane is chosen and the analysis carried through both with and without consideration of the redundancies for a number of different systems of loading, in order to give a concrete idea of the importance of the stagger wires and external drag wires and of the magnitude of the error involved by failing to take them into consideration when analyzing the stresses in an airplane of conventional type. The stresses in each member for the various conditions of loading have then been tabulated. The airplane chosen as an illustrative example closely resembles the JN4H, it being probable that more Americans are familiar with the general characteristics of this type than with any other. Assembly drawings of this airplane are given in figure 1.

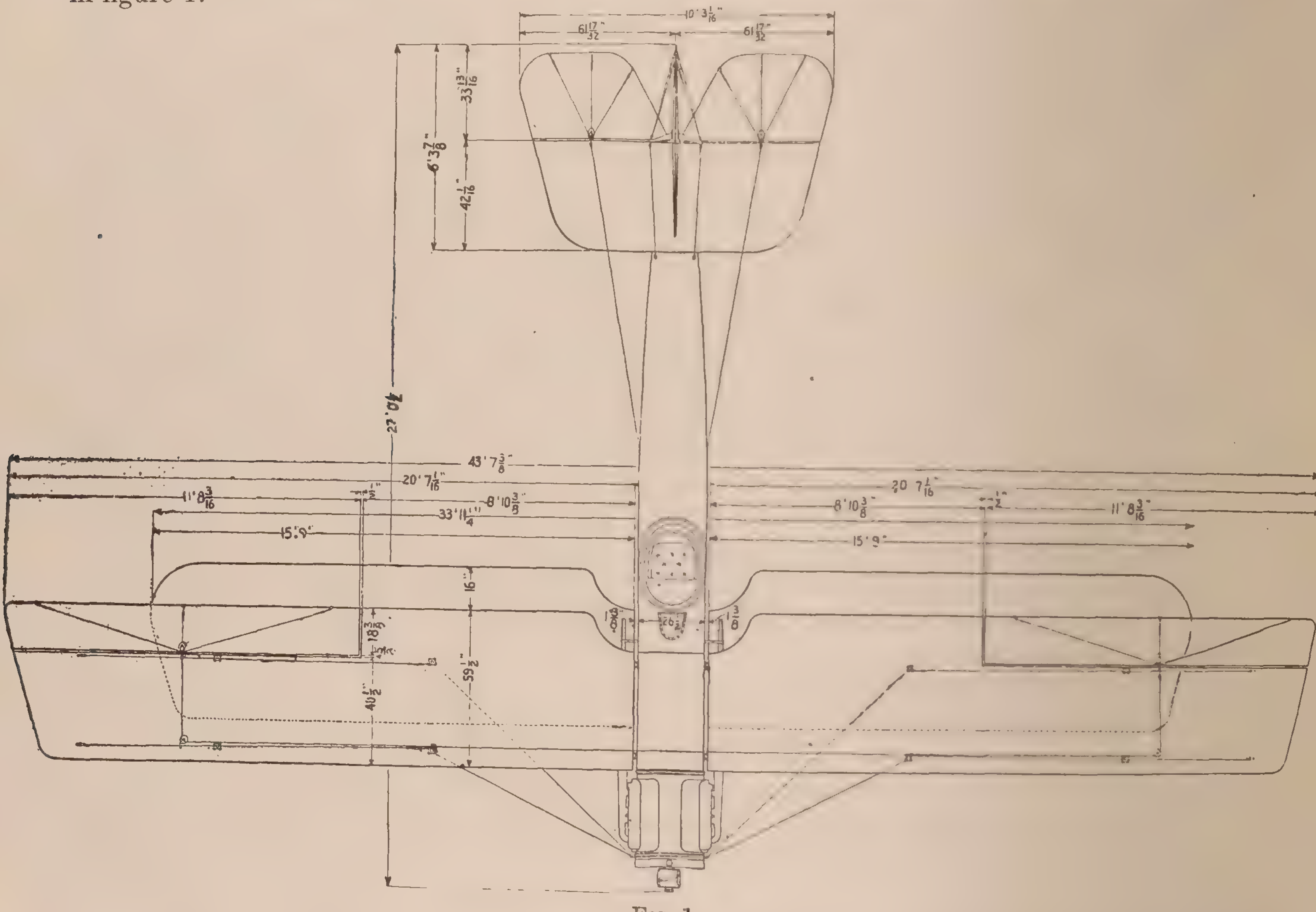


FIG. 1.

The method of least work is really nothing more than a simple method of analyzing the geometry of a structure. It is obvious that if a structure would deflect under load, and any particular redundant tension member were absent, in such a manner as to increase the distance between the points at which the ends of that redundant member are actually attached, the redundant member will resist and reduce the deflection and will modify the strains in the other members and the distribution of load among them. Castigliano's theorem offers an easy and straightforward route to the determination of this new distribution, which could otherwise be found only by a tedious process of trial and error. There are certain points which make it very difficult to apply the method of least work to airplane structures in the normal manner. The end fixation of the members is uncertain, there being an initial yield in the terminals and fittings which it is usually impossible to take into theoretical consideration. Furthermore, the stresses acting on some of the members are a combination of bending and direct end loading, and it would

[4] Incidence Wires in the Strength Calculation of Wind Structures, "Aeronautics," December 18 and 25, 1918, and January 1 and 8, 1919.

be extremely difficult to take full account of the effects of both types of stress. It appears probable that it will be safe in least work analyses of the wing structure to ignore the wooden members entirely. The tensile strength of airplane wire is about 200,000 pounds per square inch, and its modulus of elasticity is about 30,000,000 pounds per square inch. The strength of spruce in direct compression is, on the other hand, about 4,500 pounds per square inch and its modulus of elasticity is about 1,600,000 pounds per square inch. If all the members were perfectly elastic up to their ultimate strength, the strain per unit length at the instant before rupture would be a little less than one-half as great for spruce as for wire. Furthermore, the unit stress in the spruce members is always a much smaller proportion of the ultimate strength than is that in wires, because most of the wooden members are long columns of a small sectional radius of gyration, and the unit stress must therefore be kept low in order that failure may not occur by buckling. Since the work done in stressing a member depends largely on the strain imposed, being directly proportional to strain for a given stress, it is clear that the work done in stressing the wooden members will be much less than that done on the wires, and that the effect on total work and its derivatives of any change in the stress in the wooden members will therefore be relatively slight. Reliance has not, however, been placed solely on this approximate physical reasoning. An analysis has been carried through for one type of loading, taking the wooden members fully into account so far as their end loads are concerned, and the tabulation of results shows, as has just been predicted, that the effect of the wooden members is small enough to be neglected under ordi-

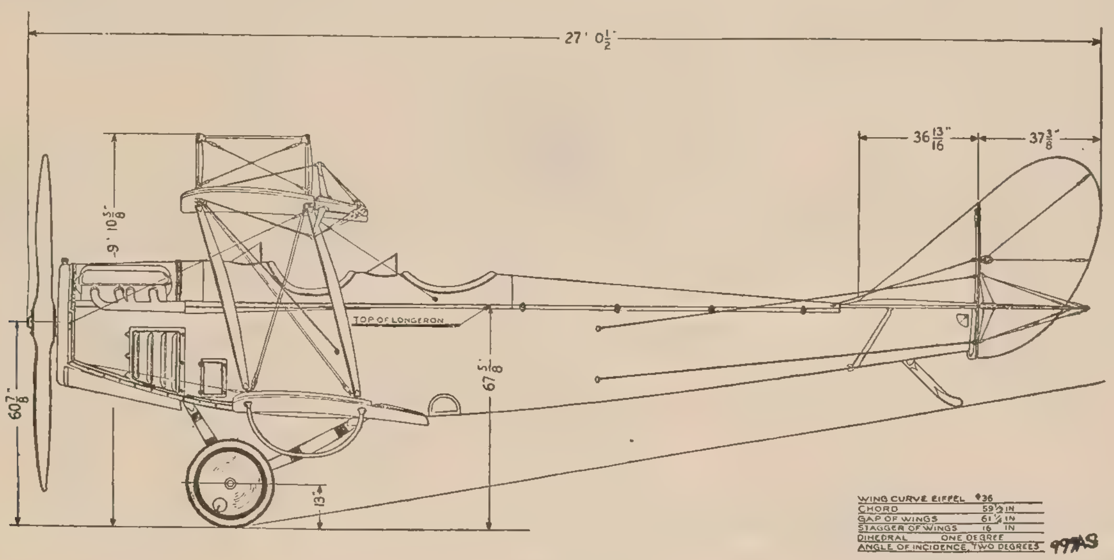


FIG. 2.

nary circumstances. The comparative analyses with and without consideration of the spars and struts will be fully discussed in their proper place.

Another question which has a considerable effect on the stress when there are redundant members is that of initial tension in the wires, and the uncertainty prevailing as to the initial tension is often used as an argument against the undertaking of further refinement of the methods of stress analysis. There is some justice in this argument for, as will be shown later, the initial tension does vary widely between different members in the same airplane and between corresponding members in different machines. To show what the maximum effect of initial tension is likely to be, an analysis has been carried through with the maximum probable initial tension in each wire.

In the application of the method of least work to aeronautical structures there arises a problem not so often encountered in the design of indeterminate bridge structures, in that some of the members are capable only of taking tension. It is necessary, then, to make some assumption in starting the analysis as to which one of an opposed pair of tension members will be in tension when all the loads are acting, and then to carry the analysis through, disregarding entirely the members opposed to those which are believed to carry tension in the final result and treating the working members for the moment as though they could take either tension or compression. If, however, the final result shows a compression in a wire, it is necessary to repeat the whole analysis with the opposing wire taken into consideration throughout in place of the one

which had a stress of opposite sign to that expected. It is usually possible, after a little practice, to guess which wire of any pair will carry tension, and the trial and error method just outlined therefore does not often have to be invoked.

The method of least work is essentially a check method. It can not be used for initial calculations, as it is necessary to know the sizes of all the members before the work equations can be written. In this respect it is like the "Berry method" of wing spar analysis by the generalized equation of three moments.

The cases treated in this report are five in number, two of them relating to loadings experienced in the air and the other three to comparison with other types of analysis and to the effects of modifying factors. The loadings considered are those experienced at a high angle of attack and a high speed, as in pulling out of a dive abruptly and in a vertical dive at limiting speed. The other three cases deal with the effect of wooden members, the effect of initial tension, and with the determination of the stresses encountered when the structure is loaded in accordance

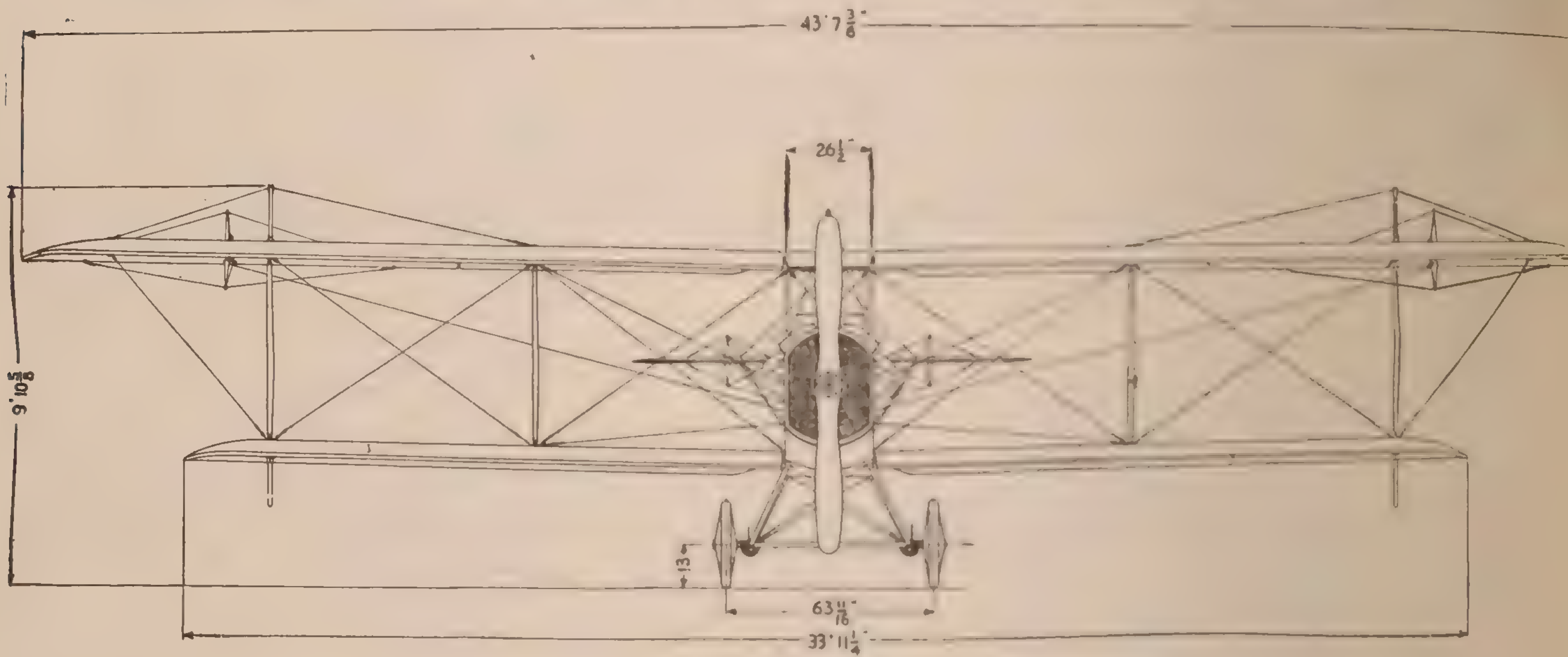


FIG. 3.

with a suggested set of specifications for static testing recently drawn up by the staff of the National Advisory Committee for Aeronautics.

CASE I.

As a first application of the analysis, the airplane was assumed to flatten out of a dive very abruptly, so that the angle of attack reached 12° in combination with a speed of 100 miles per hour. The total air load under these conditions is 5.43 times the weight of the airplane. Accelerometer tests on pursuit airplanes, conducted at the Royal Aircraft Establishment, have never shown a dynamic load factor in excess of 4.2 under the most violent handling, and ordinary stunting does not impose loads in excess of three times the weight. The conditions assumed are therefore at least as severe as any that would ever be encountered in flattening out of a dive.

A perspective view of the left wing truss, with every wire numbered, is shown in figure 2. The first step in the analysis is, as already pointed out in the introduction, to determine which wires are placed in tension by the loads being considered, as all wires which do not carry tension must be disregarded entirely. The possible redundancies are the stagger wires (not more than one at each panel acting in any given case), the two external drag wires 20 and 21, and the landing wires in the inner bay, 16′ and 17′. It is possible for the landing and flying wires to be stressed at the same time, even though there is no initial tension anywhere, as the center section struts can carry no tension and the lift reaction on the upper wing at the center section may be carried in whole or in part by the landing wire, being transmitted thence to the fuselage through the inner interplane strut and the inner flying wire. Since the point of attachment of the lower end of the landing wire is itself deflected upward by the normal lift load, that wire will not take the

center section reaction if there is any other member capable of carrying it in a reasonably direct fashion. This is the case in the front truss of the airplane here analyzed, as the wire 18 carries the reaction. The rear truss, however, is supported at the center section only by the wire 19, which runs so obliquely that a relatively large vertical deflection of the upper rear spar at the center section would ensue if there were no other restraining member. If this deflection proceeds far enough, the rear landing wire comes into play, and it is therefore necessary to take this wire into account as one of the redundancies.

As for the two external drag wires, No. 20, which runs downward and forward from the rear upper spar, is obviously in tension, as the upward deflection of the lift truss and the rearward deflection of the drag truss both act to extend that wire. No. 21, while it is extended by the deflection of the drag truss, is so much shortened by the much larger movement of the lift truss that it carries no tensile stress, and is therefore disregarded. There are, then, four redundancies in all, including the two stagger wires which are acting. One stagger wire at each panel point always comes into play, but the load may shift from one diagonal to the other as the type of loading changes. In the particular case under consideration it is the long diagonal, running downward from front to rear, which acts at both panel points, chiefly because the front lift truss carries more load than the rear, the center of pressure being far forward, and conse-

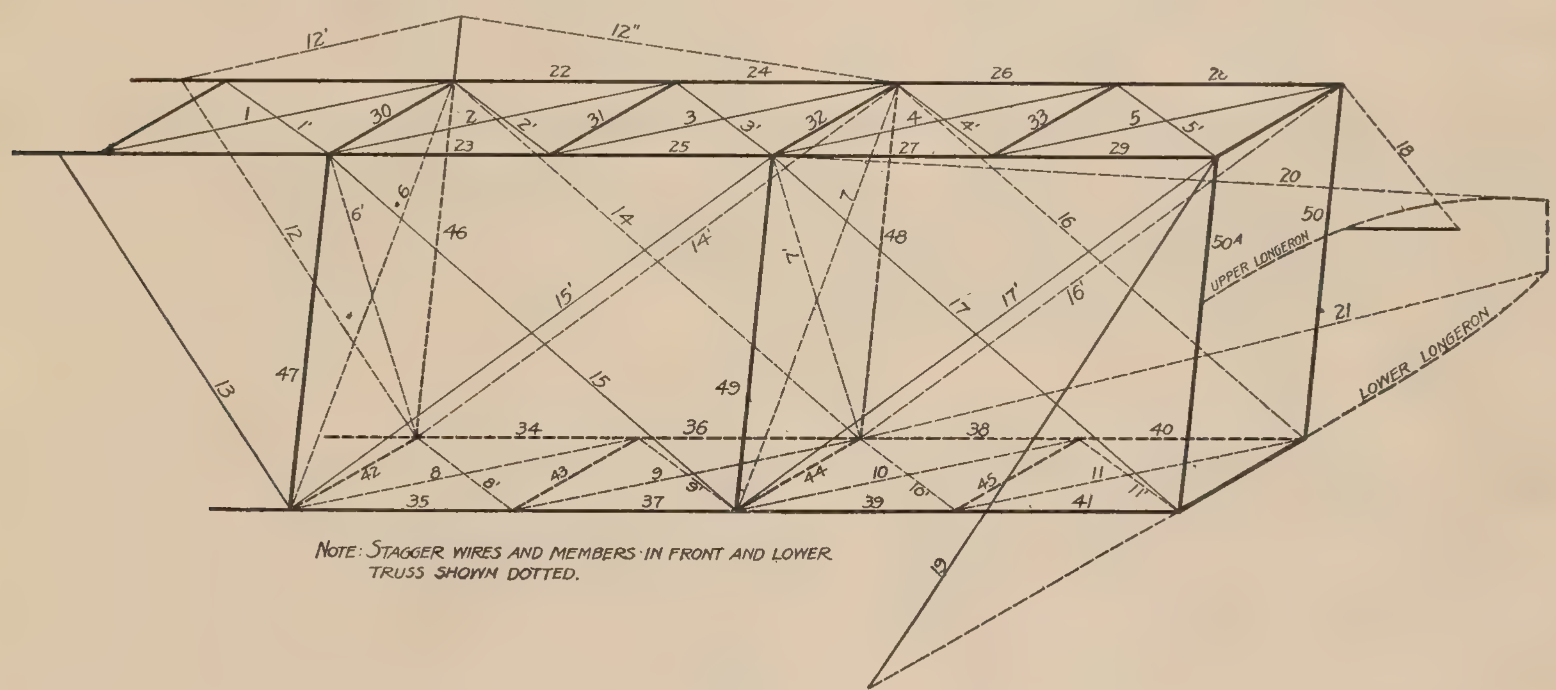


Fig. 4.

quently has a larger deflection. The long stagger wire accordingly comes into play to equalize the deflections. The distribution of the drag load also acts to stress the same wire, as the wire No. 20 acts as a partial support for the upper wing at the inner panel point, and the length of the portion of the lower wing which is cantilevered beyond its last support (not counting the stagger wires as supports) in respect of drag is therefore greater than the length of the corresponding portion of the upper wing. Part of the drag of the lower wing is therefore transferred to the upper and carried by it to the fuselage, instead of the reverse, which is generally assumed and which would hold true if it were not for the external drag wires.

The center section struts are incapable of sustaining any tension, and the reactions must therefore be taken, in the nonredundant analysis, by the wires 18 and 19. The horizontal components of the pulls in these wires combine with the center section drag truss reaction to produce an unbalanced force in the plane of the wing, and one of the center section struts must be thrown into compression to take the force. In the case under discussion at present, the unbalanced force being to the rear, the forward strut is in compression and the rear one is inoperative. The tension in 18 is very large because of the small angle which it makes with the forward strut.

The mean resultant air load on the wings was found to be 36.6 pounds per square foot. In this, as in all other cases, the variation of unit loading between the wings and the variation

along the spars were neglected, the load per running foot being assumed to be constant. The load was distributed between the front and rear spars in the usual manner, the center of pressure being 33 per cent of the way back on the chord. The lift and drag reactions at the several panel points were then determined, and each lift reaction resolved into the lines of the drag struts and the interplane struts. The perfectly general method of carrying through the work would be to resolve every force into those two lines and a line parallel to the wing spars, and also to determine the direction cosines of every member of the truss with respect to a nonrectangular system of axes parallel, respectively, to the wing spars, to the drag struts, and to the interplane struts,[5] and then to write the equations of equilibrium at every point. Having done this, the solution becomes practically automatic. It is possible, however, to very much shorten the work by a judicious use of the method of sections, especially if the stresses in the wooden members need not be determined. The first part of the problem is to solve for the stresses in all members except the redundant ones, ignoring those entirely; and this is identical with the ordinary stress analysis.

The analysis with wires 6, 7, 20, and 17′ ignored being completed, each of these, in turn, is assumed to carry a tension of 1 pound, and the stresses which every other member of the truss would bear, due to this tension, were there no other loads acting, are computed and tabulated. The total stress in any member can then be expressed by the formula:

$$T_x = f_i + T_6 \times f_6 + T_7 \times f_7 + T_{20} \times f_{20} + T_{17}' \times f_{17}'$$

where T_x is the total stress in the member in question, f_i the stress due to air loads with redundancies omitted from consideration, f_6, f_7, f_{20}, and f_{17}', the stresses due to tensions of 1 pound in 6, 7, 20, and 17′, respectively, and T_6, T_7, T_{20}, and T_{17}', the stresses which actually exist in those redundant members when the structure is loaded.

The work done in elongating the member x is

$$W_x = \frac{T_x^2 \times l}{2AE}$$

where l is the length of the member, A its cross-section area, and E the modulus of elasticity of the material composing it. Writing T in this expression in terms of T_6, T_7, T_{20}, and T_{17}', and doing the same for the expressions for W_y, W_z, and so on, for every member of the structure, the total work of deformation for any set of values of the stresses in the redundancies can be obtained by summation. In order that the work may be a minimum, as required by Castigliano's theorem, its partial derivatives with respect to each of the independent variables (in this case the tensions in the redundant wires) must all be equal to zero. Differentiating the expression for total work with respect to T_6, T_7, and so on for each of the redundancies in turn, four simultaneous equations in four unknowns are obtained, and these can at once be solved. The stresses on all the members taken into account in the usual type of analysis and ordinarily considered as nonredundant can then be determined by substituting in equations of the form given for T_x the values just found for the stresses in the redundancies by solution of the simultaneous equations for the work derivatives.

The carrying through of this process shows the tensions in the redundancies to be 87 pounds in No. 6, 143 pounds in No. 7, 707 pounds in No. 20, and 1 pound in No. 17′. The important figures in connection with each member of the truss are tabulated below. Of special interest are the listings of factors of safety as found by the ordinary statical analysis with all stagger wires and external drag wires disregarded and as found by the complete analysis with these members fully taken into account. It should be borne in mind that these are true factors of safety or "material factors," based on the worst possible loading, and are less than one-fifth as large as the hypothetical "factors of safety" which are usually specified and which are based on the loading in normal rectilinear horizontal flight in smooth air.

The presence of the redundant members reduces the stress in 11 wires and increases it in only 3 (not including the redundancies themselves). The beneficial effect on the worst-stressed members is, however, slight.

[5] This system of axes would be rectangular if there were no stagger.

The stress in the rear inner landing wire is negligible and has been omitted from consideration in computing the factors of safety. Furthermore, the effect of 17′ is actually even a little less than would appear, as the rear portion of the fuselage is subjected to a downward dynamic load, and the point of attachment of the lower end of 19 is therefore deflected downward relative to the points of attachment of the lower wing spars, so that 19 carries a larger share of the upward reaction at the center section of the upper wing than it would if its lower end remained exactly fixed. The effect of the landing wires will therefore be disregarded in all subsequent cases. The possibility of their having an effect is only mentioned as a warning that they should sometimes be taken into account, as the share of the center section load carried by the landing wires increases rapidly as the obliquity of the center section wires is increased.

CASE I.

No.	Stress without redund.	Stress due to 1 pound in No. 6.	Stress due to 1 pound in No. 7.	Stress due to 1 pound in No. 20.	Stress with all redund.	Ultimate strength of member.	F. S. without redund.	F. S. with all redund.
1	162				162	2,600	16.0	16.0
2	518	.683			577	2,600	5.02	4.50
3	433	.683			492	2,600	6.00	5.29
4	905	.654	.654	−1.008	343	4,000	4.61	11.6
5	827	.654	.654	−1.008	264	4,000	4.84	15.1
6		1.000			87	2,000		23.0
7			1.000		143	2,000		14.0
8	257	− .683			198	2,600	10.1	13.1
9	180	− .683			120	2,600	14.5	21.6
10	686	− .654	− .654		536	4,000	5.82	7.45
11	608	− .654	− .654		458	4,000	6.59	8.74
12	508				508	4,000	7.87	7.87
13	367				367	4,000	10.9	10.9
14	2,456	−1.318			2,341	8,400	3.42	3.58
15	1,772	1.318			1,886	8,400	4.74	4.45
16	5,143	−1.272	−1.272		4,851	8,400	1.64	1.73
17	3,711	1.272	1.272	− .523	3,633	8,400	2.26	2.31
18	2,606	1.282	1.282	−1.975	1,505	4,200	1.61	2.79
19	342				342	2,000	5.85	5.85
20				1.000	707	4,200		5.94
22	−2,152	1.052			−2,062			
23	−2,174	−1.583			−2,310			
24	−1,750	1.583			−1,613			
25	−2,510	−2.114			−2,692			
26	−5,425	3.107	.9922		−5,019			
27	−6,087	−3.600	−1.485	.4189	−6,365			
28	−4,742	3.600	1.485	− .7599	−4,739			
29	−6,710	−4.093	−1.979	1.179	−6,586			
30	−2,327	− .4300			− 269			
31	− 295	− .4300			− 332			
32	− 460	− .4300	− .4300		− 557			
33	− 565	− .4300	− .4300	.6625	− 214			
34	− 320				− 320			
35	31	.5301			76			
36	520	− .5301			474			
37	− 109	1.060			− 18			
38	2,619	−2.112			2,437			
39	788	2.606	.4932		1,071			
40	3,136	−2.606	− .4932		2,843			
41	329	3.099	.9864		734			
42	− 162				− 162			
43	− 134	.4300			− 96			
44	− 309	.4300			− 272			
45	− 421	.4300	.4300		− 324			
46	− 878				− 878			
47	− 633	− .7950			− 702			
48	−2,366	.7950			−2,298			
49	−1,707	− .7950	− .7950		−1,887			
50	−1,790	−1.020	−1.020	1.585	− 948			

CASE Ia. (*Effect of wooden members.*)

The loading taken in this case was the same as in the last, but full allowance was made for the effect of the wooden members, in so far as their end loads were concerned, the stresses in these members and the work done in elongating or shortening being computed exactly as for the wires, and the equations of total work being enlarged to include the work which goes into storing strain energy in the spars and struts. The strain energy of flexure has not been taken into consideration, as its variation due to the redundancies is slight, and the accurate computation of flexural work would be an undertaking of great difficulty, requiring a series of successive approximations to allow for the departure of intermediate panel points from the straight line connecting the outermost and innermost supports of the wing truss. It is only because of such departures that the work of flexure is changed by the redundant members, and these

redundancies therefore have no effect on the energy of flexure in airplanes which have no intermediate panel points, the wing truss on each side consisting of a single bay and an overhang.

The introduction of the wooden members into the work equations gives a larger stress in two of the redundancies than was found in Case I, while the stress in the third (No. 7) remains practically unchanged. The tension in No. 6 was increased to a rather surprising extent. In only one wire (No. 18) does the allowance for the struts and spars change the computed tension by as much as 5 per cent of its ultimate strength, and the effect in that one wire, as well as in most of the others, is to reduce the computed stress.

The effect of redundancies on the stresses in the wooden members themselves is small in most instances, but is not by any means small enough to be negligible. The loads in the worst-stressed portions of the wing spars are reduced by from 15 per cent to 25 per cent by the redundant wires, chiefly by the effect of No. 20. The stress in the intermediate compression rib in the inner bay of the upper wing is cut down about 85 per cent by the external drag wires. The interplane struts are but little affected, with the exception of the front center section strut, the stress in which is 64 per cent smaller than it would be if the redundant members were removed.

A tabulation, similar to that for Case I, of the stresses with and without allowance for the redundant members is given below. The differences between the stresses found in Case I and Case Ia, or the errors due to failing to include the wooden members in the work equations have been included in the tabulation.

It appears from the comparison of the results in this case and in Case I that the assumption originally made was a reasonably accurate one, and that it is safe to omit the wooden members from consideration for any except the most refined work.

CASE Ia.

[Stresses without redundancies, and effects of unit stresses in redundancies, are the same as in Case I.]

No.	Stress with all redund.	Difference between Ia and I (absolute magnitude).	No.	Stress with all redund.	Difference between Ia and I (absolute magnitude).	No.	Stress with all redund.	Difference between Ia and I (absolute magnitude).
1	162	0	18	1,186	−319	35	113	+ 37
2	624	+ 47	19	342	0	36	437	+ 37
3	539	+ 47	20	911	+204	37	56	[1] + 38
4	180	−163	22	−1,989	− 73	38	2,291	−146
5	102	−162	23	−2,420	+110	39	1,262	+191
6	155	+ 68	24	−1,504	−109	40	2,662	−181
7	140	− 3	25	−2,839	+147	41	949	+215
8	151	− 47	26	−4,803	−216	42	− 162	0
9	74	− 46	27	−6,473	+108	43	− 67	− 29
10	493	− 43	28	−4,667	− 72	44	− 242	− 30
11	414	− 44	29	−6,550	− 36	45	− 296	− 28
12	508	0	30	− 299	+ 30	46	− 878	0
13	367	0	31	− 362	+ 30	47	− 757	+ 55
14	2,251	− 90	32	− 587	+ 30	48	−2,243	− 55
15	1,977	+ 91	33	− 88	+126	49	−1,942	+ 55
16	4,767	− 84	34	− 320	0	50	− 648	−300
17	3,621	− 12						

[1] The stress is changed in sign in this case.

CASE II.

The loading in this case was that encountered in a vertical dive at 120 miles per hour. This is considerably below the limiting speed of the JN, but is as fast as it is likely to be dived. It was assumed that the upload on the rear truss was equal to the down load on the front truss, and the resultant force on the wings was therefore parallel to the chords. Since the angle of attack was negative, there was some lift on the wings under these conditions, but not enough to balance the down load on the tail. In the particular machine used as an illustration the angle of zero normal force is −5°, the zero lift angle for the Eiffel 36 section being unusually small. The true angle of attack in a vertical dive would probably be nearer −4° than −5°. The components of load acting perpendicular to the wing chord were 45 pounds per foot, giving a total force of about 1.6 times the weight of the airplane in each lift truss (including both the right

and left sides of each truss). This is unusually large, the Eiffel 36 wing having an exceptionally large diving moment at the angle of zero lift. The loading in diving the JN to 120 miles per hour is about equal to that which would be found at the terminal velocity with most airplanes using the R. A. F. 15 or other similar section. The load parallel to the wing chord (front and rear trusses together) was 7.22 pounds per foot, so that the total distributed load on the drag trusses, including the parasite resistance of the interplane bracing, but not including the components in the planes of the wings, due to stagger, of the lift truss reactions, was about 29 per cent of the weight of the airplane. In a dive to the terminal velocity this force may rise to as much as 50 per cent of the weight of the airplane for a machine with fine lines and low parasite resistance.

The front king-post bracing above the upper wing is stressed by the down-load on the front truss, and the stresses in the two lift trusses therefore are not quite symmetrical with respect to each other. If the two systems of trussing were parallel throughout, the stagger would have no effect on the net reactions in the plane of the wing, as the effects of the inclination of the lift bracing would be equal and opposite at the front and rear panel points and would exactly cancel out; but this is not actually the case, since the king-post overhang bracing lies in a plane perpendicular to the wing chord instead of being parallel to the lift truss proper.

There are three redundancies in this case, Nos. 6′, 7′, and 21. The stagger wires acting are those which run upward and to the rear, as might be expected, since the rear truss tends to move up and the forward one down, and the stagger wires acting are those which are thrown into tension in resisting this relative displacement of the lift trusses. The work equations were 424 pounds in 6′, 427 pounds in 7′, and 485 pounds in 21. It might perhaps have been anticipated that No. 20 would be in tension, as the rear truss considered alone tends to move upward and to the rear and both of these components of motion would elongate No. 20, but analysis shows that No. 20 would carry a considerable compressive load if it were capable of sustaining such a load. The physical explanation of this is dual. In the first place, the pull in stagger wires Nos. 6′ and 7′ tend to draw the upper wing forward. Secondly, and more important, the load in the rear truss is carried by the flying wires, while that in the front truss falls on the landing wires. These, being single in each bay, elongate more under a given load than do the double flying wires, and, if the two trusses were not connected together in any way, the front one would deflect downward more than the rear one would yield upward. Since the two are connected by the stagger wires and must move substantially together, the effect of the dissymmetry of the lift and antilift bracing is to cause the wing cell to deflect downward as a whole. The upper rear spar therefore moves, not upward and backward as it would if there were no redundancies, but forward and downward. Incidentally, this serves as an excellent illustration of the intricacies of a redundant structure and of the manner in which the stress in any member depends on the form and strength of every other member. For example, if the antilift wires as well as the lift wires, were double there is but little doubt that the upper drag wire (No. 20), as well as the lower one, would carry a considerable tensile load during a dive instead of going slack.

The pull of the stagger wires, drawing the upper wing forward, also has the effect, not very generally foreseen or allowed for, of throwing a load on the antidrag wires in the upper wing. A load on these wires is expected at large angles of attack, particularly in airplanes with little or no stagger, but its appearance in a vertical dive seems rather curious until a thorough analysis is made.

The nature of the load distribution in the center section is quite different from that at a large angle of attack, although three of the four members involved are active in each case. In a dive, the front wire (No. 18) takes no load. Both struts are in compression, and the forward tendency of the upper wing, due to the pull of the stagger wires, is resisted by a tension in No. 19.

A tabulation of stresses, similar to that given for Case I, appears below. There has been no recomputation of redundancies with the work done in the spars and struts taken into account in this case, but the final stresses in the wooden members have been computed with allowance for the redundancies found by writing the work equations for the wires alone.

The factors of safety in the wires are high and fairly uniform. The stresses in the wooden members, with a few exceptions (chiefly the internal drag struts), are reduced by the introduction of the redundant wires. This is particularly true of the worst-stressed portions of the spars, the maximum direct compressive loads being reduced by about 72 per cent. It is a curious fact that every bay of every spar is in compression in a dive, the effect of the stagger wires and of the king-post bracing being sufficient to overcome the tension which might normally be expected to appear in the upper front and lower rear spars.

The stagger wires are of enormous benefit as regards the lift trusses. In the lift and anti-lift wires, as in the spars, the stresses are from 55 per cent to 70 per cent lower than they would be if the stagger wires were removed.

CASE II.

No.	Stress without redund.	Stress with all redund.	F. S. without redund.	F. S. with redund.	No.	Stress without redund.	Stress with all redund.	F. S. without redund.	F. S. with redund.
1	+ 133	+133	19.6	19.6	25	−1,222	−300		
2	+ 72		36.3		26	+1,225	−253		
2′		+289		4.50	27	−3,094	−783		
3	+ 115		22.6		28	+1,375	−610		
3′		+245		5.30	29	−3,255	−416		
4	+ 200		20.0		30	− 45	−225		
4′		+488		4.10	31	− 73	−180		
5	+ 240		16.7		32	− 131	−379		
5′		+447		4.47	33	− 158	−320		
6′		+424		4.72	34	− 976	−421		
7′		+427		4.68	35	+ 90	−188		
8	+ 55	+415	47.6	6.27	36	− 933	−101		
9	+ 94	+455	28.6	5.72	37	+ 17	−539		
10	+ 142	+457	28.2	8.75	38	−2,581	−825		
11	+ 182	+497	22.0	8.05	39	+ 751	−597		
12′	+ 513	+513	3.90	3.90	40	−2,474	−480		
12″	+ 518	+518	3.86	3.86	41	+ 614	−972		
13	+ 211	+211	19.9	19.9	42	− 34	−259		
14′	+1,223	+528	3.44	7.96	43	− 59	−284		
15	+1,053	+359	7.98	23.1	44	− 93	−300		
16′	+2,206	+770	1.90	5.46	45	− 119	−326		
17	+2,206	+860	3.81	9.77	46	− 530	−530		
18	+ 529		7.94		47	− 377	−377		
19	+ 202	+502	9.90	3.98	48	− 999	−580		
21		+485		8.67	49	−1,015	−596		
22	− 410	−632			50	− 625	−615		
23	−1,133	−523			50A		−210		
24	− 354	−820							

CASE III.

The loading in this case was one devised by the authors and recommended for use as a standard in sand-load tests. It was based on an attempt to distribute the load over the wings in such a manner that both lift trusses and both drag trusses would simultaneously reach the worst load which they ever encounter in flight. The total load on the wings was taken as 5.3W. The center of gravity of the load was placed at 37 per cent of the chord from the leading edge, and the chord was assumed to be inclined at 6.5° to the horizontal, the trailing edge being lower than the leading edge (the wing truss, of course, being inverted for sand-load test). The load per running foot was 84 pounds in the front truss and 78 pounds in the rear.

The solution was exactly similar to those for Cases I and II and calls for no special comment. Since the load was nearly equally distributed between the front and rear trusses the stresses in the stagger wires were extremely small, different diagonals being stressed at the two panels and the stress in the short diagonal at the outer panel point being less than 1 pound. The larger pull in the long stagger wire at the inner panel point is due to the forward reaction of the upper external drag wire on the upper wing at that point. Both external drag wires carry some load, the upper one taking more than the lower, as the upper wing deflects more freely in the direction of the drag truss than does the lower and as the upper drag wire also assists in carrying the lift.

The nature of the stress distribution in the redundancies causes a very peculiar reversal of direction of stress in the internal drag bracing of the upper wing. The direction of the load-carrying diagonal reverses twice, so that the load-carrying members are arranged as in a Warren truss, but with all the members in tension. The compression ribs at the points where these

reversals occur carry no load at all, and a sand load in accordance with these specifications would therefore be unduly easy on the upper drag truss in the inner bay. The stress in the upper front and lower rear spars, also, are considerably less in Case III than in Case I, particularly in the inner bays. The drag wires in the inner bay of the lower wing and some of the compression ribs in both upper and lower wings, on the other hand, are stressed more severely in the sand load than they ever would be in flight. The comparison of the results of the various analyses serves to emphasize the impossibility of devising any single sand load which will truly simulate all of the "worst conditions" that may be encountered in the air.

CASE III.

No.	Stress without redund.	Stress with all redund.	F. S. with all redund.	No.	Stress without redund.	Stress with all redund.	F. S. with all redund.
1	392	392	6.63	26	−2,182	−1,923	....
2	835	835	3.11	27	−6,353	−5,870	....
3	947	946	2.75	28	−2,182	−2,164	....
4′	−1,495	8	2.47	29	−8,684	−7,169	....
5	1,598	95	42.2	30	− 388	− 388	....
6′	....	1	33.30	31	− 561	− 561	....
7	....	201	9.98	32	+ 186	− 887	....
8	396	396	6.56	33	− 35	− 35	....
9	497	497	5.22	34	+ 254	+ 254	....
10	1,090	800	5.00	35	− 70	− 70	....
11	1,192	902	4.43	36	+ 562	+ 562	....
12	404	404	9.89	37	− 456	− 457	....
13	376	376	10.6	38	+2,507	+2,338	....
14	1,954	1,955	4.30	39	+ 175	+ 390	....
15	1,819	1,818	4.62	40	+3,327	+2,941	....
16	4,093	3,879	2.17	41	− 724	− 290	....
17	3,810	3,217	2.61	42	− 128	− 128	....
18	4,302	1,358	3.09	43	− 282	− 282	....
19	351	351	5.70	44	− 521	− 394	....
20	....	1,620	2.59	45	− 751	− 555	....
21	....	207	20.3	46	− 699	− 699	....
22	−1,507	−1,508	....	47	− 650	− 650	....
23	−2,644	−2,643	....	48	−1,883	−1,860	....
24	− 857	− 858	....	49	−1,753	−1,912	....
25	−3,380	−3,379	....	50	−3,251	− 887	....

EFFECT OF INITIAL TENSIONS.

The analyses of the first two cases have been based on the assumption that all of the wires are just taut but with no initial tension. Actually, even if it were possible to secure such an adjustment it would not be desirable to do so, as some initial tension is necessary in order to keep the structure from vibrating badly and to hold it in proper alignment. It is therefore necessary to investigate the effect of initial stress on the distribution of load.

It is not correct to apply the method of least work in a straightforward manner, taking the derivatives of the work done by the external loads along, or of the change in total strain energy due to the imposition of the external loads, as might at first be assumed to be the case. The partial derivative of the total strain energy with respect to the stress in any member is equal to the deflection, parallel to the line of that member, of the point at which the force representing the stress is considered to be applied, this deflection being measured from the point at which there would be no stress in the member in question. If the frame of the structure is lined up with initial tensions in some or all of the members, the deflections which are desired in order to establish the conditions of geometrical equilibrium of the truss are those measured from the strained lengths of the members before the external loads are applied, and it is therefore necessary to make a deduction for the initial deflections due to straining of the redundant members against each other. The equations based on the work derivatives, and defining the relations between the final stresses in the redundant members, must then be written:

$$\frac{dW}{dT_x}-\frac{dw}{dt_x}=0$$

where W is the work done by external loads, w the work of deformation when the initial stresses alone are acting, T_x the final tension in any redundant member and t_x the initial tension. It is not necessary, however, to re-write all the equations, as it is sufficient to carry through the

analysis and compute the stresses without regard to the initial tensions, and then to add to the stress in each member that due to initial stress in the redundancies. It is evident that this is the case, as the equations for W and w in each member are homologous, except that the terms involving only one unknown stress do not appear in the latter, since those terms are due to the external loads. The derivatives $\frac{dW}{dT_x}$ and $\frac{dw}{dt_x}$ are then identical, except that the second involved t where the first has T, the subscripts remaining the same, and that the first has a pure numerical term which is lacking in the other. The terms combine, when the second expression is subtracted from the first, in such a way that neither T nor t appears singly, but always in the combination $(T-t)$, and it would therefore have been sufficient to write in the first place

$$\frac{dW}{d\,(T_x-t_x)}=0$$

where W is given the fictitious value $\frac{(T-t)^2\,1}{2AE}$ instead of $\frac{(T^2-t^2)\,1}{2AE}$, which is the true change in strain energy caused by the application of the external loads. The solution of the simultaneous equations then gives T_x-t_x for the redundant members, and the initial stresses must be added in to secure the total final load.

The effect of initial tension can best be illustrated by giving a couple of simple examples. As a first instance the pin-jointed structure shown in figure 3, and consisting of bars cross braced with wires, may be selected. It is assumed that the bars are so large in proportion to the wires that their strain may be neglected, and that the two diagonal wires are of equal size. If an initial tension F be placed in one wire there must be an equal and opposite initial tension resisting it in the other diagonal member in order that the structure may be in equilibrium. If an external load 0.707 P be applied as shown in the figure wire No. 1 will carry a tension of P pounds while No. 2 goes slack if there is no initial tension. If there is initial tension No. 2 will shorten by exactly the same amount that No. 1 lengthens, and the resultant tension in No. 1 will be $F+\frac{P}{2}$, while that in No. 2 is $F-\frac{P}{2}$. The tensions will vary in this manner as P is increased until $P-2F$, at which time the tensions are P and O. Thereafter the stresses are the same as if there had been no initial tension. If this very simple problem had been treated by least work with initial tension the stresses determined would have been $+\frac{P}{2}$ for 1 and $-\frac{P}{2}$ for 2. Adding these stresses algebraically to the initial tensions in the two members the same result is obtained as was just given as a result of elementary geometrical reasoning.

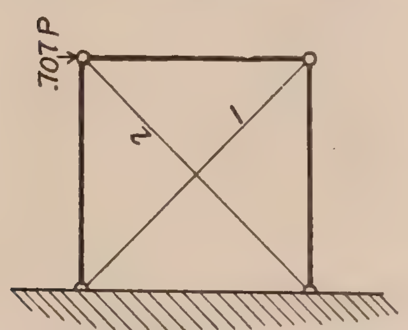


FIG. 5.

If, in this problem, No. 2 had only half the cross-section area of No. 1 the initial tensions in the two would, as before, be equal. An applied load superimposed on the original stresses would, however, produce twice as great an effect in 1 as in 2, since the increase in tensile strain of 1 as the structure deforms must be equal to the decrease of strain in 2. The unit stresses in the two are then equal if they are of the same material, and the total stresses are proportional to the cross-sectional areas. It follows from this that the total loads in the two wires are given, so long as they both remain in tension, by the formulae $F+\frac{2P}{3}$ and $F-\frac{P}{3}$ and that the lighter wire will not become slack until $P=3F$.

To afford some indication of the initial tensions existing in airplanes rigged in the field under average conditions and without using a tensiometer, tensiometer measurements of the stresses in all the exposed wires were made for 6 JN4H airplanes, four of them rigged by four different Army crews and the remaining two by a civilian crew. The averages are tabulated below, together with the mean deviations showing how widely the tensions in corresponding wires varied in the several machines. In the case of the flying wires, the mean deviations given

are the mean deviations of the total stress in the two parallel wires from the mean value of that total, and the figures in parentheses, immediately under those mean deviations, are the means of the differences between the tensions in two parallel wires on the same airplane. The tensimeter readings taken in this way do not directly represent the true initial tensions, as the weight of the cellule is an external load which was being carried by the landing wires at the time when these measurements were made. The tensions read in the landing wires were therefore a little higher than the true initial tensions, while the values for the flying wires were correspondingly too low. This effect, amounting to about 60 pounds in some wires, has been corrected for in compiling the table of means. The magnitudes of the mean deviations in initial tensions strongly indicate the advisability of using a tensiometer and straining all wires in accordance with a schedule specified by the builder of the airplane. This method has been tried in rigging one or two machines at Langley Field, the tensiometer being used by mechanics with no previous experience with such an instrument, and a great improvement in the rigging was manifested. Where it had been common for one or more wires to vibrate badly at all engine speeds when the initial tension was adjusted by feel in the usual manner, there was no vibration except at one critical speed on the machine rigged by tensiometer.

It has been assumed that the probable maximum of initial tension in any particular wire given reasonably competent and careful rigging, is equal to the mean of the tensions for the six machines examined plus twice the mean deviation. This is not by any means an absolute maximum, and it was exceeded in some wires on several of the airplanes examined, but it represents a figure which need not and should not ever be exceeded. These probable maxima have also been included in the tabulation above. In the case of the stagger wires, where both wires remain in tension and it is only the amount of unbalanced tension or the difference between the two, which must be taken into account, the assumption in the analysis has been that the wire stressed by external loads (the long one) has an initial tension equal to the average for the six airplanes plus the mean deviation and that the short wire carries a stress less than the average by an amount equal to the mean deviation for that member. The difference between the two is therefore twice the average of their mean deviation.

TABLE OF MEAN INITIAL TENSION ON SIX JN4HS.

Wire No.	Average tension.	Mean deviation.	Probable maximum.	Wire No.	Average tension.	Mean deviation.	Probable maximum.
6	665	129	923	15	691	107	905
6′	498	90	678	(Double)		(83)	
7	737	170	1,077	15′	730	215	1,160
7′	497	105	707	16	645	118	881
12	110	47	204	(Double)		(92)	
(Double)		(33)		16′	693	167	1,027
12′	205	64	333	17	684	192	1,068
13	85	38	161	(Double)		(54)	
(Double)		(27)		17′	728	122	972
13′	181	62	305	18	466	105	676
14	721	138	997	19	630	87	804
(Double)		(64)		20	225	31	287
14′	838	138	1,114	21	269	86	441

The differences between the initial tensions in any given pair of opposed wires can be computed, if the initial tensions in the redundancies are known, on the usual assumption of frictionless pin joints. Any discrepancy between the difference of stress thus computed and that found by actual measurement is then due to the partial rigidity of the joints and the continuity of the spars. If, when the structure is in perfect alignment, there is a difference between the computed and measured stresses in the nonredundant members, it shows that the wings are warped and that they have had to be initially stressed to draw them into alignment. In the average of the six machines measured this discrepancy was largest in the inner bay of the rear truss, where it amounted to a deficiency of about 200 pounds in the tension in the flying wires. This is largely due to the relative bowing of the left rear spars in order to give "droop" to that wing and balance the engine torque.

The effect of the maximum probable initial tension has been computed for all three of the loadings thus far treated, and the results are tabulated below. In general, the effect on the worst-stressed members is injurious, and the initial tensions should therefore be kept as small as possible without permitting excessive vibration. In tabulating the stresses due to initial tension it has been assumed in every case that the excess tension is in that stagger wire where it will increase the stress, as both stagger wires of an opposed pair are in tension at all times with the usual initial tension. It will be noted that the factors of safety in the stagger wires are low, as their initial tensions are a large proportion of their ultimate strengths. The change of tension in the stagger wires under load is therefore small in comparison with the initial tension, the stress in one wire increasing while that in the other decreases so that the change in each wire is equal to approximately half the tension computed by the least work analysis. This is in accordance with the results of sand load tests, where tensiometer measurements after the application of each load have shown that the stresses in the stagger wires vary only a little from their initial values. In addition to always taking the worst condition as regards the initial distribution of load between the stagger wires, the stresses in the external drag wires have been taken as the probable minimum, instead of the probable maximum, wherever that would be the worst condition as regards the resultant stress in any particular member.

In a few cases the influence of the initial stress is great enough to control the direction of the diagonal which carries load in the internal drag bracing, the load shifting from the drag to the antidrag wires, or vice versa, if the excess unbalanced tension is transferred from one stagger wire to the opposed member. In some cases this leads to difficulty where the worst loads in the spars and in the internal drag wires occur under different conditions of initial adjustment and where the worst load in the spars corresponds with a reversal of stress and a transfer to the opposite diagonal from that which normally carries the tension in the internal truss. When this occurs it would be necessary, in order to secure strictly accurate results, to carry the whole analysis through from the start with the antidrag wires included and the drag wires omitted, but a close approximation can be made without the necessity of repeating the work in this manner. This approximation is based on the assumption that a compression in one diagonal of a rectangular frame can be replaced by a tension in the opposite diagonal, an assumption which would be true if the frame were exactly symmetrical and if the drag and antidrag wires were of the same size. If any particular combination of initial tensions gives a negative result for the total force in a drag wire this wire is therefore replaced by the opposed member, and it is assumed that the resultant stress determined is unchanged in magnitude but reversed in sign. A correction has to be applied to the stresses in the spars in the panel where this reversal occurs, as the drag and antidrag wires do not affect the same portions of the spars. In the second panel from the tip of the upper wing, for example, the stress in 22 (see figure 1) is affected by a force in 2′ but not by one in 2, whereas exactly the opposite is the case with 23. It would therefore be necessary, in arbitrarily passing from 2 to 2′ as the load-carrying member, to subtract (algebraically) from the direct load on each spar panel an amount equal to the component parallel to the transverse axis of the stress in the wire. The correction is subtractive in each case, as there is taken away from 23 a tension due to the fictitious compression in 2, while there is added to 22 a compression arising from the real tension in 2′, this tension being equal in magnitude, as already noted, to the theoretical compression found in 2. In the tabulation, wherever an approximation of this sort has been made the stress for the member affected is placed in parentheses.

In the members (interplane struts and compression ribs) directly interposed between two points of attachment of stagger wires, the fact that both wires remain in tension under all conditions has been allowed for. The final stress in any stagger wire is approximately equal to the initial stress plus or minus half the computed stagger wire tension (the stress being increased in the diagonal which was originally assumed to be stressed, decreased in the other). This, again, is only an approximation, but approximations are essential if the work is not to be complicated beyond all endurance by the introduction and simultaneous treatment of about 20 redundancies.

INITIAL TENSIONS, CASE I.

No.	Stress without initial tension.	Stress due to initial tension in 6.	Stress due to initial tension in 7.	Stress due to initial tension in 20.	Total resultant stress.	No.	Stress without initial tension.	Stress due to initial tension in 6.	Stress due to initial tension in 7.	Stress due to initial tension in 20.	Total resultant stress.
1	163	0	0	0	163	27	−5,307	−1,040	−627	312	−6,662
2	577	197	0	0	774	28	−4,739	− 691	−264	−2,181	(−6,005)
3	482	197	0	0	679	29	−5,528	−1,183	−833	436	−7,108
4	343	189	276	−164	644	30	− 269	− 395	0	0	− 647
5	264	189	276	−164	565	31	− 332	− 124	0	0	− 456
6	86	923	0	0	[1] 966	32	− 557	− 124	−463	0	[1]−1,114
7	140	0	1,077	0	[1] 1,147	33	− 214	− 124	−182	108	− 411
8	198	131	0	0	329	34	− 320	0	0	0	− 320
9	120	131	0	0	251	35	76	153	0	0	229
10	536	126	116	0	778	36	474	102	0	0	576
11	458	126	116	0	700	37	− 18	....	0	0	[3](228) (222)
12	508	0	0	0	508						
13	367	0	0	0	367	38	2,437	406	0	0	2,843
14	2,341	253	0	0	2,594	39	1,071	753	208	0	2,032
15	1,886	381	0	0	2,267	40	2,843	500	88	0	3,431
16	4,851	244	226	0	5,321	41	734	896	416	0	(2,042)
17	3,633	368	537	− 85	4,453	42	− 162	−360	0	0	[1]− 503
18	1,505	371	541	−322	2,095	43	− 96	− 83	0	0	− 179
19	342	0	0	0	[2] 804	44	− 272	− 83	−375	0	− 701
20	677	0	0	287	964	45	− 324	− 83	− 76	0	− 483
22	−2,062	−202	0	0	−2,264	46	− 878	−671	0	0	[1]1,514
23	−2,310	−458	0	0	−2,768	47	− 702	−734	0	0	[1]1,401
24	−1,613	−304	0	0	−1,917	48	−2,298	−153	−699	0	[1]−3,045
25	−2,692	−611	0	0	−3,303	49	−1,887	−230	−856	0	[1]−2,917
26	−5,019	−597	−177	0	(−5,824)	50	− 948	−295	−430	258	−1,415

INITIAL TENSIONS, CASE II.

No.	Stress without initial tension.	Stress due to initial tension in 6′.	Stress due to initial tension in 7′.	Stress due to initial tension in 21.	Total resultant stress.	No.	Stress without initial tension.	Stress due to initial tension in 6′.	Stress due to initial tension in 7′.	Stress due to initial tension in 21.	Total resultant stress.
1	133	0	0	0	133	28	−610	− 599	−174	0	−1,383
2′	289	132	0	0	420	29	−416	−1,036	−629	0	(−2.092)
3′	245	132	0	0	377	30	−225	− 360	0	0	[1]− 473
4′	487	126	116	0	729	31	−180	− 83	0	0	− 263
5′	447	126	116	0	689	32	−379	− 83	−376	0	[1]− 725
6′	424	498	0	0	[1] 710	33	−320	− 83	− 76	0	− 479
7′	427	497	0	0	[1] 711	34	−421	− 304	0	0	− 725
8	415	132	0	0	547	35	−188	− 102	0	0	− 290
9	455	132	0	0	586	36	−101	− 456	0	0	− 557
10	457	126	116	− 74	624	37	−539	− 205	0	0	− 743
11	497	126	116	− 74	664	38	−825	− 894	421	−328	(−2,767)
12′	513	0	0	0	513	39	−597	− 504	−504	56	−1,132
13	518	0	0	0	518	40	−480	−1,036	−629	−623	(−2,996)
14′	528	380	0	0	908	41	−972	− 599	−599	112	−1,633
15	359	380	0	0	739	42	−259	− 360	0	0	− [1] 507
16′	770	367	540	− 18	1,657	43	−284	− 83	0	0	− 367
17	860	367	540	0	1,766	44	−300	− 83	−376	49	− [1] 646
19	502	109	100	− 3	708	45	−326	− 83	− 76	49	− 436
21	485	0	0	441	926	46	−530	− 734	0	0	[1]− 1,055
22	−632	−102	0	0	− 734	47	−377	− 734	0	0	[1]− 902
23	−523	−304	0	0	− 836	48	−580	− 229	−856	0	[1]− 1,455
24	−820	−205	0	0	−1,025	49	−596	− 229	−856	0	[1]− 1,471
25	−300	−456	0	0	− 756	50	−615	− 237	−348	12	−1,188
26	−253	−504	− 87	0	− 844	50A	−210	− 65	− 60	− 22	− 22
27	−783	−894	−421	0	−2,098						

[1] The maximum probable stress here is not equal to the sum of the figures in the first four columns, as the stress in any stagger wire, so long as both diagonals remain in tension, varies only half as rapidly as does the computed stagger wire tension under air loads, and this variation is negative in one stagger wire of each pair. The stress in interplane struts and compression ribs caused by the pull of the stagger wires between which they are interposed, is therefore less, in most cases, when the air load is acting than when the machine is resting on the ground. In other words, the air load decreases the initial tress due to the opposition of the stagger wires, but adds another component of stress from an entirely different source.

[2] The maximum probable stress here was arbitrarily taken as equal to the maximum probable initial tension found by measurement. Computation of the initial tension and combination of the stresses is impossible in this case, as any initial tension in 19 would have to be balanced by a compression in 50A, and 50A was left out of consideration in making the least work analysis.

[3] The stress in 37 may be either tensile or compressive, depending on which stagger wire carries an excess of initial tension.

CASE III.

No.	Stress without initial tension.	Stress with worst initial tension.	No.	Stress without initial tension.	Stress with worst initial tension.	No.	Stress without initial tension	Stress with worst initial tension.
1	392	392	18	1,358	1,886	35	− 70	− 172
2	835	1,033	19	351	351	36	562	− 664
3	946	1,144	20	1,620	1,907	37	− 457	− 660
4′	8	538	21	207	648	38	2,338	2,664
5	95	396	22	−1,508	−1,711	39	390	1,609
6′	1	678	23	−2,643	−3,103	40	2,941	3,393
7	201	1,187	24	− 858	−1,163	41	− 290	− 951
8	396	526	25	−3,379	−3,992	42	− 128	− 488
9	497	627	26	−1,923	−2,779	43	− 282	− 365
10	800	968	27	−5,870	−7,320	44	− 394	− 756
11	902	1,018	28	−2,164	−3,732	45	− 555	− 663
12	404	404	29	−7,169	−8,756	46	− 699	−1,370
13	376	376	30	− 388	− 785	47	− 650	−1,384
14	1,955	2,210	31	− 561	− 685	48	−1,860	−2,687
15	1,818	2,204	32	− 887	−1,417	49	−1,912	−3,088
16	3,879	4,432	33	− 35	− 35	50	− 887	−1,217
17	3,217	4,039	34	254	− 254			

The results of the investigations, as recorded in these tables, emphasize the great importance of initial tension, the deleterious effects of which have too seldom been appreciated. In almost every instance the stresses under the worst probable distribution of initial tensions are greater than those which arise from the air load alone, either with or without redundancies. In short, the stagger wires, as they are usually set up, are actually harmful and weaken the structure under most conditions of flight, whereas they should be an important element of strength. The initial tensions in the external drag wires are much more innocuous, although the values selected there should always be as small as are consistent with the rigidity of the structure and with freedom from vibration when in flight. The stagger wires, being disposed in directly opposed pairs, can and should be so adjusted that there will be little or no unbalanced tension to affect the remainder of the truss. Even if this is done, however, the initial tensions should be kept small to ease the strain on the stagger wires themselves and on the interplane struts and drag struts or compression ribs which make up the parallelogram frames at panel points. If the alignment of the air plane is carried out with a tensiometer the element of guesswork is definitely removed, the factors of safety in some important and badly stressed members are increased by from 25 per cent to 50 per cent, and the time required for rigging is increased very little, if at all. In fact, it is probable that a crew which has had a little experience with a tensiometer can work quite as rapidly with as without it, as the amount of trial and error required to bring the machine into true alignment is less than by the ordinary method.

In order that mechanics may have some reliable guide for use in rigging the designers of airplanes should draw up schedules of initial tensions to be used. The primary principle to be followed in drawing up such a schedule is that there should be no unbalanced tension in either of two directly opposed members. In a rectangular frame this means that the initial tension must be equal. (This of course applies to the total tensions where there are two or more members in parallel. Where, for example, two flying wires oppose a single landing wire the initial tension in each flying wire should be just half that in the landing wire.) Where the frame is not rectangular, but has two parallel sides, as in the stagger panels of an airplane with stagger or in the lift truss of a machine with interplane struts sloping outwardly and with the same amount of dihedral in the upper and lower wings, the condition is that the diagonal wires should have equal components perpendicular to the parallel sides. In the case of a stagger panel, this means that the tensions in the two stagger wires should be inversely proportional to the sines of the angles which they make with the wing chords, so that the long diagonal has the larger tension.

In drawing up a tension schedule the periods of vibration of all the wires should be high enough not to synchronize with the natural period of the engine, and should be approximately the same throughout the structure. The fundamental frequency of a stretched wire can be shown[6] to be equal to $\frac{1}{2l}\sqrt{\frac{T}{m}}$, where l is the length of the wire, T the tension, and m the

[6] Textbook on Sound, J. H. Poynting & J. J. Thomson: p. 88.

mass per unit length, which is of course directly proportional to the sectional area and so to the strength of the wire. The tension to give a constant frequency must therefore be proportional to the ultimate strength and to the square of the length, and it is necessary that very long wires be supported at some intermediate point, as the initial tension required to prevent vibration if this were not done would be dangerously high. It has been found by actual experiment that an initial tension of 220 pounds in the upper drag wire of a JN is enough to prevent vibration. Since this wire carries an additional load of about 140 pounds when flying normally with a load factor of 1, the total resultant tension for satisfactory results is 360 pounds, and this may be taken as a basis for the determination of the other tensions. The flying and landing wires are substantially equal in length to the upper drag wire, but they have an intermediate point of support where they cross each other. The area of all these members are the same, and the resultant tension in the flying and landing wires must therefore be at least 90 pounds (the effective length being halved). With a load factor of 2, which is as high a value as is likely to be maintained steadily, the air load reduces the stress in the inner landing wires by about 630 pounds (the total air load on the wires in the inner bay being 1,880 pounds, of which two-thirds is taken by an increase in the stress in the double flying wires, while the remaining third shows as a reduction in the landing wire tension), and the initial tensions therefore should be at least 720 pounds. The initial tension in each flying wire, as already noted, should be half this amount. In the outer bay a tension of 390 pounds in the landing wires is sufficient, as the air load effect there is less. The length of the long stagger wire is approximately two-thirds that of the upper drag wire, and there is a center support where the two stagger wires cross. The area of the stagger wire is about half that of the external drag wire, so that the resultant tension for Nos. 6 and 7 in the conspectus only needs to be one-eighteenth of that for No. 20, or 20 pounds. Under normal conditions of flight (load factor of 2 or less) the tension in the stagger wires is not changed more than 30 pounds by the air load, and the initial tension thus does not need to exceed 50 pounds. Making some extra allowance to secure rigidity, 150 pounds for the long wire and 120 pounds for the short one appears ample, and tests in flight have shown it to be so.

The complete tension schedule for the JN is given below, and will serve as a guide in drawing up such schedule for other machines of similar type.

Member.	Initial tension (including weight of wings).	Member.	Initial tension (including weight of wings).
Inner front flying wires (each)	360	Stagger wires, long	150
Inner rear flying wires (each)	270	Stagger wires, short	120
Outer flying wires (each)	180	Front center section wires	150
Inner landing wires	780	Rear center section wires	230
Outer landing wires	420	Upper drag wire	220
Overhang flying wires (each)	70	Lower drag wire	340
Overhang landing wires (king-post)	400		

The pulls in the flying and landing wires in the inner bay are not exactly balanced because the vertical components of the tensions in the external drag wires are balanced by modification of the flying wire stresses.

PRACTICAL CONCLUSIONS AND SUMMARY.

The conclusions to be drawn from this work will first be tabulated and will then be examined more in detail where they call for such examination.

(*i*) The making of a least work analysis of a new design for at least one case is thoroughly justified. The labor of making such an analysis is not excessive and it gives an idea of the nature and magnitude of the true stresses which can not be obtained in any other way.

(*ii*) The wooden members may be omitted from consideration in the work equations without causing any serious error.

(*iii*) The effect of the stagger wires is unimportant when the load is approximately equally distributed between the front and rear trusses. In diving the effect of the stagger wires is

very important, and greatly reduces the load on the lift trusses. The effect of the stagger wires depends in part on the arrangement of the external drag wires. If there is no external drag wire attached to the upper wing, and if the center section wires have as little forward inclination as they have on the JN, the stagger wires running upward from front to rear will be in tension at all times, transferring drag from the upper to the lower wing, and must be taken into account.

(*iv*) The tension in the external drag wires varies widely with the conditions of loading. Only very rarely are both wires stressed at the same time, and most of the work now done by the two wires could be accomplished equally well by a single one.

(*v*) The initial tensions are almost always excessive, particularly in the stagger wires, and are sometimes so large as to be dangerous, especially as regards the compression ribs at the lift truss panel points. The initial tension is sometimes so high that the total effect of the redundancies becomes harmful, whereas it should be distinctly beneficial to the total strength of the truss.

RECOMMENDATIONS.

I. Only one external drag wire should be used on each side of the plane of symmetry That one can be kept in tension nearly all the time, whereas, as already noted, it is only rarely that the upper and lower drag wires are in tension simultaneously. The structure should of course be designed to fly normally (not to be stunted) without any external drag wires at all. A single drag wire should be attached at the lower front spar, so that it will resist the downward and backward deflection of the truss during a dive. If two external wires are used the second one should be attached either to the upper front or the upper rear spar. The first position is probably the more effective in most instances, as the drag wire then relieves the very heavy load on the front lift truss at large angles. The same result can be obtained without the use of a second drag wire by increasing the strength of the flying wires in the inner bay and attaching them to the fuselage a little forward of the wing spars, as has been done in several recent designs, in order that they may resist the drag on the upper wing. Attachment of the drag wire at the lower rear spar should not be employed.

II. The stagger wire which runs upward from front to rear carries a heavy load at times and may well be made stronger than the other diagonal. If a steel tube, with no opposing member, is used for stagger bracing it should run upward from front to rear. If there is no drag wire attached to the upper wing, such a tube need not be designed to carry a compressive load of more than one-eighth the weight of the airplane, but it should be capable of sustaining a tension equal in magnitude to the total weight of the machine. If picture-frame struts are used, and they are highly recommended, they should be designed to carry from five to eight times as large a compressive load in the direction of the long diagonal (for a machine with positive stagger) as in the direction of the other diagonal.

III. Airplanes should be rigged, whenever possible, by means of a tensiometer and in accordance with a schedule of initial tensions to be provided by the designer. Detailed instructions for drawing up such a schedule have already been given. In particular, the tensions in the stagger wires should be far less than has been the common practice, and opposing members should exactly balance each other. One great advantage of the picture-frame strut is that it eliminates all danger of excessive initial tension.

REPORT No. 93

AERODYNAMIC CHARACTERISTICS OF AEROFOILS

BY NATIONAL ADVISORY
COMMITTEE FOR AERONAUTICS

REPORT No. 93.

AERODYNAMIC CHARACTERISTICS OF AEROFOILS.

By National Advisory Committee for Aeronautics.

INTRODUCTION.

This collection of data on aerofoils has been made from the published reports of a number of the leading aerodynamic laboratories of this country and Europe. The information which was originally expressed according to the different customs of the several laboratories is here presented in a uniform series of charts and tables suitable for the use of designing engineers and for purposes of general reference.

It is a well-known fact that the results obtained in different laboratories, because of their individual methods of testing, are not strictly comparable even if proper scale corrections for size of model and speed of test are supplied. It is, therefore, unwise to compare too closely the coefficients of two wing sections tested in different laboratories. Tests of different wing sections from the same source, however, may be relied on to give true relative values.

The absolute system of coefficients has been used, since it is thought by the National Advisory Committee for Aeronautics that this system is the one most suited for international use, and yet is one for which a desired transformation can be easily made. For this purpose a set of transformation constants is included in this report.

Each aerofoil section is given a reference number, and the test data are presented in the form of curves from which the coefficients can be read with sufficient accuracy for design purposes. The dimensions of the profile of each section are given at various stations along the chord in per cent of the chord, using as datum the line shown on the curves. The shape of the section is also shown in reasonable accuracy to enable one to more clearly visualize the section under consideration, together with its characteristics. To obtain more accurately the dimensions of the profile of each section, a separate data sheet for each section has been included which gives an additional decimal place for the greater portion of the ordinates.

The authority for the results here presented is given as the name of the laboratory at which the experiments were conducted, with the size of model, wind velocity, and date of test.

TRANSFORMATION COEFFICIENTS.

For the convenience of those who prefer to use a system of units other than the absolute system there is given below a table of transformation constants based on the standard condition adopted by the National Advisory Committee for Aeronautics of:

Temperature	= 15° C.
Pressure	= 760 mm. Hg.
Humidity	= 0.
Gravity	= 9.80 m./sec.^{2} = 32.2 ft./sec.^{2}

thus giving values of specific weight of air

$$W = 0.1225 \text{ kg./m.}^3 = 0.07636 \text{ lbs./ft.}^3$$

and of density

$\sigma=0.01250$ in the French engineering or kilogram, meter, second system.

or

$=0.00238$ in the English or foot, pound, second system.

In absolute units	$P=C\sigma V^2$.
In kg./m.² —— m./sec.	$P=.1250\ CV^2$.
In kg./m.² —— km./hr.	$P=.009645\ CV^2$.
In lbs./sq. ft. —— ft./sec.	$P=.002378\ CV^2$.
In lbs./sq. ft. —— mi./hr.	$P=.005116\ CV^2$.

INDEX.

Three separate types of index are given; chart indexes which make it possible for a designer to select the wing section most suitable for the particular design in which he is interested; a group index which is arranged in the same order as the curve sheets, i. e., by countries and laboratories at which tests were conducted, each section also being designated by a reference number; and an alphabetical index.

CHART INDEX.

In order that the designer may easily pick out a wing section which is suited to the type of machine on which he is working, four index charts are given which classify the wings according to their aerodynamic and structural properties.

In Chart No. 1 the minimum drag is plotted against the L/D at one-fourth the maximum lift. This chart should be used in choosing a wing section for a high-speed machine, the wing sections being more suited for this use the farther they are from the lower left-hand corner.

In Chart No. 2 the mean spar depth is plotted against the maximum lift in order to show the possible strength and lightness of the wing structure. The higher the maximum lift coefficient is the smaller will be the wing area and the lighter the structural weight, and in the same way the greater the depth of the spars the lighter will be their weight, so that the sections the greatest distance from the lower left-hand corner will give the lightest and strongest wings.

The maximum L/D is plotted against the maximum lift in Chart No. 3, which is of use in choosing the wing section for a slow and efficient machine. In the same way as before, the sections farthest from the lower left-hand corner are the best for this purpose.

In Chart No. 4 the L/D at two-thirds the maximum lift is plotted against the maximum lift, so that this chart can be used for choosing a section that will give an efficient climb or a long range at cruising speed. The best sections for this purpose will be the farthest from the lower left-hand corner of the chart.

GROUP INDEX.

UNITED STATES.

Aerofoil.	Wind tunnel.	Report reference No.	Aerofoil.	Wind tunnel.	Report reference No.
U. S. A. 1	M. I. T.	1	U. S. A. T. S. 7	M. I. T.	30
U. S. A. 2	M. I. T.	2	U. S. A. T. S. 8	M. I. T.	31
U. S. A. 3	M. I. T.	3	U. S. A. T. S. 9	M. I. T.	32
U. S. A. 4	M. I. T.	4	U. S. A. T. S. 10	M. I. T.	33
U. S. A. 5	M. I. T.	5	U. S. A. T. S. 11	M. I. T.	34
U. S. A. 6	M. I. T.	6	U. S. A. T. S. 12	M. I. T.	35
U. S. A. 7	M. I. T.	7	U. S. A. T. S. 13	M. I. T.	36
U. S. A. 8	M. I. T.	8	U. S. A. T. S. 14	M. I. T.	37
U. S. A. 9	M. I. T.	9	U. S. A. T. S. 15	M. I. T.	38
U. S. A. 10	M. I. T.	10	U. S. A. T. S. 16	M. I. T.	39
U. S. A. 11	M. I. T.	11	U. S. A. T. S. 17	M. I. T.	40
U. S. A. 12	M. I. T.	12	U. S. A. T. S. 18	M. I. T.	41
U. S. A. 14	M. I. T.	13	Durand Propeller No. 4	M. I. T.	42
U. S. A. 15	M. I. T.	14	Durand Propeller No. 7	M. I. T.	43
U. S. A. 16	M. I. T.	15	Durand Propeller No. 10	M. I. T.	44
U. S. A. 17	M. I. T.	16	Durand Propeller No. 13	M. I. T.	45
U. S. A. 18	M. I. T.	17	Durand Propeller No. 16	M. I. T.	46
U. S. A. 19	M. I. T.	18	Offenstein	M. I. T.	47
U. S. A. 20	M. I. T.	19	Offenstein (modified)	M. I. T.	48
U. S. A. 21	M. I. T.	20	Spad	M. I. T.	49
U. S. A. 23	M. I. T.	21	Standard Aircraft Corp. No. 48	M. I. T.	50
U. S. A. 24	M. I. T.	22	V. E. Clark	M. I. T.	51
U. S. D. 9A	M. I. T.	23	W-1	M. I. T.	52
U. S. A. T. S. 1	M. I. T.	24	Washington Navy Yard 1	W. N. Y.	53
U. S. A. T. S. 2	M. I. T.	25	Washington Navy Yard 2	W. N. Y.	54
U. S. A. T. S. 3	M. I. T.	26	Washington Navy Yard 3	W. N. Y.	55
U. S. A. T. S. 4	M. I. T.	27	Washington Navy Yard 4	W. N. Y.	56
U. S. A. T. S. 5	M. I. T.	28	Sloane	W. N. Y.	57
U. S. A. T. S. 6	M. I. T.	29			

BRITISH.

Aerofoil.	Wind tunnel.	Report reference No.	Aerofoil.	Wind tunnel.	Report reference No.
R. A. F. 3	N. P. L.	58	Naylor & Griffiths 8	N. P. L.	86
R. A. F. 4	N. P. L.	59	Naylor & Griffiths 9	N. P. L.	87
R. A. F. 5	N. P. L.	60	Naylor & Griffiths 10	N. P. L.	88
R. A. F. 6	N. P. L.	61	Naylor & Griffiths 11	N. P. L.	89
R. A. F. 6 (modified)	N. P. L.	62	Naylor & Griffiths 12	N. P. L.	90
R. A. F. 6a	N. P. L.	63	Naylor & Griffiths 13	N. P. L.	91
R. A. F. 6c	N. P. L.	64	Naylor & Griffiths 14	N. P. L.	92
R. A. F. 6c (both surfaces)	N. P. L.	65	Naylor & Griffiths 15	N. P. L.	93
R. A. F. 8	N. P. L.	66	Naylor & Griffiths 16	N. P. L.	94
R. A. F. 9	N. P. L.	67	Naylor & Griffiths 17	N. P. L.	95
R. A. F. 12	N. P. L.	68	Naylor & Griffiths 18	N. P. L.	96
R. A. F. 13	N. P. L.	69	Naylor & Griffiths 19	N. P. L.	97
R. A. F. 14	N. P. L.	70	Naylor & Griffiths 20	N. P. L.	98
R. A. F. 14 (modified)	N. P. L.	71	Naylor & Griffiths 21	N. P. L.	99
R. A. F. 15	N. P. L.	72	Naylor & Griffiths 22	N. P. L.	100
R. A. F. 15 (modified)	N. P. L.	73	Naylor & Griffiths 23	N. P. L.	101
R. A. F. 16	N. P. L.	74	Naylor & Griffiths 24	N. P. L.	102
R. A. F. 17	N. P. L.	75	Naylor & Griffiths 25	N. P. L.	103
R. A. F. 18	N. P. L.	76	Naylor & Griffiths 26	N. P. L.	104
R. A. F. 19	N. P. L.	77	Naylor & Griffiths 27	N. P. L.	105
R. A. F. 20	N. P. L.	78	Naylor & Griffiths 28	N. P. L.	106
Naylor & Griffiths 1	N. P. L.	79	Naylor & Griffiths 29	N. P. L.	107
Naylor & Griffiths 2	N. P. L.	80	Naylor & Griffiths 30	N. P. L.	108
Naylor & Griffiths 3	N. P. L.	81	Naylor & Griffiths 31	N. P. L.	109
Naylor & Griffiths 4	N. P. L.	82	A. D. No. 1	N. P. L.	110
Naylor & Griffiths 5	N. P. L.	83	A. D. No. 4	N. P. L.	111
Naylor & Griffiths 6	N. P. L.	84	N. P. L. 64	N. P. L.	112
Naylor & Griffiths 7	N. P. L.	85	Albatross	N. P. L.	113

BRITISH—Continued.

Aerofoil.	Wind tunnel.	Report reference No.	Aerofoil.	Wind tunnel.	Report reference No.
Avro	N. P. L	114	N. P. L. 73	N. P. L	135
Bristol	N. P. L	115	N. P. L. 214	N. P. L	136
B. I. R. 1a	N. P. L	116	Portholme	N. P. L	137
B. I. R. 3	N. P. L	117	Scout E	N. P. L	138
B. I. R. 33a	N. P. L	118	Sopwith	N. P. L	139
Curtiss	N. P. L	119	White	N. P. L	140
DeH-2	N. P. L	120	Cowley & Levy—A. 1	N. P. L	141
DeH-3	N. P. L	121	Cowley & Levy—A. 2	N. P. L	142
F. 2. B	N. P. L	122	Cowley & Levy—A. 3	N. P. L	143
Fairey	N. P. L	123	Cowley & Levy—A. 4	N. P. L	144
Handley Page 166	N. P. L	124	Cowley & Levy—A. 5	N. P. L	145
Handley Page 166a	N. P. L	125	Cowley & Levy—A. 6	N. P. L	146
Handley Page 166b	N. P. L	126	Cowley & Levy—A. 7	N. P. L	147
Handley Page 166c	N. P. L	127	Cowley & Levy—B. 1	N. P. L	148
N. P. L. 4	N. P. L	128	Cowley & Levy—B. 2	N. P. L	149
N. P. L. 4a	N. P. L	129	Cowley & Levy—B. 3	N. P. L	150
N. P. L. 4b	N. P. L	130	Cowley & Levy—B. 4	N. P. L	151
N. P. L. 4c	N. P. L	131	Cowley & Levy—B. 5	N. P. L	152
N. P. L. 4cα	N. P. L	132	Cowley & Levy—B. 6	N. P. L	153
N. P. L 4cβ	N. P. L	133	Cowley & Levy—B. 7	N. P. L	154
N. P. L. 4cγ	N. P. L	134			

FRENCH.

Eiffel 8	Eiffel	155	Eiffel 43	Eiffel	184
Eiffel 9	Eiffel	156	Eiffel 44 (Voisin)	Eiffel	185
Eiffel 10 (Wright)	Eiffel	157	Eiffel 45 (Buch)	Eiffel	186
Eiffel 11 (Voisin)	Eiffel	158	Eiffel 46 (Buch)	Eiffel	187
Eiffel 12 (M. Farman)	Eiffel	159	Eiffel 47 (Howard-Wright)	Eiffel	188
Eiffel 13 (Bleriot 11)	Eiffel	160	Eiffel 48 (Howard-Wright)	Eiffel	189
Eiffel 13 bis (Bleriot 11a)	Eiffel	161	Eiffel 49 (Howard-Wright)	Eiffel	190
Eiffel 14 (Breguet)	Eiffel	162	Eiffel 52 (Nieuport)	Eiffel	181
Eiffel 15 (M. Ernoult)	Eiffel	163	Eiffel 53 (Nieuport)	Eiffel	192
Eiffel 16 (M. Drzewiecki)	Eiffel	164	Eiffel 54 (Deperdussin)	Eiffel	193
Eiffel 16a	Eiffel	165	Eiffel 55 (Deperdussin)	Eiffel	194
Eiffel 16b	Eiffel	166	Eiffel 56 (Deperdussin)	Eiffel	195
Eiffel 16c	Eiffel	167	Eiffel 57	Eiffel	196
Eiffel 16d	Eiffel	168	Eiffel 58	Eiffel	197
Eiffel 17 (M. Drzewiecki)	Eiffel	169	Eiffel 59	Eiffel	198
Eiffel 18 (M. Drzewiecki)	Eiffel	170	Eiffel 60	Eiffel	199
Eiffel 30	Eiffel	171	Eiffel 61	Eiffel	200
Eiffel 31	Eiffel	172	Eiffel 62	Eiffel	201
Eiffel 32 (Lanier-Lawrance)	Eiffel	173	Dorand	Eiffel	202
Eiffel 33 (Breguet)	Eiffel	174	Halbroun 2	Eiffel	203
Eiffel 34 (Colliex)	Eiffel	175	Halbroun 3	Eiffel	204
Eiffel 35 (Dorand)	Eiffel	176	S. E. A	Eiffel	205
Eiffel 36 (Odier)	Eiffel	177	St. Cyr 1	Eiffel	206
Eiffel 37 (Kauffmann)	Eiffel	178	St. Cyr 2	Eiffel	207
Eiffel 38 (Coanda)	Eiffel	179	St. Cyr 3	Eiffel	208
Eiffel 39 (16b modified)	Eiffel	180	Turin 1	Eiffel	209
Eiffel 40	Eiffel	181	Turin 2	Eiffel	210
Eiffel 41	Eiffel	182	Bleriot Triplane	Eiffel	211
Eiffel 42	Eiffel	183			

MISCELLANEOUS.

Italian 1	Crocco	212	Italian 3	Crocco	214
Italian 2	Crocco	213			

ALPHABETICAL INDEX.

Aerofoil.	Report reference No.	Aerofoil.	Report reference No.
Naylor & Griffiths 29	107	U. S. A. 6	6
Naylor & Griffiths 30	108	U. S. A. 7	7
Naylor & Griffiths 31	109	U. S. A. 8	8
Offenstein	47	U. S. A. 9	9
Offenstein (modified)	48	U. S. A. 10	10
Portholme	137	U. S. A. 11	11
R. A. F. 3	58	U. S. A. 12	12
R. A. F. 4	59	U. S. A. 14	13
R. A. F. 5	60	U. S. A. 15	14
R. A. F. 6	61	U. S. A. 16	15
R. A. F. 6 (modified)	62	U. S. A. 17	16
R. A. F. 6a	63	U. S. A. 18	17
R. A. F. 6c	64	U. S. A. 19	18
R. A. F. 6c (both surfaces)	65	U. S. A. 20	19
R. A. F. 8	66	U. S. A. 21	20
R. A. F. 9	67	U. S. A. 23	21
R. A. F. 12	68	U. S. A. 24	22
R. A. F. 13	69	U. S. A. T. S. 1	24
R. A. F. 14	70	U. S. A. T. S. 2	25
R. A. F. 14 (modified)	71	U. S. A. T. S. 3	26
R. A. F. 15	72	U. S. A. T. S. 4	27
R. A. F. 15 (modified)	73	U. S. A. T. S. 5	28
R. A. F. 16	74	U. S. A. T. S. 6	29
R. A. F. 17	75	U. S. A. T. S. 7	30
R. A. F. 18	76	U. S. A. T. S. 8	31
R. A. F. 19	77	U. S. A. T. S. 9	32
R. A. F. 20	78	U. S. A. T. S. 10	33
S. E. A	205	U. S. A. T. S. 11	34
St.-Cyr 1	206	U. S. A. T. S. 12	35
St.-Cyr 2	207	U. S. A. T. S. 13	36
St.-Cyr 3	208	U. S. A. T. S. 14	37
Scout E	138	U. S. A. T. S. 15	38
Sloane	57	U. S. A. T. S. 16	39
Sopwith	139	U. S. A. T. S. 17	40
Spad	49	U. S. A. T. S. 18	41
Standard Aircraft Corp. 48	209	U. S. D. 9A	23
Turin 1	209	W-1	52
Turin 2	210	Washington Navy Yard 1	53
U. S. A. 1	1	Washington Navy Yard 2	54
U. S. A. 2	2	Washington Navy Yard 3	55
U. S. A. 3	3	Washington Navy Yard 4	56
U. S. A. 4	4	White	140
U. S. A. 5	5		

The following tables have been prepared to give additional decimal places for the greater portion of the ordinates:

Per cent of chord.	Ref. No. 1. U. S. A. 1. Upper.	Ref. No. 1. U. S. A. 1. Lower.	Ref. No. 2. U. S. A. 2. Upper.	Ref. No. 2. U. S. A. 2. Lower.
0	1.22	0.81	0.80	0.0
1.25	2.56	0.42	1.80	0.27
2.5	3.44	0.19	2.98	0.49
5	4.77	0.0	4.38	0.88
7.5	5.58	0.12	5.50	1.23
10	6.11	0.39	6.38	1.51
15	6.80	0.90	7.65	1.94
20	7.28	1.60	8.37	2.23
30	7.61	2.27	8.80	2.57
40	7.55	2.28	8.50	2.73
50	7.11	1.72	7.85	2.53
60	6.36	1.04	6.88	1.95
70	5.32	0.34	5.62	1.27
80	3.90	0.0	4.12	0.72
90	2.47	0.03	2.48	0.30
95	1.50	0.23	1.50	0.15
100	0.83	0.49	0.70	0.0

Per cent of chord.	Ref. No. 3. U. S. A. 3. Upper.	Ref. No. 3. U. S. A. 3. Lower.	Ref. No. 4. U. S. A. 4. Upper.	Ref. No. 4. U. S. A. 4. Lower.	Ref. No. 5. U. S. A. 5. Upper.	Ref. No. 5. U. S. A. 5. Lower.	Ref. No. 6. U. S. A. 6. Upper.	Ref. No. 6. U. S. A. 6. Lower.
0	0.79	0.0	0.81	0.0	0.73	0.33	0.72	0.72
1.25	1.87	0.13	2.43	0.25	2.10	0.17	1.77	0.08
2.5	2.64	0.30	3.44	0.51	3.03	0.03	2.39	0.0
*5	3.91	0.62	4.88	1.07	4.40	0.03	3.49	0.086
7.5	5.00	0.90	6.00	1.47	5.40	0.25	4.41	0.32
10	5.98	1.15	6.83	1.82	6.20	0.57	5.13	0.63
15	7.30	1.65	7.87	2.35	7.16	1.10	6.12	1.25
20	8.20	1.87	8.57	2.57	7.92	1.55	6.90	1.97
30	8.68	2.30	8.90	2.83	8.30	2.02	7.40	2.77
40	8.44	2.50	8.57	2.93	8.14	2.17	7.36	2.90
50	7.75	2.33	7.82	2.63	7.55	1.96	6.85	2.52
60	6.80	1.78	6.90	1.98	6.75	1.55	6.03	1.93
70	5.57	1.15	5.60	1.22	5.63	1.16	4.91	1.32
80	4.08	0.63	4.08	0.62	4.24	0.76	3.52	0.72
90	2.39	0.33	2.32	0.30	2.52	0.55	2.01	0.12
95	1.55	0.15	1.45	0.18	1.50	0.35	1.15	0.0
100	0.69	0.0	0.71	0.0	0.50	0.0	0.29	0.29
*3.5	....	....	....	....	....	0.0	....	....

Per cent of chord.	Ref. No. 7. U. S. A. 7. Upper.	Ref. No. 7. U. S. A. 7. Lower.	Ref. No. 8. U. S. A. 8. Upper.	Ref. No. 8. U. S. A. 8. Lower.	Ref. No. 9. U. S. A. 9. Upper.	Ref. No. 9. U. S. A. 9. Lower.	Ref. No. 10. U. S. A. 10. Upper.	Ref. No. 10. U. S. A. 10. Lower.
0	1.25	0.0	1.25	0.0	1.25	0.0	1.25	0.0
1.25	3.40	−1.07	4.65	−0.38	3.25	0.0	2.43	0.0
2.5	5.38	−1.58	6.33	−0.58	4.58	0.0	3.43	0.0
5	8.70	−2.32	8.73	−0.81	6.35	0.0	4.50	0.0
7.5	11.39	−2.80	10.21	−0.95	7.47	0.0	5.73	0.0
10	13.28	−3.13	11.25	−1.05	8.22	0.0	6.32	0.0
15	16.00	−3.65	12.45	−1.15	9.15	0.0	6.90	0.0
20	17.93	−3.80	13.33	−1.22	9.70	0.0	7.27	0.0
30	20.00	−4.02	13.93	−1.29	10.07	0.0	7.50	0.0
40	20.50	−3.87	13.80	−1.28	10.00	0.0	7.45	0.0
50	19.33	−3.52	13.00	−1.20	9.47	0.0	7.10	0.0
60	17.23	−3.03	11.63	−1.09	8.43	0.0	6.43	0.0
70	14.37	−2.50	9.74	−0.91	7.12	0.0	5.38	0.0
80	10.60	−1.92	7.28	−0.66	5.56	0.0	4.20	0.0
90	6.25	−1.17	4.50	−0.40	3.62	0.0	2.92	0.0
95	3.95	−0.75	2.95	−0.15	2.60	0.0	2.23	0.0
100	1.50	0.0	1.40	0.0	1.50	0.0	1.50	0.0

Per cent of chord.	Ref. No. 11. U. S. A. 11. Upper.	Ref. No. 11. U. S. A. 11. Lower.	Ref. No. 12. U. S. A. 12. Upper.	Ref. No. 12. U. S. A. 12. Lower.	Ref. No. 13. U. S. A. 14. Upper.	Ref. No. 13. U. S. A. 14. Lower.	Ref. No. 14. U. S. A. 15. Upper.	Ref. No. 14. U. S. A. 15. Lower.
0	1.25	0.0	1.25	0.0	1.25	0.633	1.22	0.63
1.25	2.32	0.0	4.73	0.0	2.277	0.467	2.56	0.46
2.5	3.21	0.0	6.88	0.0	3.100	0.333	3.44	0.33
5	4.45	0.0	9.34	0.0	4.250	0.150	4.80	0.15
7.5	5.30	0.0	11.13	0.0	4.970	0.033	5.58	0.03
10	5.85	0.0	12.27	0.0	5.417	0.0	6.11	0.0
15	6.42	0.0	13.55	0.0	6.050	0.117	6.88	0.12
20	6.78	0.0	14.56	0.0	6.567	0.333	7.28	0.33
30	7.12	0.0	15.59	0.0	6.850	0.867	7.61	0.86
40	7.05	0.0	15.07	0.0	6.867	1.000	7.55	1.00
50	6.63	0.0	14.20	0.0	6.600	0.717	7.11	0.72
60	5.95	0.0	12.71	0.0	5.967	0.277	6.36	0.28
70	5.00	0.0	10.63	0.0	5.017	0.0	5.32	0.0
80	3.92	0.0	8.03	0.0	3.850	0.110	3.90	0.11
90	2.77	0.0	4.97	0.0	2.433	0.233	2.50	0.23
95	2.12	0.0	2.65	0.0	1.620	0.370	1.67	0.37
100	1.50	0.0	1.50	0.0	0.833	0.500	0.83	0.50

Per cent of chord.	Ref. No. 15. U. S. A. 16. Ordinates. Upper.	Lower.	Ref. No. 16. U. S. A. 17. Upper.	Lower	Ref. No. 17. U. S. A. 18. Upper.	Lower.	Ref. No. 18. U. S. A. 19. Upper.	Lower.
0	0.0	0.0	0.0	0.0	1.07	1.070	0.75	0.0
1.25	2.10	−0.42	1.60	0.40	1.80	0.467	1.80	0.10
2.5	2.99	−0.64	2.55	−0.64	2.98	0.339	2.40	0.00
5	3.88	−0.76	3.70	−0.70	4.38	0.150	3.47	0.10
7.5	4.52	−0.83	4.46	−0.76	5.50	0.033	4.35	0.30
10	4.95	−0.83	4.90	−0.83	6.38	0.000	5.07	0.55
15	5.42	−0.76	5.42	−0.96	7.65	0.117	6.11	1.25
20	5.67	−0.51	5.67	−1.02	8.37	0.333	6.73	1.77
30	5.86	−0.06	5.80	−1.08	8.80	0.867	7.17	2.45
40	5.74	0.0	5.60	−1.15	8.50	1.000	6.97	2.50
50	5.29	−0.19	5.23	−1.09	7.85	0.717	6.37	1.90
60	4.65	−0.70	4.65	−1.02	6.88	0.277	5.40	1.20
70	4.01	−0.76	4.01	−0.89	5.62	0.000	4.17	0.47
80	3.25	−0.70	3.06	−0.76	4.12	0.110	3.17	0.03
90	2.23	−0.446	1.97	−0.57	2.48	0.233	2.18	0.29
95	1.37	−0.27	1.10	−0.35	1.50	0.370	2.00	0.58
100	0.0	0.0	0.0	0.0	0.74	0.740	1.50	0.0

Per cent of chord.	Ref. No. 19. U. S. A. 20. Ordinates. Upper.	Lower.	Ref. No. 20. U. S. A. 21. Upper.	Lower.	Ref. No. 21. U. S. A. 23. Upper.	Lower.	Ref. No. 22. U. S. A. 24. Upper.	Lower.
0	1.30	0.0	1.667	0.0	1.50	1.50	1.50	1.50
1.25	3.05	0.50	2.340	0.72	2.52	0.65	2.60	1.13
2.5	3.90	0.32	2.967	0.34	3.00	0.50	3.10	0.46
5	4.83	0.13	4.370	0.15	3.81	0.20	4.12	0.20
7.5	5.47	0.03	5.500	0.033	4.45	0.07	5.00	0.05
10	5.86	0.0	6.367	0.0	5.00	0.0	5.65	0.0
15	6.37	0.16	7.600	0.100	5.77	0.09	6.65	0.09
20	6.60	0.30	8.367	0.300	6.32	0.30	7.30	0.30
30	6.77	0.67	8.800	0.866	6.92	0.80	7.93	0.80
40	6.60	0.83	8.500	1.000	6.95	1.00	7.90	1.00
50	6.10	0.62	7.833	0.717	6.60	0.94	7.45	0.94
60	5.50	0.15	6.867	0.277	5.82	0.74	6.60	0.74
70	4.83	0.0	5.600	0.0	4.72	0.55	5.33	0.55
80	4.17	0.30	4.300	0.270	3.44	0.38	3.83	0.38
90	3.47	1.05	3.317	1.000	2.00	0.19	2.10	0.19
95	3.25	1.60	2.93	1.550	1.12	0.10	1.16	0.10
100	2.55	0.0	2.643	0.0	0.10	0.10	0.10	0.10

Per cent of chord.	Ref. No. 23. U. S. D. 9A. Ordinates. Upper.	Lower.	Ref. No. 24. U. S. A. T. S. 1. Upper.	Lower.	Ref. No. 25. U. S. A. T. S. 2. Upper.	Lower.	Ref. No. 26. U. S. A. T. S. 3. Upper.	Lower.
0	1.31	1.31	2.0	+2.00	2.0	+2.00	2.0	2.00
1.25	2.67	0.55	4.4	−1.30	4.4	0.0	4.4	0.30
2.5	3.46	0.33	5.5	−2.30	5.5	−0.50	5.5	0.00
5	4.62	0.11	7.4	−4.00	7.4	−1.35	7.4	0.50
7.5	5.35	0.05	8.8	−5.40	8.8	−1.85	8.8	0.85
10	5.90	0.0	10.0	−6.50	10.0	−2.25	10.0	1.15
15	6.36	0.10	11.8	−7.90	11.8	−2.80	11.8	1.60
20	6.62	0.26	13.1	−8.90	13.1	−3.40	13.1	2.00
30	6.70	0.63	14.7	−9.55	14.7	−4.00	14.7	2.30
40	6.59	0.62	14.8	−9.10	14.8	−4.10	14.8	2.50
50	6.38	0.42	13.9	−8.30	13.9	−4.00	13.9	2.40
60	5.95	0.22	12.3	−7.30	12.3	−3.85	12.3	2.10
70	5.31	0.06	10.3	−5.85	10.3	−3.00	10.3	1.80
80	4.24	0.0	7.8	−4.10	7.8	−2.00	7.8	1.30
90	2.67	0.08	4.9	−2.10	4.9	−1.00	4.9	0.70
95	1.77	0.15	3.3	−1.00	3.3	−0.45	3.3	0.30
100	0.59	0.59	1.0	+1.00	1.0	+1.00	1.0	1.0

Per cent of chord.	Ref. No. 27. U. S. A. T. S. 4. Ordinates. Upper.	Lower.	Ref. No. 28. U. S. A. T. S. 5. Upper.	Lower.	Ref. No. 29. U. S. A. T. S. 6. Upper.	Lower.	Ref. No. 30. U. S. A. T. S. 7. Upper.	Lower.
0	2.0	2.00	2.0	+2.00	2.0	+2.00	2.0	+2.00
1.25	4.4	0.45	4.4	0.0	4.4	0.10	4.4	0.60
2.5	5.5	0.20	5.5	−0.80	5.5	−0.40	5.5	0.0
5	7.4	0.80	7.4	−1.80	7.4	−0.75	7.4	0.20
7.5	8.8	1.30	8.8	−2.50	8.8	−1.10	8.8	0.5
10	10.0	1.80	10.0	−3.00	10.0	−1.45	10.0	0.74
15	11.8	2.65	11.8	−3.40	11.8	−1.80	11.8	1.00
20	13.1	3.20	13.1	−3.50	13.1	−2.00	13.1	1.25
30	14.7	3.90	14.7	−2.90	14.7	−2.10	14.7	1.47
40	14.8	3.90	14.8	−1.50	14.8	−1.35	14.8	1.00
50	13.9	3.60	13.9	−0.65	13.9	0.0	13.9	0.0
60	12.3	3.10	12.3	−0.30	12.3	+1.00	12.3	−1.20
70	10.3	2.65	10.3	−0.20	10.3	1.40	10.3	−2.00
80	7.8	2.00	7.8	−0.10	7.8	1.35	7.8	−2.50
90	4.9	1.00	4.9	0.0	4.9	0.80	4.9	−1.90
95	3.3	0.70	3.3	0.0	3.3	0.50	3.3	−1.20
100	1.2	1.20	1.0	+1.00	1.0	1.00	1.0	+1.00

Per cent of chord.	Ref. No. 31.		Ref. No. 32.		Ref. No. 33.		Ref. No. 34.	
	U. S. A. T. S. 8.		U. S. A. T. S. 9.		U. S. A. T. S. 10.		U. S. A. T. S. 11.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	2.0	2.00	1.50	1.5	1.50	1.5	1.50	1.5
1.25	4.4	+0.50	3.90	0.2	4.00	0.1	3.25	0.1
2.5	5.5	−0.20	5.75	0.0	5.60	0.0	4.25	0.0
5.0	7.4	−1.20	8.50	0.0	8.30	0.0	5.90	0.0
7.5	8.8	−1.50	10.56	0.0	10.00	0.0	7.10	0.0
10	10.0	−1.90	12.30	0.0	11.40	0.0	8.40	0.0
15	11.8	−1.90	14.50	0.0	13.30	0.0	10.10	0.0
20	13.1	−1.45	15.70	0.0	14.50	0.0	11.30	0.0
30	14.7	0.00	16.90	0.0	15.75	0.0	12.70	0.0
40	14.8	1.20	16.90	0.0	15.70	0.0	13.00	0.0
50	13.9	1.90	15.80	0.0	14.70	0.0	12.30	0.0
60	12.3	1.50	14.00	0.0	13.10	0.0	11.00	0.0
70	10.3	0.73	11.50	0.0	10.90	0.0	9.00	0.0
80	7.8	0.0	8.50	0.0	8.00	0.0	6.90	0.0
90	4.9	−0.50	5.30	0.0	5.00	0.0	4.46	0.0
95	3.3	−0.30	3.60	0.0	3.40	0.0	3.20	0.0
100	1.0	+1.00	1.00	1.0	1.00	1.0	1.00	1.0

Per cent of chord.	Ref. No. 35.		Ref. No. 36.		Ref. No. 37.		Ref. No. 38.	
	U. S. A. T. S. 12.		U. S. A. T. S. 13.		U. S. A. T. S. 14.		U. S. A. T. S. 15.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	1.50	1.5	1.40	1.4	1.50	1.5	2.00	2.0
1.25	3.30	0.1	3.50	0.1	3.50	0.1	2.00	0.4
2.5	4.35	0.0	5.40	0.0	5.25	0.0	2.20	0.0
5	6.40	0.0	8.00	0.0	7.65	0.0	3.90	0.0
*7.5	7.90	0.0	9.70	0.0	9.30	0.0	....	....
10	9.00	0.0	11.00	0.0	10.50	0.0	10.50	0.0
15	11.00	0.0	12.90	0.0	12.10	0.0	12.30	0.0
20	12.40	0.0	13.75	0.0	13.20	0.0	13.50	0.0
30	14.00	0.0	14.30	0.0	14.00	0.0	14.75	0.0
40	14.30	0.0	13.70	0.0	13.50	0.0	14.75	0.0
50	13.80	0.0	12.30	0.0	12.00	0.0	13.80	0.0
60	12.75	0.0	10.80	0.0	10.10	0.0	12.40	0.0
70	11.00	0.0	8.90	0.0	8.00	0.0	10.30	0.0
80	8.60	0.0	6.80	0.0	6.00	0.0	7.90	0.0
90	5.65	0.0	4.35	0.0	3.80	0.0	5.20	0.0
95	3.75	0.0	3.00	0.0	2.80	0.0	3.70	0.0
100	1.00	1.0	1.00	1.0	1.00	1.0	1.50	1.5
*6.75	....	....	....	....	....	....	8.70	0.0

Per cent of chord.	Ref. No. 39.		Ref. No. 40.		Ref. No. 41.		Ref. No. 42.	
	U. S. A. T. S. 16.		U. S. A. T. S. 17.		U. S. A. T. S. 18.		Durand Prop. 4.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	2.0	2.0	2.0	2.00	2.0	2.0	0.60	0.60
1.25	4.4	0.3	4.4	0.25	4.4	0.2	5.70	0.0
2.5	5.5	−0.1	5.5	−0.25	5.5	−0.3	8.25	−0.20
5.0	7.4	−1.1	7.4	−0.70	7.4	−1.0	12.70	−1.10
7.5	8.8	−1.8	8.8	−1.00	8.8	−1.5	15.70	−1.90
10	10.0	−2.5	10.0	−1.30	10.0	−1.9	18.41	−2.60
15	11.8	−3.4	11.8	−1.80	11.8	−2.6	22.70	−3.60
20	13.1	−4.3	13.1	−2.10	13.1	−3.1	25.10	−4.30
30	14.7	−5.3	14.7	−2.70	14.7	−3.7	28.10	−5.30
40	14.8	−5.8	14.8	−3.00	14.8	−4.0	28.50	−5.80
50	13.9	−5.7	13.9	−2.90	13.9	−3.7	27.00	−6.00
60	12.3	−4.9	12.3	−2.60	12.3	−3.2	23.80	−5.90
70	10.3	−3.9	10.3	−2.10	10.3	−2.6	19.50	−5.40
80	7.8	−2.7	7.8	−1.50	7.8	−1.8	14.40	−4.70
90	4.9	−1.2	4.9	−0.70	4.9	−0.9	8.50	−3.00
95	3.3	−0.55	3.3	−0.30	3.3	−0.5	5.40	−1.25
100	1.0	+1.0	1.0	+1.00	1.0	+1.0	1.20	+1.20

Per cent of chord.	Ref. No. 43.		Ref. No. 44.		Ref. No. 45.		Ref. No. 46.	
	Durand Prop. 7.		Durand Prop. 10		Durand Prop. 13		Durand Prop. 16	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	1.00	1.00	1.00	1.00	2.00	2.00	1.00	1.00
1.25	4.10	0.0	3.20	0.0	4.40	0.30	2.60	0.0
2.5	6.00	0.0	5.00	0.0	5.50	0.0	3.50	0.0
5.0	9.00	0.0	7.50	0.0	7.40	0.0	5.00	0.0
7.5	11.40	0.0	9.60	0.0	8.80	0.0	6.30	0.0
10	13.16	0.0	11.50	0.0	10.00	0.0	7.30	0.0
15	16.60	0.0	14.40	0.0	11.80	0.0	9.00	0.0
20	19.00	0.0	16.50	0.0	13.10	0.0	10.00	0.0
30	21.85	0.0	18.60	0.0	14.70	0.0	11.00	0.0
40	22.00	0.0	18.60	0.0	14.80	0.0	11.10	0.0
50	20.32	0.0	17.10	0.0	13.90	0.0	10.70	0.0
60	18.05	0.0	15.00	0.0	12.30	0.0	9.60	0.0
70	14.65	0.0	12.40	0.0	10.30	0.0	8.00	0.0
80	10.80	0.0	9.20	0.0	7.80	0.0	6.20	0.0
90	6.55	0.0	5.60	0.0	4.90	0.0	4.10	0.0
95	4.20	0.0	3.70	0.0	3.30	0.0	3.00	0.0
100	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00

Per cent of chord.	Ref. No. 47.		Ref. No. 48.		Ref. No. 49.		Ref. No. 50.	
	Offenstein.		Offenstein-Mod.		Spad.		St. Air. Co. No. 48.	
	Ordinates.							
	Upper.	Lower.	Upper	Lower.	Upper.	Lower.	Upper.	Lower.
0	1.00	1.00	1.00	1.00	0.26	0.26	1.27	1.27
1.25	1.83	0.55	1.96	0.570	1.55	0.07	2.10	0.45
2.5	2.50	0.70	2.50	0.274	2.14	0.03	2.68	0.24
5	3.57	0.04	3.572	0.0357	3.00	0.08	3.57	0.0
7.5	4.21	0.04	4.210	0.0357	3.75	0.16	4.20	0.04
10	4.65	0.11	4.65	0.1071	4.37	0.26	4.68	0.15
15	5.20	0.32	5.25	0.310	5.31	0.55	5.30	0.36
20	5.56	0.48	5.56	0.478	5.95	0.85	5.62	0.56
30	5.84	0.67	5.84	0.666	6.60	1.25	5.85	0.67
40	5.79	0.63	5.79	0.631	6.67	1.31	5.84	0.60
50	5.51	0.51	5.51	0.512	6.28	1.10	5.55	0.48
60	5.04	0.43	5.045	0.429	5.53	0.82	5.06	0.37
70	4.36	0.32	4.360	0.3214	4.51	0.51	4.35	0.30
80	3.42	0.19	3.419	0.1904	3.26	0.29	3.46	0.22
90	1.96	0.83	1.962	0.0834	1.80	0.15	2.34	0.11
95	1.30	0.78	1.050	0.0190	1.06	0.07	1.64	0.04
100	0.0	0.0	0.00	0.000	0.32	0.0	0.48	0.48

Per cent of chord.	Ref. No. 51.		Ref. No. 52.		Ref. No. 53.		Ref. No. 54.	
	V. E. Clark.		W-1.		W. N. Y. 1.		W. N. Y. 2.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper	Lower.	Upper.	Lower.
0	1.0	0.8	0.0	0.0	1.018	1.018	1.018	1.018
1.25	2.6	0.7	1.25	−1.10	1.802	0.548	1.802	0.548
2.5	3.7	0.5	1.95	−1.40	2.508	0.235	2.508	0.235
5	4.8	0.0	3.00	−1.90	3.605	0.0	3.605	0.0
7.5	5.6	0.3	3.90	−2.15	4.467	0.235	4.467	0.235
10	6.3	0.7	4.50	−2.50	5.265	0.687	5.265	0.687
15	7.0	1.7	5.35	−2.85	6.270	1.489	6.270	1.489
20	7.3	2.2	5.70	−3.00	7.175	2.228	7.175	2.228
30	7.5	2.7	5.80	−3.00	7.870	2.954	7.870	2.954
40	7.1	2.3	5.70	−3.00	7.680	2.725	7.450	2.725
50	6.5	1.8	5.20	−2.80	7.060	2.060	6.560	2.062
60	5.5	1.3	4.50	−2.50	6.070	1.210	5.310	1.210
70	4.7	0.5	3.60	−2.10	4.770	0.573	4.100	0.573
80	3.5	0.2	2.75	−1.75	3.460	0.229	2.905	0.229
90	2.2	0.0	1.75	−1.25	2.150	0.051	1.810	0.051
95	1.3	0.1	1.30	−1.10	1.450	0.080	1.260	0.080
100	0.8	0.7	0.0	0.0	0.814	0.229	0.814	0.229

Per cent of chord.	Ref. No. 55.		Ref. No. 56.		Ref. No. 57		Ref. No. 58.	
	W. N. Y. 3.		W. N. Y. 4.		Sloane.		R. A. F. 3.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	1.018	1.018	0.0	0.0	0.71	0.71	0.80	0.80
1.25	1.802	0.548	1.179	0.0	1.80	0.26	2.00	0.0
2.5	2.508	0.235	2.044	0.0	2.56	0.07	3.00	0.0
5	3.605	0.0	3.880	0.0	3.40	0.01	4.40	0.90
7.5	4.467	0.235	4.409	0.235	3.95	0.05	5.50	1.30
10	5.265	0.687	5.265	0.687	4.38	0.12	6.40	1.60
15	6.270	1.489	6.270	1.489	4.93	0.36	7.68	2.10
20	7.175	2.228	7.175	2.228	5.25	0.51	8.40	2.40
30	7.870	2.954	7.870	2.954	5.62	0.63	8.80	2.90
40	7.190	2.725	7.450	2.725	5.62	0.55	8.50	3.20
50	5.925	2.062	6.560	2.062	5.30	0.46	7.80	3.10
60	4.575	1.210	5.310	1.210	4.82	0.38	6.90	2.60
70	3.356	0.573	4.100	0.573	4.10	0.33	5.60	2.10
80	2.355	0.229	2.905	0.229	3.26	0.21	4.10	1.40
90	1.502	0.051	1.810	0.051	2.14	0.08	2.40	0.80
95	1.150	0.080	1.260	0.080	1.43	0.04	1.60	0.40
100	0.814	0.229	0.814	0.229	0.58	0.0	0.70	0.0

Per cent of chord.	Ref. No. 59.		Ref. No. 60.		Ref. No. 61.		Ref. No. 62.	
	R. A. F. 4.		R. A. F. 5.		R. A. F. 6.		R. A. F. 6-Mod.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.80	0.80	0.80	0.80	0.50	0.0	0.50	0.0
1.25	2.20	0.05	2.25	0.20	2.30	0.10	2.00	0.05
2.5	3.10	0.0	3.10	0.0	3.21	0.0	3.05	0.13
5	4.30	0.90	4.30	−0.50	4.43	0.22	4.50	0.24
7.5	5.25	1.30	5.20	−0.75	5.35	0.35	5.40	0.34
10	6.00	1.60	6.00	−0.80	6.01	0.41	6.10	0.45
15	6.90	2.00	6.90	−0.55	7.08	0.70	7.15	0.63
20	7.40	2.10	7.40	−0.10	7.40	0.68	7.80	0.70
30	7.50	2.20	7.50	+1.70	7.59	0.77	8.20	0.80
40	7.20	2.10	7.20	2.20	7.47	0.68	7.70	0.80
50	6.60	1.90	6.60	2.00	7.11	0.53	7.10	0.75
60	5.90	1.60	5.90	1.60	6.51	0.41	6.43	0.64
70	4.90	1.30	4.90	1.30	5.65	0.30	5.80	0.80
80	3.80	0.90	3.80	0.90	4.42	0.19	5.33	1.45
90	2.50	0.40	2.50	0.40	2.73	0.09	4.90	2.50
95	1.70	0.22	1.65	0.30	1.85	0.05	4.76	3.20
100	0.80	0.10	0.80	0.10	0.50	0.0	4.30	4.30

Per cent of chord.	REF. NO. 63.		REF. NO. 64.		REF. NO. 65.		REF. NO. 66.	
	R. A. F. 6a.		R. A. F. 6c.		R. A. F. 6c (BS)		R. A. F. 8.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	5.00	0.0	0.0	0.0	0.0	0.0	0.0	0.0
1.25	2.30	0.0	2.25	0.0	2.25	−2.25	3.30	0.0
2.5	3.21	0.0	3.10	0.11	3.10	−3.10	4.10	0.10
5	4.43	0.0	4.53	0.25	4.53	−4.53	5.30	0.35
7.5	5.35	0.0	5.50	0.37	5.50	−5.50	6.00	0.57
10	6.01	0.0	6.17	0.45	6.17	−6.17	6.47	0.76
15	7.08	0.0	6.95	0.67	6.95	−6.95	7.20	1.00
20	7.40	0.0	7.40	0.80	7.40	−7.40	7.52	0.95
30	7.59	0.0	7.85	0.90	7.85	−7.85	7.58	0.85
40	7.47	0.0	7.70	0.80	7.70	−7.70	7.40	0.70
50	7.11	0.0	7.30	0.65	7.30	−7.30	7.10	0.55
60	6.51	0.0	6.80	0.50	6.80	−6.80	6.50	0.30
70	5.65	0.0	5.90	0.35	5.90	−5.90	5.60	0.18
80	4.42	0.0	4.70	0.20	4.70	−4.70	4.50	0.0
90	2.73	0.0	3.20	0.10	3.20	−3.20	2.80	0.0
95	1.85	0.0	2.30	0.05	2.30	−2.30	1.85	0.0
100	0.50	0.0	0.0	0.0	0.0	0.0	0.0	0.0

Per cent of chord.	REF. NO. 67.		REF. NO. 68.		REF. NO. 69.		REF. NO. 70.	
	R. A. F. 9.		R. A. F. 12.		R. A. F. 13.		R. A. F. 14.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.0	0.0	0.0	0.0	0.0	0.0	0.68	0.68
1.25	2.75	0.02	2.00	−0.46	2.05	−0.50	1.80	0.13
2.5	3.52	0.06	2.74	−0.63	2.85	−0.65	2.54	0.05
5	4.82	0.20	3.83	−0.89	3.95	−0.89	3.65	0.03
7.5	5.70	0.30	4.55	−1.05	4.80	−1.06	4.52	0.16
10	6.33	0.35	5.05	−1.22	5.34	−1.18	5.25	0.39
15	7.00	0.40	5.55	−1.33	6.00	−1.33	6.18	0.94
20	7.23	0.37	5.76	−1.40	6.50	−1.38	6.63	1.22
30	7.33	0.35	5.75	−1.38	7.05	−1.42	7.06	1.39
40	7.10	0.30	5.59	−1.36	6.88	−1.36	6.82	1.27
50	6.70	0.25	5.27	−1.28	6.25	−1.28	6.50	1.06
60	6.10	0.20	4.80	−1.17	4.60	−1.17	6.08	0.85
70	5.20	0.16	4.58	−0.99	4.08	−0.99	5.43	0.62
80	4.00	0.10	3.79	−0.76	3.14	−0.76	4.28	0.42
90	2.50	0.07	2.48	−0.47	1.96	−0.47	2.65	0.19
95	1.67	0.03	1.62	0.26	1.24	−0.27	1.70	0.17
100	0.0	0.0	0.0	0.0	0.0	0.0	0.38	0.38

Per cent of chord.	REF. NO. 71.		REF. NO. 72.		REF. NO. 73.		REF. NO. 74.	
	R. A. F. 14–Mod.		R. A. F. 15.		R. A. F. 15–Mod.		R. A. F. 16.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.66	0.66	0.20	+0.20	1.33	1.33	0.30	+0.30
1.25	1.74	0.42	1.90	−0.45	3.00	0.55	2.20	−0.47
2.5	2.55	0.40	2.80	−0.73	3.75	0.26	3.15	−0.75
5	3.82	0.40	3.90	−0.90	4.85	0.10	4.42	−0.90
7.5	4.76	0.42	4.60	−1.00	5.56	0.05	5.20	−1.00
10	5.50	0.40	5.05	−1.00	6.00	0.0	5.70	−1.03
15	6.40	0.80	5.58	−0.80	6.55	0.17	6.25	−0.80
20	6.80	1.26	5.76	−0.50	6.70	0.50	6.50	−0.50
30	6.90	1.24	5.80	−0.10	6.70	0.80	6.58	−0.10
*40	6.78	1.12	5.58	−0.05	6.49	0.91	6.32	0.0
50	6.47	0.86	5.17	−0.28	6.13	0.67	6.00	0.0
60	6.00	0.40	4.68	−0.47	5.62	0.34	5.49	0.0
*70	5.37	−0.08	4.07	−0.62	4.98	0.10	4.70	0.0
*80	4.16	−0.50	3.28	−0.67	4.22	0.16	3.80	0.0
90	2.63	−0.40	2.24	−0.35	3.33	0.55	2.68	0.0
95	1.66	−0.15	1.63	−0.20	2.77	0.76	1.90	0.0
100	0.40	0.40	0.30	+0.30	1.33	1.33	0.30	+0.30
*35	....	....	5.74	0.0	....	....	....	....
*36.7	....	....	....	....	....	....	....	0.0
*68	....	0.0	....	....	....	....	....	....
*73.3	....	....	....	....	....	0.0	....	....

Per cent of chord.	REF. NO. 75.		REF. NO. 76.		REF. NO. 77.		REF. NO. 78.	
	R. A. F. 17.		R. A. F. 18.		R. A. F. 19.		R. A. F. 20.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.30	+0.30	0.70	0.70	1.20	1.20	0.0	0.0
1.25	1.80	−0.32	1.82	0.25	3.90	0.22	0.90	−0.53
2.5	2.76	−0.48	2.41	0.05	5.80	0.03	1.60	−0.68
*5	3.90	−0.70	3.30	0.0	8.50	0.10	2.70	−0.90
7.5	4.55	−0.80	4.00	0.05	10.38	0.62	3.50	−0.96
10	5.00	−0.85	4.50	0.16	11.75	1.40	4.06	−1.05
15	5.54	−0.87	5.20	0.52	13.70	3.15	4.86	−1.20
20	5.78	−0.82	5.62	0.90	14.72	4.85	5.43	−1.27
30	5.80	−1.00	5.90	1.40	15.20	6.90	6.00	−1.40
40	5.62	−0.95	5.90	1.30	14.70	7.50	6.00	−1.50
50	5.20	−0.87	5.60	1.00	13.40	7.20	5.70	−1.50
60	4.65	−0.80	5.10	0.70	11.70	6.20	5.10	−1.40
70	3.98	−0.70	4.40	0.40	9.50	5.00	4.20	−1.40
80	3.22	−0.60	3.50	0.20	7.10	3.40	3.10	−1.10
90	2.24	−0.40	2.20	0.0	4.30	1.70	1.80	−0.80
95	1.64	−0.30	1.50	0.0	2.60	0.80	1.10	−0.60
*100	0.42	+0.42	0.30	0.30	0.60	0.60	0.0	0.0
*3.5	....	....	....	....	....	0.0	....	....
*99	....	....	....	....	....	0.0	....	....

Per cent of chord.	Ref. No. 79.		Ref. No. 80.		Ref. No. 81.		Ref. No. 82.	
	N. & G. 1.		N. & G. 2.		N. & G. 3.		N. & G. 4.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
1.25	1.13	0.0	1.44	0.0	1.69	0.0	2.25	0.0
2.5	1.58	0.0	2.01	0.0	2.36	0.0	3.14	0.0
5	2.29	0.0	2.94	0.0	3.44	0.0	4.58	0.0
7.5	2.78	0.0	3.56	0.0	4.16	0.0	5.55	0.0
10	3.07	0.0	3.94	0.0	4.61	0.0	6.14	0.0
15	3.47	0.0	4.45	0.0	5.21	0.0	6.94	0.0
20	3.70	0.0	4.74	0.0	5.55	0.0	7.40	0.0
30	3.89	0.0	4.99	0.0	5.84	0.0	7.78	0.0
*40	3.85	0.0	4.94	0.0	5.78	0.0	7.70	0.0
50	3.65	0.0	4.68	0.0	5.48	0.0	7.30	0.0
60	3.40	0.0	4.36	0.0	5.10	0.0	6.80	0.0
70	2.95	0.0	3.78	0.0	4.40	0.0	5.90	0.0
80	2.35	0.0	3.01	0.0	3.50	0.0	4.70	0.0
90	1.60	0.0	2.05	0.0	2.40	0.0	3.20	0.0
95	1.15	0.0	1.47	0.0	1.73	0.0	2.30	0.0
100	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
*32	3.90	0.0	5.00	0.0	5.85	0.0	7.80	0.0

Per cent of chord.	Ref. No. 83.		Ref. No. 84.		Ref. No. 85.		Ref. No. 86.	
	N. & G. 5.		N. & G. 6.		N. & G. 7.		N. & G. 8.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
1.25	2.37	0.0	2.53	0.0	1.78	0.0	1.90	0.0
2.5	3.40	0.0	3.71	0.0	2.55	0.0	2.78	0.0
5	4.85	0.0	5.24	0.0	3.64	0.0	3.93	0.0
7.5	5.83	0.0	6.16	0.0	4.37	0.0	4.62	0.0
10	6.44	0.0	6.77	0.0	4.83	0.0	5.08	0.0
15	7.20	0.0	7.39	0.0	5.40	0.0	5.54	0.0
20	7.52	0.0	7.73	0.0	5.64	0.0	5.80	0.0
*30	7.79	0.0	7.76	0.0	5.84	0.0	5.82	0.0
40	7.59	0.0	7.47	0.0	5.70	0.0	5.60	0.0
50	7.16	0.0	7.03	0.0	5.37	0.0	5.27	0.0
60	6.59	0.0	6.47	0.0	4.94	0.0	4.85	0.0
70	5.67	0.0	5.60	0.0	4.25	0.0	4.21	0.0
80	4.53	0.0	4.45	0.0	3.40	0.0	3.34	0.0
90	3.11	0.0	3.04	0.0	2.33	0.0	2.28	0.0
95	2.27	0.0	2.20	0.0	1.70	0.0	1.65	0.0
100	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
*24	....	....	7.80	0.0	....	....	5.85	0.0
*28	7.80	0.0	....	....	5.85	0.0	....	....

Per cent of chord.	Ref. No. 87.		Ref. No. 88.		Ref. No. 89.		Ref. No. 90.	
	N. & G. 9.		N. & G. 10.		N. & G. 11.		N. & G. 12.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.0	−0.0	0.0	0.0	0.0	0.0	0.0	0.0
1.25	1.90	−0.65	1.90	−0.325	1.90	−1.14	1.90	−0.76
2.5	2.78	−0.95	2.78	−0.475	2.78	−1.75	2.78	−1.17
5	3.93	−1.34	3.93	−0.670	3.93	−2.50	3.93	−1.67
7.5	4.62	−1.58	4.62	−0.790	4.62	−2.84	4.62	−1.89
10	5.08	−1.74	5.08	−0.870	5.08	−2.99	5.08	−1.99
•*15	5.54	−1.89	5.54	−0.945	5.54	−2.68	5.54	−1.79
20	5.80	−1.98	5.80	−0.990	5.80	−1.80	5.80	−1.20
*30	5.82	−1.99	5.82	−0.995	5.82	−0.20	5.82	−0.133
40	5.60	−1.91	5.60	−0.955	5.60	0.0	5.60	0.0
50	5.27	−1.80	5.27	−0.900	5.27	0.0	5.27	0.0
60	4.85	−1.66	4.85	−0.830	4.85	0.0	4.85	0.0
70	4.21	−1.44	4.21	−0.720	4.21	0.0	4.21	0.0
80	3.34	−1.14	3.34	−0.570	3.34	0.0	3.34	0.0
90	2.28	−0.78	2.28	−0.340	2.28	0.0	2.28	0.0
95	1.65	−0.56	1.65	−0.280	1.65	0.0	1.65	0.0
100	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
•*10.7	....	....	....	....	....	−3.0	....	−2.0
*24	5.85	−0.20	5.85	−0.10	5.85	....	5.85	....

Per cent of chord.	Ref. No. 91.		Ref. No. 92.		Ref. No. 93.		Ref. No. 94.	
	N. & G. 13.		N. & G. 14.		N. & G. 15.		N. & G. 16.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
1.25	1.90	−0.380	1.90	−0.94	1.90	−0.470	1.90	−1.30
2.5	2.78	−0.580	2.78	−1.36	2.78	−0.680	2.78	−1.50
5	3.93	−0.830	3.93	−1.83	3.93	−0.915	3.93	−1.38
7.5	4.62	−0.950	4.62	−2.00	4.62	−1.000	4.62	−1.20
10	5.08	−0.997	5.08	−1.85	5.08	−0.925	5.08	−1.07
•*15	5.54	−0.890	5.54	−1.20	5.54	−0.600	5.54	−0.70
20	5.80	−0.600	5.80	−0.70	5.80	−0.350	5.80	−0.30
*30	5.82	−0.067	5.82	−0.10	5.82	−0.050	5.82	−0.06
*40	5.60	0.0	5.60	0.0	5.60	0.0	5.60	0.0
50	5.27	0.0	5.27	0.0	5.27	0.0	5.27	0.0
60	4.85	0.0	4.85	0.0	4.85	0.0	4.85	0.0
70	4.21	0.0	4.21	0.0	4.21	0.0	4.21	0.0
80	3.34	0.0	3.34	0.0	3.34	0.0	3.34	0.0
90	2.28	0.0	2.28	0.0	2.28	0.0	2.28	0.0
95	1.65	0.0	1.65	0.0	1.65	0.0	1.65	0.0
100	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
•*10.7	....	−1.0	....	....	....	....	....	....
*24	5.85	....	5.85	....	5.85	....	5.85	....
*33.3	....	....	....	0.0	....	....	....	....

Per cent of chord.	Ref. No. 95.		Ref. No. 96.		Ref. No. 97.		Ref. No. 98.	
	N. & G. 17.		N. & G. 18.		N. & G. 19.		N. & G. 20.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
1.25	1.90	−0.866	1.90	−0.433	1.13	−0.380	1.13	−0.470
2.5	2.78	−1.000	2.78	−0.500	1.58	−0.580	1.58	−0.680
5	3.93	−0.920	3.93	−0.460	2.29	−0.830	2.29	−0.915
7.5	4.62	−0.800	4.62	−0.400	2.78	−0.950	2.78	−1.000
10	5.08	−0.713	5.08	−0.357	3.07	−0.997	3.07	−0.925
*15	5.54	−0.466	5.54	−0.233	3.47	−0.890	3.47	−0.600
20	5.80	−0.200	5.80	−0.100	3.70	−0.600	3.70	−0.350
*30	5.82	−0.040	5.82	−0.020	3.89	−0.067	3.89	−0.050
*40	5.60	0.0	5.60	0.0	3.85	0.0	3.85	0.0
50	5.27	0.0	5.27	0.0	3.65	0.0	3.65	0.0
60	4.85	0.0	4.85	0.0	3.40	0.0	3.40	0.0
70	4.21	0.0	4.21	0.0	2.95	0.0	2.95	0.0
80	3.34	0.0	3.34	0.0	2.35	0.0	2.35	0.0
90	2.28	0.0	2.28	0.0	1.60	0.0	1.60	0.0
95	1.65	0.0	1.65	0.0	1.15	0.0	1.15	0.0
100	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
*10.7	....	....	....	....	....	−1.0	....	....
*24	5.85	....	5.85	....	....	....	....	....
*32	....	....	....	....	3.90	....	3.90	....
*33.3	....	....	....	....	....	0.0	....	0.0

Per cent of chord.	Ref. No. 99.		Ref. No. 100.		Ref. No. 101.		Ref. No. 102.	
	N. & G. 21.		N. & G. 22.		N. & G. 23.		N. & G. 24.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
1.25	1.13	−0.235	1.13	−0.290	1.13	−0.145	2.25	−0.290
2.5	1.58	−0.340	1.58	−0.400	1.58	−0.200	3.14	−0.400
5	2.29	−0.458	2.29	−0.590	2.29	−0.295	4.58	−0.590
7.5	2.78	−0.500	2.78	−0.760	2.78	−0.380	5.55	−0.760
10	3.07	−0.463	3.07	−0.790	3.07	−0.395	6.14	−0.790
15	3.47	−0.300	3.47	−0.890	3.47	−0.445	6.94	−0.890
20	3.70	−0.175	3.70	−0.950	3.70	−0.475	7.40	−0.950
30	3.89	−0.025	3.89	−0.997	3.89	−0.498	7.78	−0.997
*40	3.85	0.0	3.85	−0.990	3.85	−0.495	7.70	−0.990
50	3.65	0.0	3.65	−0.940	3.65	−0.470	7.30	−0.940
60	3.40	0.0	3.40	−0.870	3.40	−0.435	6.80	−0.870
70	2.95	0.0	2.95	−0.760	2.95	−0.380	5.90	−0.760
80	2.35	0.0	2.35	−0.600	2.35	−0.300	4.70	−0.600
90	1.60	0.0	1.60	−0.410	1.60	−0.205	3.20	−0.410
95	1.15	0.0	1.15	−0.290	1.15	−0.145	2.30	−0.290
100	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
*32	3.90	....	3.90	−1.000	3.90	−0.500	7.80	−1.000
*33.3	....	0.0	....	....	....	....	....	....

Per cent of chord.	Ref. No. 103.		Ref. No. 104.		Ref. No. 105.		Ref. No. 106.	
	N. & G. 25.		N. & G. 26.		N. & G. 27.		N. & G. 28.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.0	0.0	0.0	0.0	0.834	0.834	1.67	1.67
1.25	1.69	−0.290	1.44	−0.290	....	....	....	....
2.5	2.36	−0.400	2.01	−0.400	....	....	....	....
5.0	3.44	−0.590	2.94	−0.590	4.580	0.0	....	....
7.5	1.16	−0.760	3.56	−0.760	5.550	0.0	....	....
10	4.61	−0.790	3.94	−0.790	6.140	0.0	6.14	0.0
15	5.21	−0.890	4.45	−0.890	6.940	0.0	6.94	0.0
20	5.55	−0.950	4.74	−0.950	7.400	0.0	7.40	0 0
30	5.84	−0.997	4.99	−0.997	7.780	0.0	7.78	0.0
*40	5.78	−0.990	4.94	−0.990	7.700	0.0	7.70	0.0
50	5.48	−0.940	4.68	−0.940	7.300	0.0	7.30	0.0
60	5.10	−0.870	4.36	−0.870	6.800	0.0	6.80	0.0
70	4.40	−0.760	3.78	−0.760	5.900	0.0	5.90	0.0
80	3.50	−0.600	3.01	−0.600	4.700	0.0	4.70	0.0
90	2.40	−0.410	2.05	−0.410	3.200	0.0	3.20	0.0
95	1.73	−0.290	1.47	−0.290	2.300	0.0	2.30	0.0
100	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
*32	5.85	−1.000	5.00	−1.000	7.800	0.0	7.80	0.0

Per cent of chord.	Ref. No. 107.		Ref. No. 108.		Ref. No. 109.		Ref. No. 110.	
	N. & G. 29.		N. & G. 30.		N. & G. 31.		A. D. No. 1.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	3.34	3.34	5.00	5.00	−0.834	−0.834	0.0	0.0
1.25	....	....	....	....	....	....	1.10	−0.780
2.5	....	....	....	....	....	....	1.63	−1.000
5	....	....	....	....	....	....	2.62	−1.160
7.5	....	....	....	....	....	....	3.50	−1.270
10	....	....	....	....	6.14	0.0	4.21	−1.360
15	6.94	0.0	....	....	6.94	0.0	5.20	−1.450
20	7.40	0.0	7.40	0.0	7.40	0.0	5.67	−1.500
30	7.78	0.0	7.78	0.0	7.78	0.0	6.32	−1.600
*40	7.70	0.0	7.70	0.0	7.70	0.0	6.38	−1.620
50	7.30	0.0	7.30	0.0	7.30	0.0	6.08	−1.610
60	6.80	0.0	6.80	0.0	6.80	0.0	5.38	−1.600
70	5.90	0.0	5.90	0.0	5.90	0.0	4.50	−1.400
80	4.70	0.0	4.70	0.0	4.70	0.0	3.43	−0.900
90	3.20	0.0	3.20	0.0	3.20	0.0	1.90	−0.700
95	2.30	0.0	2.30	0.0	2.30	0.0	1.10	−0.620
100	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
*32	7.80	0.0	7.80	0.0	7.80	0.0	....	....

Per cent of chord.	Ref. No. 111.		Ref. No. 112.		Ref. No. 113.		Ref. No. 114.	
	A. D. 4.		N. P. L. 64		Albatross.		Avro.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.0	0.0	0.0	0.0	0.74	0.74	0.34	0.34
*1.25	1.600	−0.700	2.3	0.0	2.20	0.0	1.28	0.015
2.5	2.403	−0.945	3.3	0.0	3.20	0.20	2.08	0.06
5	3.717	−1.125	4.6	0.0	5.00	0.60	3.30	0.17
7.5	4.60	−1.300	5.5	0.0	6.20	0.95	4.30	0.26
10	5.310	−1.440	6.2	0.0	7.05	1.25	5.20	0.42
15	6.300	−1.647	7.2	0.0	8.25	1.75	6.68	0.88
20	6.858	−1.737	7.6	0.0	9.00	2.12	7.50	1.25
30	7.200	−1.800	8.0	0.0	9.64	2.50	8.30	1.78
40	6.912	−1.773	7.2	0.0	9.45	2.62	8.43	1.93
50	6.300	−1.710	7.1	0.0	8.55	2.50	8.05	1.76
60	5.472	−1.575	6.2	0.0	7.40	2.17	7.20	1.40
70	4.383	−1.368	5.1	0.0	6.02	1.76	5.75	1.04
80	3.177	−1.125	3.9	0.0	4.30	1.25	4.00	0.66
90	1.845	−0.882	2.5	0.0	2.47	0.65	2.23	0.30
95	1.060	−0.660	1.6	0.0	1.50	0.30	1.30	0.08
*100	0.0	0.0	0.0	0.0	0.35	0.35	0.20	0.20
*1	....	....	....	....	....	....	....	0.0
*96	....	....	....	....	....	....	....	0.0

Per cent of chord.	Ref. No. 115.		Ref. No. 116.		Ref. No. 117.		Ref. No. 118.	
	Bristol.		B. I. R. 1a.		B. I. R. 3.		B. I. R. 33a.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	1.74	1.39	0.0	0.0	0.0	0.0	0.0	0.0
1.25	2.38	1.00	2.18	0.0	2.18	1.20	2.15	0.21
2.5	2.90	0.75	3.40	0.0	3.40	2.05	3.42	0.34
5	3.81	0.40	5.30	0.0	5.30	3.20	5.30	0.53
7.5	4.50	0.18	6.65	0.0	6.65	3.95	6.52	0.65
10	5.08	0.04	7.62	0.0	7.62	4.55	7.50	0.75
15	5.92	0.08	8.80	0.0	8.80	5.34	8.87	0.88
20	6.41	0.35	9.50	0.0	9.50	5.74	9.50	0.95
30	6.77	0.83	9.98	0.0	9.98	6.00	9.95	0.99
*40	6.76	1.05	9.88	0.0	9.88	5.84	9.80	0.98
50	6.65	0.99	9.16	0.0	9.16	5.50	9.05	0.90
60	6.14	0.71	8.10	0.0	8.10	4.80	8.00	0.80
70	5.31	0.27	6.64	0.0	6.64	3.92	6.50	0.65
80	4.14	0.0	4.78	0.0	4.78	2.85	4.75	0.47
90	2.72	0.12	2.65	0.0	2.65	1.50	2.60	0.26
95	1.90	0.40	1.50	0.0	1.50	0.78	1.42	0.14
100	1.04	0.69	0.0	0.0	0.0	0.0	0.0	0.0
*33.2	....	....	10.00	0.0	10.00	0.0	....	....

Per cent of chord.	Ref. No. 119.		Ref. No. 120.		Ref. No. 121.		Ref. No. 122.	
	Curtiss.		DeH–2.		DeH–3.		F. 2. B.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	1.43	1.43	2.00	2.00	2.30	2.30	0.8	0.8
1.25	3.50	0.22	2.80	1.30	3.15	1.60	1.87	0.3
2.5	4.25	0.07	3.40	0.95	3.78	1.12	2.7	0.1
*5	5.21	0.04	4.33	0.58	4.67	0.50	4.0	0.0
7.5	5.86	0.18	4.93	0.45	5.26	0.20	4.92	0.03
10	6.30	0.40	5.36	0.38	5.63	0.10	5.7	0.1
15	6.84	0.73	5.83	0.43	6.10	0.0	6.8	0.5
20	7.21	0.90	6.10	0.60	6.35	0.07	7.3	1.0
30	7.40	0.90	6.45	0.83	6.53	0.38	7.6	1.7
40	7.21	0.83	6.50	0.86	6.57	0.57	7.6	1.6
50	6.83	0.73	6.44	0.87	6.47	0.57	7.3	1.4
60	6.13	0.52	5.73	0.77	6.03	0.50	6.1	1.1
70	5.28	0.42	5.31	0.62	5.35	0.35	5.8	0.8
80	4.08	0.28	4.07	0.43	4.20	0.17	4.5	0.4
*90	2.35	0.15	2.53	0.23	2.73	0.0	2.9	0.1
95	1.37	0.07	1.70	0.12	1.93	0.10	1.8	0.0
*100	0.13	0.13	0.40	0.40	0.70	0.70	0.3	0.3
*3.3	....	0.0	....	....	....	....	....	....
*88.0	....	....	....	....	....	0.0	....	....
*98.3	....	0.0	....	....	....	....	....	....

Per cent of chord.	Ref. No. 123.		Ref. No. 124.		Ref. No. 125.		Ref. No. 126.	
	Fairey.		H. P. 166.		H. P. 166a.		H. P. 166b.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.13	0.13	0.70	0.70	0.0	0.0	0.0	0.0
*1.25	1.90	0.08	2.62	0.02	2.62	0.33	2.62	0.66
2.5	3.06	0.03	3.55	0.05	3.55	0.33	3.55	0.63
5	4.86	0.07	5.14	0.15	5.14	0.35	5.14	0.60
7.5	6.20	0.40	6.25	0.30	6.25	0.40	6.25	0.60
10	7.10	0.70	7.00	0.40	7.00	0.50	7.00	0.60
15	8.10	1.10	8.00	0.60	8.00	0.63	8.00	0.61
20	8.85	1.46	8.50	0.72	8.50	0.72	8.50	0.72
30	9.40	1.80	8.84	0.88	8.84	0.88	8.84	0.88
40	9.22	1.80	8.72	0.96	8.72	0.96	8.72	0.96
50	8.72	1.67	8.20	0.85	8.20	0.85	8.20	0.85
60	7.83	1.47	7.36	0.68	7.36	0.68	7.36	0.68
70	6.45	1.10	6.28	0.46	6.28	0.46	6.28	0.46
80	4.85	0.70	4.90	0.30	4.90	0.30	4.90	0.30
90	2.90	0.33	3.27	0.14	3.27	0.14	3.27	0.14
95	1.83	0.17	2.28	0.08	2.28	0.08	2.28	0.08
100	0.67	0.0	0.0	0.0	0.0	0.0	0.0	0.0
*0.7	....	....	....	0.0	....	....	....	....

Per cent of chord.	Ref. No. 127.		Ref. No. 128.		Ref. No. 129.		Ref. No. 130.	
	H. P. 166c.		N. P. L. 4.		N. P. L. 4a.		N. P. L. 4b.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
1.25	2.62	0.94	2.30	0.0	2.3	0.46	2.30	0.92
2.5	3.55	0.88	3.73	0.0	3.73	0.746	3.73	1.49
5	5.14	0.75	5.70	0.0	5.70	1.14	5.70	2.28
7.5	6.25	0.71	7.13	0.0	7.13	1.43	7.13	2.85
10	7.00	0.70	8.00	0 0	8.00	1.60	8.00	3.20
15	8.00	0.70	9.10	0.0	9.10	1.82	9.10	3.64
20	8.50	0.72	9.70	0.0	9.70	1.94	9.70	3.88
30	8.84	0.88	9.90	0.0	9.90	1.98	9.90	3.96
40	8.72	0.96	9.70	0.0	9.70	1.94	9.70	3.88
50	8.20	0.85	8.80	0.0	8.80	1.76	8.80	3.52
60	7.36	0.68	7.80	0.0	7.80	1.56	7.80	3.12
70	6.28	0.46	6.30	0.0	6.30	1.26	6.30	2.92
80	4.90	0.30	4.50	0.0	4.50	0.90	4.50	1.80
90	3.27	0.14	2.50	0.0	2.50	0.50	2.50	1.00
95	2.28	0.08	1.40	0.0	1.40	0.28	1.40	0.56
100	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0

Per cent of chord.	Ref. No. 131.		Ref. No. 132.		Ref. No. 133.		Ref. No. 134.	
	N. P. L. 4c.		N. P. L. 4cα.		N. P. L. 4cβ		N. P. L. 4cγ.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.0	0.0	1.50	1.50	2.39	2.39	3.90	3.90
1.25	2.3	1.38	3.50	0.0	4.30	0.32	6.60	1.00
2.5	3.73	2.24	4.60	0.25	5.35	0.0	7.50	0.25
5	5.70	3.42	6.20	1.90	6.90	0.58	8.30	0.15
7.5	7.13	4.28	7.30	3.25	7.90	2.20	8.86	1.64
10	8.00	4.80	8.10	4.10	8.50	3.60	9.20	3.30
15	9.10	5.46	9.20	5.20	9.26	5.00	9.56	4.20
20	9.70	5.82	9.70	5.80	9.70	5.60	9.80	5.03
30	9.90	5.95	9.90	5.90	9.90	5.90	9.90	5.90
40	9.70	5.82	9.70	5.80	9.70	5.80	9.70	5.80
50	8.80	5.28	8.80	5.30	8.80	5.30	8.80	5.30
60	7.80	4.68	7.80	4.70	7.80	4.70	7.80	4.70
70	6.30	3.78	6.30	3.80	6.30	3.80	6.30	3.80
80	4.50	2.70	4.50	2.70	4.50	2.70	4.50	2.70
90	2.50	1.50	2.50	1.50	2.50	1.50	2.50	1.50
95	1.40	0.84	1.40	0.80	1.40	0.80	1.40	0.80
100	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0

Per cent of chord.	Ref. No. 135.		Ref. No. 136.		Ref. No. 137.		Ref. No. 138.	
	N. P. L. 73.		N. P. L. 214.		Portholme.		Scout E.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.50	0.50	0.0	0.0	1.00	1.00	1.10	1.10
1.25	2.80	0.0	2.20	0.31	2.15	+0.40	2.50	0.50
2.5	3.95	0.08	3.50	0.70	3.00	0.12	3.50	0.20
5	5.40	0.20	5.25	1.25	4.30	0.02	4.90	0.0
7.5	6.28	0.34	6.58	1.70	5.08	−0.04	5.82	0.02
10	6.90	0.40	7.50	2.09	5.80	−0.10	6.40	0.10
15	7.95	0.60	8.78	2.70	6.70	−0.20	7.10	0.40
20	8.50	0.80	9.51	3.25	7.20	−0.27	7.40	0.70
30	8.80	1.00	9.99	3.90	7.50	−0.35	7.50	1.10
40	8.70	1.00	9.84	3.60	7.34	−0.30	7.30	1.20
50	8.30	0.90	9.21	3.20	6.93	−0.25	7.00	1.10
60	7.30	0.70	7.92	2.55	6.30	−0.20	6.50	0.80
70	6.20	0.50	6.39	1.98	5.50	−0.12	5.60	0.40
80	5.00	0.30	4.73	1.30	4.30	−0.10	4.30	0.20
90	3.30	0.10	2.72	0.67	2.60	−0.05	2.90	0.0
95	2.10	0.05	1.65	0.32	1.65	−0.02	2.00	0.0
*100	0.40	0.40	0.50	0.0	0.20	+0.02	0.30	0.30
*97.5	....	....	....	....	....	0.0	....	....

Per cent of chord.	Ref. No. 139		Ref. No. 140.		Ref. No. 141.		Ref. No. 142.	
	Sopwith.		White.		C. & L.—A. 1.		C. & L.—A. 2.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.80	0.80	0.58	0.58	Rad.=1.04.		Rad.=1.04.	
1.25	2.40	0.20	2.33	0.02	2.15	0.0	2.167	0.0
*2.5	3.15	0.05	3.23	0.16	2.40	0.0	2.45	0.0
5	4.30	0.0	4.55	0.70	2.97	0.0	3.07	0.0
7.5	5.20	0.0	5.50	1.13	3.46	0.0	3.57	0.0
10	5.90	0.0	6.20	1.55	3.91	0.0	4.08	0.0
15	7.00	0.25	7.20	2.20	4.75	0.0	5.01	0.0
20	7.80	0.55	7.67	2.68	5.49	0.0	5.77	0.0
30	8.60	1.10	7.70	3.22	6.64	0.0	7.00	0.0
40	8.40	1.00	6.83	3.48	7.30	0.0	7.73	0.0
50	7.80	0.70	5.70	3.30	7.50	0.0	8.00	0.0
60	7.00	0.30	7.81	2.90	7.30	0.0	7.73	0.0
70	5.80	0.0	7.10	2.40	6.64	0.0	7.00	0.0
80	4.40	0.0	5.42	1.76	5.49	0.0	5.77	0.0
90	2.80	0.0	3.26	0.91	3.91	0.0	4.08	0.0
95	1.80	0.0	2.05	0.46	2.97	0.0	3.07	0.0
*100	0.60	0.60	0.24	0.24	Rad.=1.04.		Rad.=1.04.	
*1.5	....	....	2.59	0.0	....	....	....	....
*99.6	....	....	0.88	0.0	....	....	....	....

Per cent of chord.	Ref. No. 143.		Ref. No. 144.		Ref. No. 145.		Ref. No. 146	
	C. & L.—A. 3.		C. & L.—A. 4.		C. & L.—A. 5.		C. & L.—A. 6.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	Rad.=1.04.		Rad.=1.04.		Rad.=1.04.		Rad.=1.04.	
1.25	2.184	0.0	2.20	0.0	2.30	0.0	2.40	0.0
2.5	2.50	0.0	2.56	0.0	2.75	0.0	3.10	0.0
5	3.19	0.0	3.37	0.0	3.75	0.0	4.50	0.0
7.5	3.80	0.0	4.08	0.0	4.65	0.0	5.83	0.0
10	4.44	0.0	4.79	0.0	5.54	0.0	7.07	0.0
15	5.51	0.0	6.00	0.0	7.05	0.0	9.21	0.0
20	6.44	0.0	7.07	0.0	8.42	0.0	11.07	0.0
30	7.88	0.0	8.71	0.0	10.43	0.0	13.84	0.0
40	8.72	0.0	9.67	0.0	11.62	0.0	15.45	0.0
50	9.00	0.0	10.00	0.0	12.00	0.0	16.00	0.0
60	8.72	0.0	9.67	0.0	11.62	0.0	15.45	0.0
70	7.88	0.0	8.71	0.0	10.43	0.0	13.84	0.0
80	6.44	0.0	7.07	0.0	8.42	0.0	11.07	0.0
90	4.44	0.0	4.79	0.0	5.54	0.0	7.07	0.0
95	3.19	0.0	3.37	0.0	3.75	0.0	4.50	0.0
100	Rad.=1.04.		Rad.=1.04.		Rad.=1.04.		Rad.=1.04.	

Per cent of chord.	Ref. No. 147.		Ref. No. 148.		Ref. No. 149.		Ref. No. 150.	
	C. & L.—A. 7.		C. & L.—B. 1.		C. & L.—B. 2.		C. & L.—B. 3.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	Rad.=1.04.		Rad.=1.04.		Rad.=1.04.		Rad.=1.04.	
1.25	2.50	0.0	2.20	0.0	2.30	0.0	2.30	0.0
2.5	3.50	0.0	2.60	0.0	2.70	0.0	2.80	0.0
5	5.42	0.0	3.28	0.0	3.40	0.0	3.65	0.0
7.5	7.14	0.0	3.91	0.0	4.08	0.0	4.44	0.0
10	8.76	0.0	4.45	0.0	4.75	0.0	5.15	0.0
15	11.53	0.0	5.49	0.0	5.77	0.0	6.44	0.0
20	13.87	0.0	6.28	0.0	6.65	0.0	7.45	0.0
30	17.36	0.0	7.30	0.0	7.73	0.0	8.72	0.0
*40	19.34	0.0	....	....	....	....	....	....
50	20.00	0.0	7.30	0.0	7.73	0.0	8.72	0.0
60	19.34	0.0	6.85	0.0	7.15	0.0	8.10	0.0
70	17.36	0.0	6.00	0.0	6.30	0.0	7.05	0.0
80	13.87	0.0	4.90	0.0	5.15	0.0	5.70	0.0
90	8.76	0.0	3.55	0.0	3.70	0.0	3.95	0.0
95	5.42	0.0	2.75	0.0	2.85	0.0	2.95	0.0
100	Rad.=1.04.		Rad.=1.04.		Rad.=1.04.		Rad.=1.04.	
*37.5	....	....	7.50	0.0	8.00	0.0	9.00	0.0

Per cent of chord.	Ref. No. 151.		Ref. No. 152.		Ref. No. 153.		Ref. No. 154.	
	C. & L.—B. 4.		C. & L.—B. 5.		C. & L.—B. 6.		C. & L.—B. 7.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	Rad.=1.04.		Rad.=1.04.		Rad.=1.04.		Rad.=1.04.	
1.25	2.30	0.0	2.45	0.0	2.64	0.0	2.80	0.0
2.5	2.80	0.0	3.10	0.0	3.55	0.0	4.15	0.0
5	3.80	0.0	4.35	0.0	5.35	0.0	6.55	0.0
7.5	4.79	0.0	5.54	0.0	7.07	0.0	8.75	0.0
10	5.60	0.0	6.55	0.0	8.50	0.0	10.65	0.0
15	7.07	0.0	8.42	0.0	11.07	0.0	13.87	0.0
20	8.20	0.0	9.84	0.0	13.00	0.0	16.25	0.0
30	9.67	0.0	11.62	0.0	15.45	0.0	19.34	0.0
*40	....	....	....	....	....	....	....	....
50	9.67	0.0	11.62	0.0	15.45	0.0	19.34	0.0
60	8.90	0.0	10.70	0.0	14.20	0.0	17.88	0.0
70	7.75	0.0	9.30	0.0	12.20	0.0	15.45	0.0
80	6.20	0.0	7.30	0.0	9.60	0.0	11.95	0.0
90	4.25	0.0	4.85	0.0	6.00	0.0	7.50	0.0
95	3.10	0.0	3.40	0.0	3.95	0.0	4.72	0.0
100	Rad.=1.04.		Rad.=1.04.		Rad.=1.04.		Rad.=1.04.	
*37.5	10.00	0.0	12.00	0.0	16.00	0.0	20.00	0.0

Per cent of chord.	Ref. No. 155.		Ref. No. 156.		Ref. No. 157.		Ref. No. 158.	
	Eiffel 8.		Eiffel 9.		Eiffel 10.		Eiffel 11.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.0	0.0	3.60	3.60	2.73	0.0		0.0
1.25	0.66	0.12	6.35	1.0	2.96	0.32		0.20
2.5	1.24	0.32	7.28	0.22	3.23	0.60		0.40
*5	2.40	0.70	8.60	0.21	3.75	1.17		0.80
7.5	3.55	1.10	9.60	1.73	4.23	1.70		1.20
10	4.55	1.40	10.45	3.25	4.68	2.23		1.60
15	6.62	2.18	11.80	5.75	5.50	3.15		2.30
20	8.45	3.00	12.70	7.35	6.14	3.90		3.00
30	10.20	3.80	13.70	8.45	7.01	4.86	1.33 thick.	3.96
40	10.65	4.05	13.75	8.33	7.18	5.27		4.12
50	10.20	3.85	12.83	7.70	6.93	5.18		3.64
60	9.18	3.50	11.25	6.70	6.23	4.56		2.93
70	7.45	2.80	9.10	5.45	5.10	3.70		2.19
80	5.26	1.95	6.64	4.00	3.70	2.55		1.46
90	2.75	1.00	3.65	2.06	2.20	1.27		0.76
95	1.43	0.45	2.12	1.30	1.43	0.65		0.35
100	0.0	0.0	0.50	0.0	0.64	0.0		0.0
*3.75	....	....	....	0.0	....	....		....

Per cent of chord.	Ref. No. 159.		Ref. No. 160.		Ref. No. 161.		Ref. No. 162.	
	Eiffel 12.		Eiffel 13.		Eiffel 13 bis.		Eiffel 14.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.53	0.53	0.80	.80	0.70	0.70	1.25	1.25
1.25	1.48	0.10	2.88	0.12	2.06	0.16	2.75	0.0
2.5	1.60	0.16	4.20	0.88	2.85	0.45	3.28	0.15
5	1.84	0.30	6.10	2.10	4.15	1.50	4.20	0.60
7.5	2.08	0.43	7.40	3.60	5.10	1.58	4.95	1.05
10	2.30	0.53	8.30	3.80	5.83	2.05	5.60	1.40
15	2.68	0.65	9.30	4.90	6.90	2.80	6.60	2.10
20	2.85	0.74	9.70	5.65	7.58	3.35	7.25	2.73
30	3.10	0.85	9.75	6.10	8.00	3.65	7.78	3.80
40	3.05	0.84	9.40	5.93	7.40	3.22	7.55	4.27
50	2.95	0.75	8.60	5.45	6.45	2.73	6.95	4.30
60	2.80	0.60	7.43	4.60	5.40	2.13	6.05	3.90
70	2.50	0.48	5.95	3.50	4.28	1.58	4.90	3.30
80	1.96	0.30	4.25	2.33	3.10	1.04	3.50	2.40
90	1.05	0.14	2.48	1.16	1.95	0.45	2.00	1.25
95	0.68	0.06	1.56	0.58	1.31	0.23	1.24	0.60
100	0.30	0.0	0.55	0.0	0.65	0.0	0.45	0.0

Per cent of chord.	Ref. No. 163.		Ref. No. 164.		Ref. No. 165.		Ref. No. 166.	
	Eiffel 15.		Eiffel 16.		Eiffel 16a.		Eiffel 16b.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower	Upper.	Lower.
0	1.90	1.90	0.0	0.0	0.0	0.0	0.0	0.0
1.25	3.37	0.32	1.00	0.0	1.74	0.0	3.10	0.0
2.5	3.60	0.06	1.85	0.0	2.86	0.0	5.05	0.0
5	3.86	0.15	3.17	0.0	4.80	0.0	7.87	0.0
7.5	3.93	0.26	4.24	0.0	6.36	0.0	9.80	0.0
10	3.95	0.37	5.15	0.0	7.70	0.0	11.22	0.0
15	3.82	0.43	6.30	0 0	9.38	0.0	12.83	0.0
20	3.60	0.34	6.66	0.0	9.99	0.0	13.32	0.0
30	3.13	0.0	6.25	0.0	9.63	0.0	13.05	0.0
40	2.92	0.16	5.33	0.0	8.66	0.0	11.99	0.0
50	2.90	0.56	4.63	0.0	7.33	0.0	10.30	0.0
60	2.95	1.15	3.79	0.0	5.99	0.0	8.46	0.0
70	3.20	1.83	2.90	0.0	4.57	0.0	6.42	0.0
80	3.75	2.70	1.99	0.0	3.13	0.0	4.39	0.0
90	4.50	3.80	1.10	0.0	1.76	0.0	2.42	0.0
95	4.94	4.42	0.65	0.0	1.05	0.0	1.43	0.0
100	5.40	5.10	0.0	0.0	0.0	0.0	0.0	0.0

Per cent of chord.	Ref. No. 167.		Ref. No. 168.		Ref. No. 169.		Ref. No. 170.	
	Eiffel 16c.		Eiffel 16d.		Eiffel 17.		Eiffel 18.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.0	0.0	0.0	0.0	0.0	0.0	....	....
1.25	4.75	0.0	7.25	0.0	0.83	0.0	....	....
2.5	7.20	0.0	10.15	0.0	1.25	0.0	....	....
5	10.57	0.0	13.65	0.0	2.06	0.0	....	....
7.5	12.80	0.0	15.93	0.0	2.80	0.0	....	....
10	14.38	0.0	17.56	0.0	3.53	0.0	....	....
15	16.10	0.0	19.42	0.0	4.75	0.0	....	....
20	16.65	0.0	19.99	0.0	5.66	0.0	6.65	0.0
30	16.05	0.0	19.37	0.0	6.54	0.0	6.50	0.0
40	14.66	0.0	17.99	0.0	6.54	0.0	6.10	0.0
50	12.86	0.0	15.83	0.0	6.00	0.0	5.35	0.0
60	10.66	0.0	12.99	0.0	5.30	0.0	4.35	0.0
70	8.00	0.0	9.83	0.0	4.18	0.0	3.35	0.0
80	5.33	0.0	6.66	0.0	3.00	0.0	2.35	0.0
90	2.84	0.0	3.50	0.0	1.75	0.0	1.40	0.0
95	1.56	0.0	1.90	0.0	1.10	0.0	0.90	0.0
100	0.0	0.0	0.0	0.0	0.0	0.0	0.40	0.0

Per cent of chord.	Ref. No. 171.		Ref. No. 172.		Ref. No. 173.		Ref. No 174.	
	Eiffel 30.		Eiffel 31.		Eiffel 32.		Eiffel 33.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.20	0.0	0.0	0.0	1.00	1.00	1.10	0.0
1.25	0.73	0.17	0.68	0.26	2.00	0.37	2.80	0.0
2.5	1.33	0.40	1.33	0.55	2.67	0.18	3.63	0.10
5	2.45	0.82	2.53	1.07	3.75	0.0	5.00	0.43
7.5	3.52	1.20	3.65	1.56	4.65	0.24	6.07	0.77
10	4.60	1.60	4.75	2.00	5.35	0.64	6.93	1.12
15	6.50	2.36	6.85	2.84	6.56	1.50	8.20	1.80
20	8.16	3.06	8.67	3.50	7.36	2.24	8.88	2.36
30	10.50	4.08	10.90	4.55	7.92	2.93	9.27	2.98
40	11.12	4.40	11.40	4.75	7.26	2.70	9.00	3.35
50	10.60	4.43	11.20	4.66	5.93	1.96	8.15	3.30
60	9.28	4.08	9.86	4.15	4.50	1.13	7.10	2.94
70	7.40	3.37	8.10	3.34	3.33	0.54	5.76	2.39
80	5.30	2.44	5.80	2.35	2.36	0.26	4.16	1.68
90	2.84	1.24	3.10	1.20	1.50	0.0	2.33	0.93
95	1.58	0.63	1.72	0.60	1.20	0.10	1.38	0.45
100	0.20	0.0	0.0	0.0	0.70	0.70	0.43	0.0

Per cent of chord.	Ref. No. 175.		Ref. No. 176.		Ref. No. 177.		Ref. No. 178.	
	Eiffel 34.		Eiffel 35.		Eiffel 36.		Eiffel 37.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.50	0.50	0.90	0.90	0.70	0.70	3.00	3.00
1.25	1.65	0.10	2.28	0.08	1.90	0.23	3.60	1.80
2.5	2.30	0.41	2.66	0.26	2.84	0.05	4.06	1.00
*5	3.62	1.06	3.33	0.95	4.30	0.04	4.88	0.18
7.5	4.90	1.67	4.03	1.56	5.28	0.15	5.60	0.02
10	6.13	2.20	4.66	2.00	6.05	0.35	6.28	0.30
15	8.25	3.45	5.76	2.76	7.05	0.76	7.25	1.06
20	10.00	4.45	6.66	3.33	7.80	1.15	7.95	2.00
30	11.90	5.40	7.70	4.32	8.66	1.78	8.50	3.30
40	12.40	5.70	8.00	4.74	8.78	2.10	8.24	3.96
50	11.50	5.20	7.95	5.00	8.40	2.30	7.60	4.10
60	9.82	4.20	7.34	4.74	7.46	2.15	6.76	3.95
70	7.95	3.30	6.22	4.33	5.92	1.86	5.55	3.50
80	5.85	2.30	4.24	3.00	4.23	1.33	4.00	2.65
90	3.56	1.20	2.16	1.40	2.34	0.70	2.33	1.50
95	2.32	0.63	1.16	0.67	1.40	0.40	1.40	0.83
100	1.00	0.0	0.10	0.10	0.40	0.0	0.0	0.0
*4.4	....	....	....	....	....	0.0	....	....

Per cent of chord.	Ref. No. 179.		Ref. No. 180.		Ref. No. 181.		Ref. No. 182.	
	Eiffel 38.		Eiffel 39.		Eiffel 40.		Eiffel 41.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.50	0.50	0.40	0.0	0.35	0.0	0.10	0.0
1.25	1.65	0.0	0.94	0.0	0.90	0.03	1.10	0.15
2.5	2.46	0.0	1.75	0.0	1.67	0.23	3.00	0.30
5	3.76	0.20	4.50	0.0	3.50	0.63	4.00	0.70
7.5	4.75	0.60	9.55	0.0	4.25	1.06	4.65	1.10
10	5.62	1.22	11.22	0.0	4.95	1.45	5.20	1.45
15	6.88	2.43	12.83	0.0	6.10	2.13	6.40	2.06
20	7.74	3.22	13.32	0.0	6.95	2.62	6.75	2.60
30	8.70	4.32	13.05	0.0	7.63	3.05	7.65	3.65
40	8.70	4.52	11.99	0.0	7.70	3.18	7.68	3.92
50	8.32	4.13	10.30	0.0	7.50	3.05	7.35	4.00
60	7.55	3.35	8.46	0.0	6.75	2.76	6.54	3.64
70	6.13	2.60	6.42	0.0	5.62	2.16	5.46	3.00
80	4.65	1.80	4.39	0.0	4.23	1.42	4.16	2.10
90	2.85	0.94	2.42	0.0	2.54	0.70	2.35	1.06
95	1.85	0.50	1.43	0.0	1.68	0.30	1.34	0.54
100	1.25	0.0	0.0	0.0	0.70	0.0	0.50	0.0

Per cent of chord.	Ref. No. 183.		Ref. No. 184.		Ref. No. 185.		Ref. No. 186.	
	Eiffel 42.		Eiffel 43.		Eiffel 44.		Eiffel 45.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.13	0.0	0.13	0.0	1.05	1.05	1.38	1.38
1.25	1.08	0.25	1.08	0.25	2.10	0.0	3.74	0.02
2.5	2.80	0.50	2.80	0.50	2.20	0.05	4.72	0.0
5	6.15	0.93	6.15	0.93	2.45	0.23	5.62	0.03
7.5	7.54	1.33	7.54	1.33	2.73	0.45	6.45	0.48
10	8.68	1.68	8.68	1.68	3.13	0.78	7.20	1.32
15	10.24	2.30	10.24	2.30	4.23	1.77	7.95	3.27
20	10.90	2.90	10.90	2.90	6.30	2.93	8.30	4.10
30	10.90	3.64	10.90	3.64	9.08	4.00	8.55	4.37
40	10.47	3.84	10.47	3.84	8.70	3.90	8.10	4.06
50	9.55	3.90	9.55	3.90	7.80	3.35	7.33	3.75
60	8.30	3.63	8.30	3.63	6.16	2.45	6.25	3.18
70	6.75	3.14	6.75	3.14	4.22	1.45	5.10	2.46
80	4.86	2.25	4.86	2.25	2.45	0.67	3.80	1.68
90	2.86	1.20	2.86	1.20	1.14	0.14	2.26	0.83
95	1.73	0.55	1.73	0.55	0.63	0.02	1.44	0.40
100	0.53	0.0	0.53	0.0	0.30	0.0	0.60	0.0

Per cent of chord.	Ref. No. 187.		Ref. No. 188.		Ref. No. 189.		Ref. No. 190.	
	Eiffel 46.		Eiffel 47.		Eiffel 48.		Eiffel 49.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	1.25	1.25	0.75	0.75	0.60	0.60	0.40	0.40
1.25	2.30	0.08	2.35	0.06	2.55	0.14	2.12	0.0
2.5	3.24	0.0	3.30	0.30	3.78	0.38	2.86	0.13
5	5.07	0.32	4.93	0.78	5.40	0.80	4.00	0.56
7.5	6.30	1.05	6.23	1.20	6.30	1.17	4.78	0.95
10	6.96	1.90	7.12	1.60	6.65	1.50	5.38	1.30
15	9.30	3.40	8.16	2.26	6.42	2.03	6.10	1.90
20	10.60	4.70	8.68	2.75	6.08	2.40	6.26	2.27
30	10.93	6.10	8.30	3.38	5.25	2.85	5.58	2.75
*40	10.40	6.40	7.45	3.85	7.82	2.95	6.64	2.95
50	9.24	6.00	6.03	4.00	7.40	2.80	6.53	2.94
60	7.65	4.8	8.93	3.75	5.56	2.45	5.53	2.80
•*70	5.75	3.32	7.95	3.15	5.85	2.00	5.40	2.40
80	3.65	1.76	5.90	2.20	6.53	1.43	5.55	1.66
90	1.70	0.60	3.70	1.16	4.10	0.78	3.75	0.85
95	1.00	0.16	2.45	0.56	2.72	0.40	2.37	0.40
100	0.50	0.0	1.15	0.0	1.30	0.0	0.85	0.0
*33.9	....	....	....	....	....	....	4.92	2.95
•*66.82	....	....	....	....	....	....	4.28	2.31

Per cent of chord.	Ref. No. 191.		Ref. No. 192.		Ref. No. 193.		Ref. No. 194.	
	Eiffel 52.		Eiffel 53.		Eiffel 54.		Eiffel 55.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.70	0.70	0.30	0.30	0.60	0.60	0.67	0.67
1.25	2.00	0.30	1.82	0.14	1.70	0.11	1.32	0.20
2.5	3.00	0.20	2.82	0.30	2.50	0.0	1.56	0.13
5	4.40	0.05	4.33	0.66	3.75	0.05	2.00	0.03
7.5	5.43	0.07	5.36	1.00	4.70	0.15	2.36	0.0
10	6.20	0.30	6.14	1.35	5.40	0.30	2.70	0.0
15	7.33	1.00	7.15	1.83	6.35	0.62	3.30	0.02
20	8.10	1.85	7.85	2.10	6.70	0.80	3.75	0.06
30	9.13	3.00	8.70	2.70	6.50	1.00	4.44	0.28
40	9.54	3.50	8.86	2.80	6.00	1.17	4.92	0.40
50	9.40	3.54	8.55	2.60	5.25	1.15	5.10	0.22
60	8.60	2.65	7.40	2.40	4.50	1.05	4.95	0.08
70	6.96	1.57	5.76	1.30	3.60	0.54	4.38	0.03
80	4.53	0.58	3.97	0.60	2.65	0.06	3.55	0.01
90	2.00	0.03	2.10	0.13	1.90	0.21	2.45	0.0
95	1.26	0.0	1.35	0.04	1.71	0.56	1.80	0.03
100	0.65	0.0	0.50	0.0	1.65	1.00	0.70	0.70

Per cent of chord.	Ref. No. 195.		Ref. No. 196.		Ref. No. 197.		Ref. No. 198.	
	Eiffel 56.		Eiffel 57.		Eiffel 58.		Eiffel 59.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.0	0.0	8.00	8.00	4.10	4.10	0.60	0.60
1.25	0.70	−0.64	12.70	3.50	7.06	1.14	1.95	0.15
2.5	0.90	−0.66	14.72	2.00	7.90	0.35	2.83	0.45
5	1.30	−0.70	17.55	0.50	8.90	0.0	4.46	1.03
7.5	1.62	−0.77	19.30	0.0	9.56	0.0	5.90	1.53
10	1.90	−0.85	20.60	0.0	9.90	0.0	7.13	2.00
*15	2.36	−1.00	22.40	0.0	9.92	0.0	9.37	2.84
20	2.66	−1.13	23.12	0.0	9.50	0.0	10.87	3.50
30	2.85	−1.55	22.00	0.0	8.50	0.0	13.30	4.55
40	2.86	−1.92	19.74	0.0	7.50	0.0	14.20	4.75
50	2.85	−2.24	16.94	0.0	6.43	0.0	13.66	4.66
60	2.76	−2.35	13.90	0.0	5.27	0.0	12.13	4.15
70	2.50	−2.40	10.60	0.0	4.05	0.0	9.75	3.34
80	1.94	−2.20	7.20	0.0	2.85	0.0	7.07	2.35
90	1.32	−1.70	3.78	0.0	1.62	0.0	3.72	1.20
95	0.90	−1.20	2.05	0.0	1.00	0.0	2.05	0.60
100	0.0	0.0	0.35	0.0	0.20	0.20	0.30	0.0
*12.5	....	....	....	....	10.00	0.0	....	....

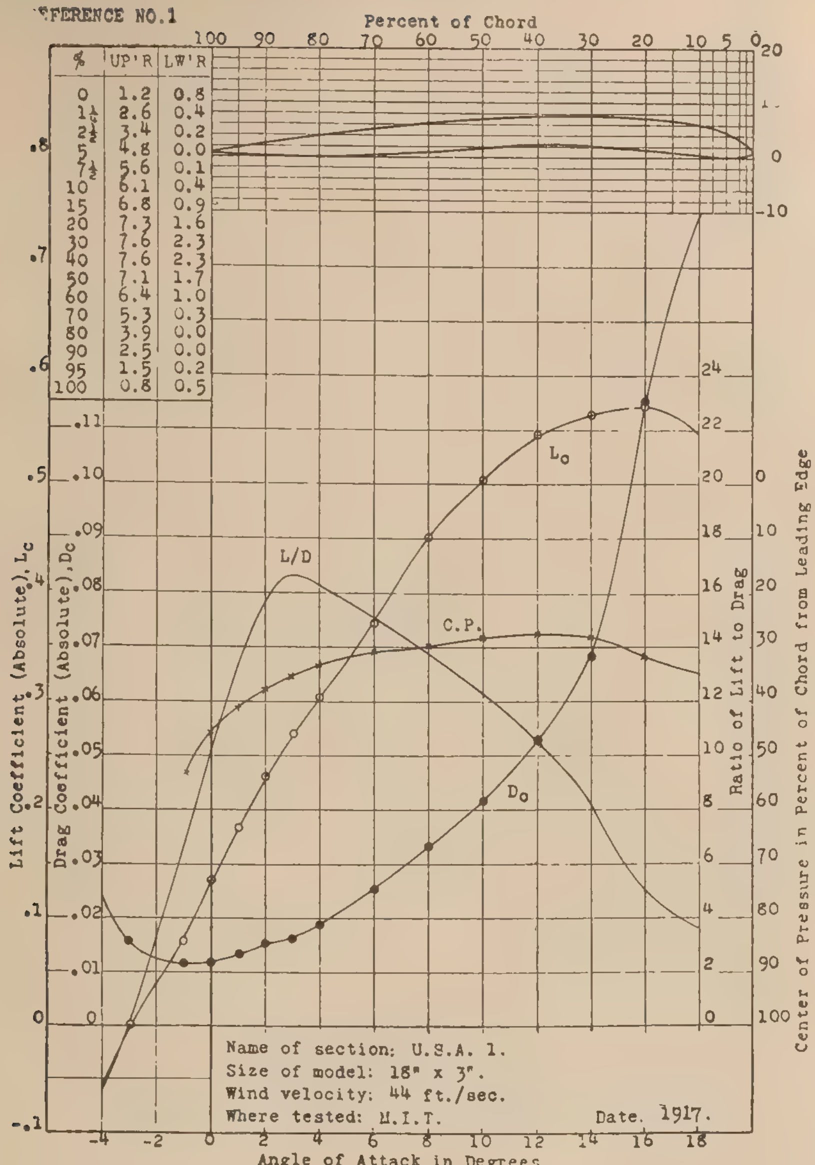
Per cent of chord.	Ref. No. 199.		Ref. No. 200.		Ref. No. 201.		Ref. No. 202.	
	Eiffel 60.		Eiffel 61.		Eiffel 62.		Dorand.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.60	0.60	0.60	0.60	0.60	0.60	1.00	1.00
1.25	2.22	0.15	2.30	0.15	2.40	0.15	2.25	0.03
2.5	3.37	0.45	3.73	0.45	3.94	0.45	2.70	0.23
5	5.50	1.03	6.18	1.03	6.80	1.03	3.50	0.63
7.5	7.28	1.53	8.33	1.53	9.34	1.53	4.25	1.06
10	8.87	2.00	10.26	2.00	11.70	2.00	4.95	1.45
15	11.57	2.84	13.63	2.84	15.56	2.84	6.10	2.13
20	13.73	3.50	16.40	3.50	18.60	3.50	6.95	2.62
30	16.55	4.55	19.87	4.55	23.00	4.55	7.63	3.05
40	17.73	4.75	21.06	4.75	24.53	4.75	7.70	3.18
50	16.84	4.66	19.85	4.66	23.00	4.66	7.50	3.05
60	15.00	4.15	17.33	4.15	19.90	4.15	6.75	2.76
70	12.15	3.34	13.65	3.34	16.00	3.34	5.62	2.16
80	8.73	2.35	9.67	2.35	11.33	2.35	4.23	1.42
90	4.65	1.20	5.15	1.20	6.03	1.20	2.54	0.70
95	2.50	0.60	2.75	0.60	3.20	0.60	1.68	0.30
100	0.35	0.0	0.40	0.0	0.40	0.0	0.70	0.0

Per cent of chord.	Ref. No. 203.		Ref. No. 204.		Ref. No. 205.		Ref. No. 206.	
	Halbronn 2.		Halbronn 3.		S. E. A.		St. Cyr. 1.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	2.73	2.73	0.75	0.75	0.90	0.90	0.25	0.25
1.25	3.72	1.35	1.70	0.25	2.00	0.50	0.70	0.0
2.5	4.52	0.66	2.36	0.0	2.80	0.33	0.96	0.0
5	5.66	0.0	3.39	0.0	4.05	0.08	1.45	0.0
7.5	6.55	0.0	4.30	0.08	4.90	0.0	1.88	0.0
10	7.27	0.0	4.99	0.18	5.55	0.10	2.26	0.0
15	8.00	0.10	5.90	0.40	6.45	0.35	2.76	0.0
20	8.30	0.37	6.41	0.56	6.95	0.65	3.00	0.0
30	8.48	0.66	6.78	0.94	7.30	1.00	3.10	0.0
40	8.12	0.75	6.60	1.22	7.20	0.95	3.10	0.0
50	7.36	0.66	6.02	0.94	6.90	0.65	3.10	0.0
60	6.42	0.48	5.18	0.66	6.27	0.27	3.10	0.0
70	5.28	0.23	4.20	0.37	5.42	0.12	3.10	0.0
80	4.20	0.09	3.15	0.19	4.10	0.08	3.00	0.0
90	2.64	0.0	1.85	0.0	2.25	0.03	2.11	0.0
95	1.63	0.15	1.15	0.05	1.20	0.0	1.38	0.0
100	0.47	0.47	0.47	0.19	0.0	0.0	0.25	0.25

Per cent of chord.	Ref. No. 207.		Ref. No. 208.		Ref. No. 209.		Ref. No. 210.	
	St. Cyr. 2.		St. Cyr. 3.		Turin 1.		Turin 2.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	1.10	1.10	2.30	2.30	0.65	0.65	0.60	0.60
1.25	2.00	0.25	3.65	1.25	1.85	0.0	1.80	0.05
2.5	2.45	0.0	4.60	0.60	2.52	0.10	2.55	0.15
5	3.25	0.50	5.85	0.0	3.60	0.30	3.80	0.55
7.5	3.85	1.10	6.60	0.40	4.42	0.60	4.65	0.95
10	4.25	1.70	7.20	1.00	5.00	0.90	5.25	1.30
15	5.05	2.50	8.00	2.10	5.55	1.35	5.80	1.75
20	5.65	3.00	8.50	3.10	5.75	1.60	6.07	2.00
30	6.40	3.50	9.35	4.60	5.80	1.70	6.20	2.10
40	6.25	3.25	9.85	5.25	5.70	1.60	5.90	1.75
50	5.75	2.85	9.80	5.20	5.15	1.35	5.50	1.40
60	4.95	2.40	9.40	4.45	4.28	0.80	4.82	1.10
70	4.00	1.90	8.20	3.30	3.30	0.40	3.95	0.70
80	2.90	1.25	5.85	1.95	2.35	0.0	3.15	0.50
90	1.70	0.65	2.90	0.40	1.40	0.0	1.90	0.20
95	1.10	0.30	1.85	0.0	1.00	0.0	1.29	0.05
100	0.30	0.30	1.00	1.00	0.40	0.40	0.40	0.40

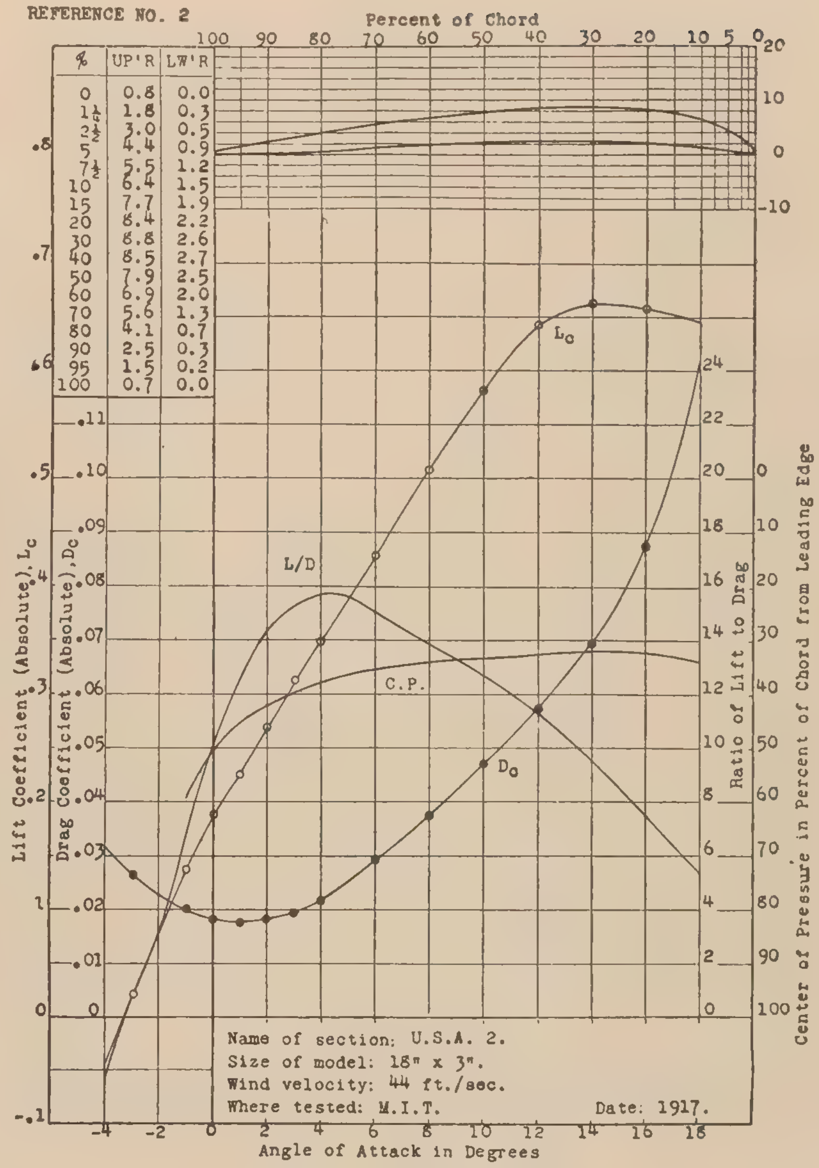
Per cent of chord.	Ref. No. 211.		Ref. No. 212.		Ref. No. 213.		Ref. No. 214.	
	Bleriot 3-pl.		Italian 1.		Italian 2.		Italian 3.	
	Ordinates.							
	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.	Upper.	Lower.
0	0.0	0.0	0.58	0.58	0.80	0.80	0.65	0.65
1.25	2.40	0.0	1.40	0.10	1.60	0.30	2.10	0.15
2.5	3.20	0.0	2.10	0.0	2.22	0.05	3.05	0.0
5	4.30	0.0	3.12	0.05	3.35	0.08	4.36	0.18
7.5	4.80	0.0	3.90	0.20	4.20	0.20	5.20	0.70
10	5.30	0.0	4.58	0.48	4.90	0.45	5.87	1.38
15	5.60	0.0	5.48	1.12	6.05	1.10	6.60	2.10
20	5.70	0.0	6.00	1.83	6.67	1.65	7.00	2.40
30	6.00	0.0	6.35	2.42	7.40	2.70	7.15	2.50
40	5.80	0.0	6.14	2.48	7.50	3.00	6.95	2.37
50	5.20	0.0	5.70	2.00	7.18	2.80	6.38	2.10
60	4.60	0.0	5.05	1.47	6.70	2.50	5.58	1.68
70	3.70	0.0	3.95	0.93	5.40	1.75	4.72	1.17
80	2.60	0.0	2.55	0.60	3.80	0.73	3.70	0.62
90	1.60	0.0	1.37	0.28	2.00	0.35	2.18	0.30
95	1.05	0.0	0.75	0.20	1.30	0.38	1.27	0.0
100	0.45	0.0	0.25	0.25	0.38	0.38	0.0	0.0

REFERENCE NO. 1



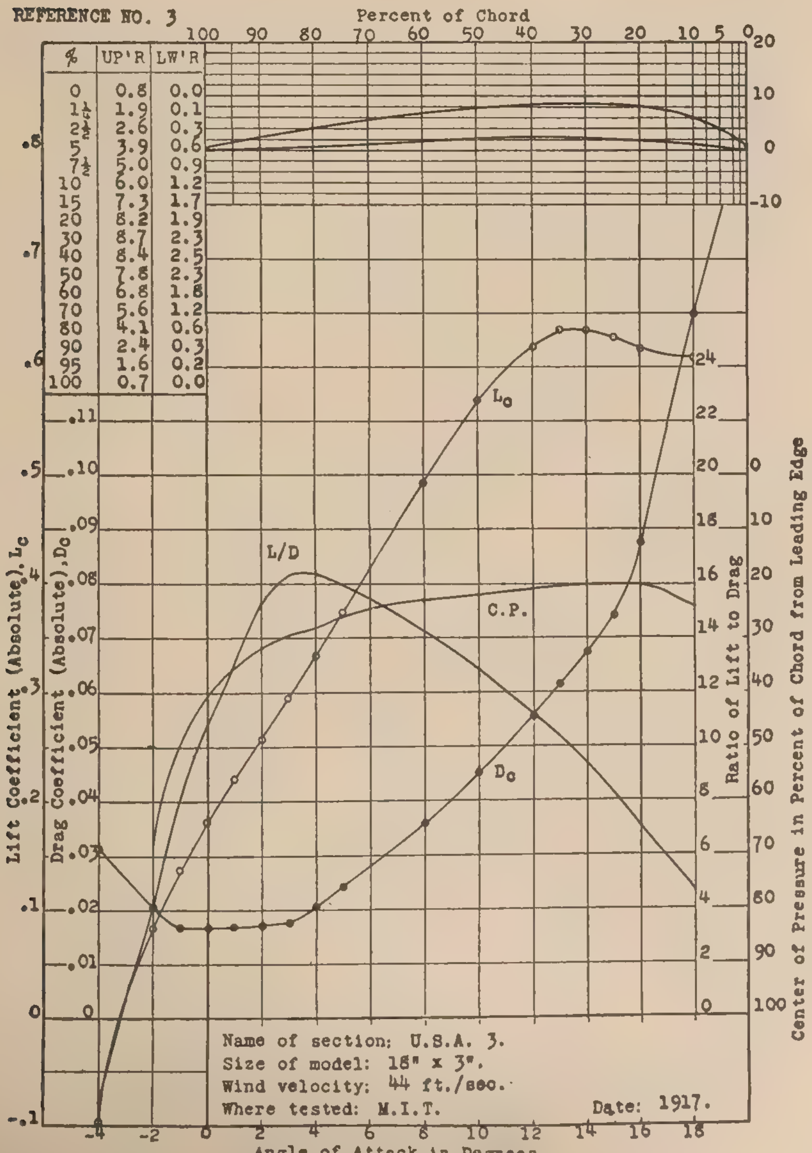
%	UP'R	LW'R
0	1.2	0.8
1¼	2.6	0.4
2½	3.4	0.2
5	4.8	0.0
7½	5.6	0.1
10	6.1	0.4
15	6.8	0.9
20	7.3	1.6
30	7.6	2.3
40	7.6	2.3
50	7.1	1.7
60	6.4	1.0
70	5.3	0.3
80	3.9	0.0
90	2.5	0.0
95	1.5	0.2
100	0.8	0.5

REFERENCE NO. 2



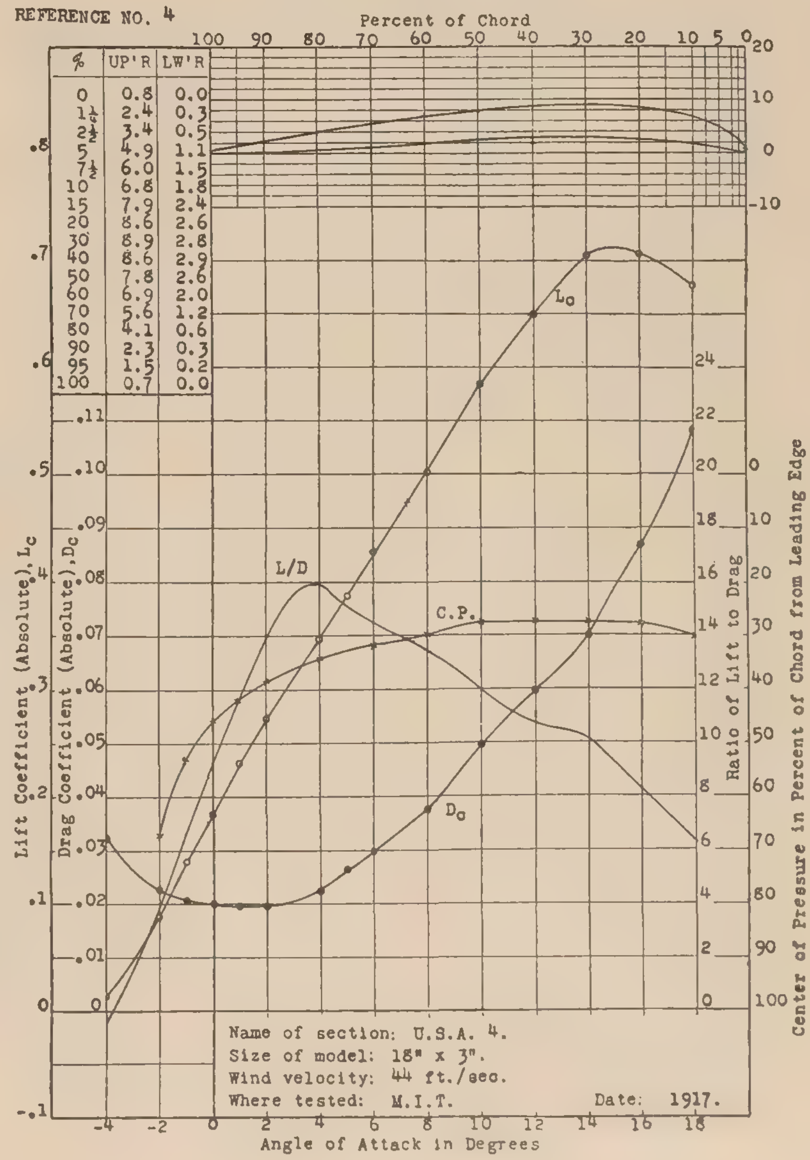
%	UP'R	LW'R
0	0.8	0.0
1¼	1.8	0.3
2½	3.0	0.5
5	4.4	0.9
7½	5.5	1.2
10	6.4	1.5
15	7.7	1.9
20	8.4	2.2
30	8.8	2.6
40	8.5	2.7
50	7.9	2.5
60	6.9	2.0
70	5.6	1.3
80	4.1	0.7
90	2.5	0.3
95	1.5	0.2
100	0.7	0.0

REFERENCE NO. 3



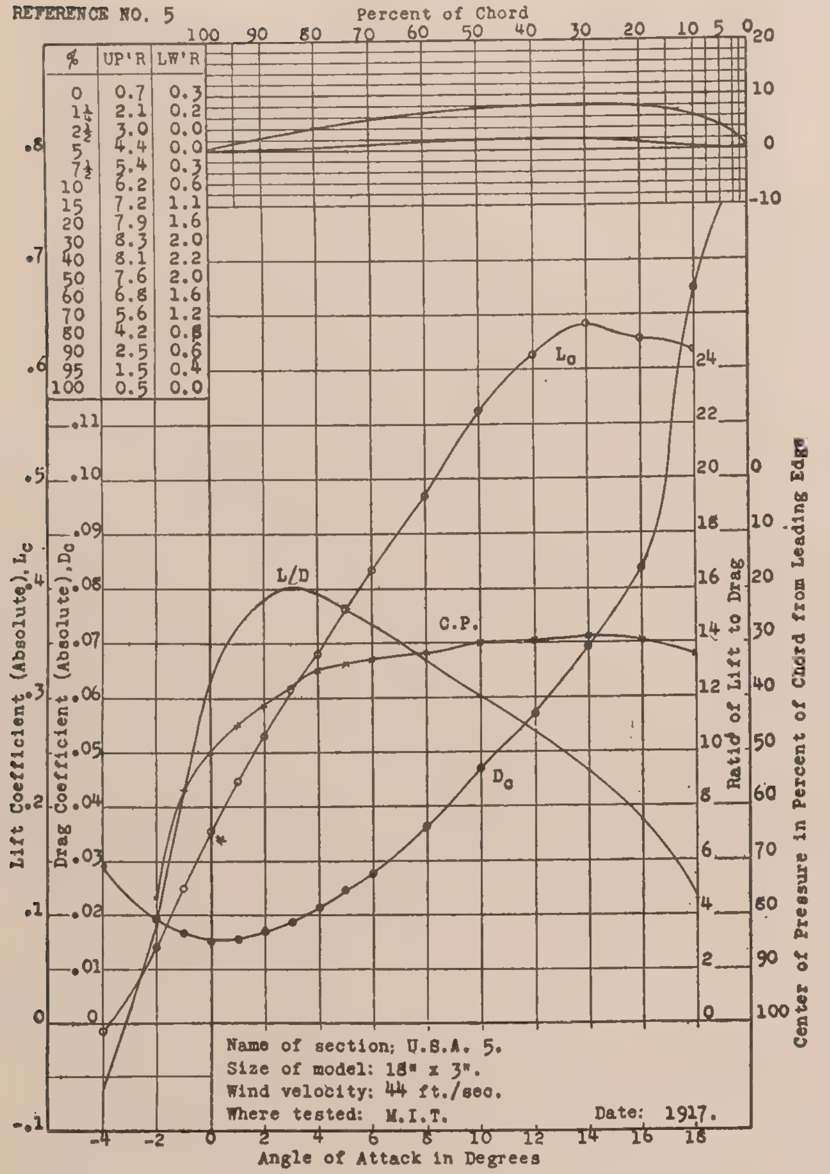
%	UP'R	LW'R
0	0.8	0.0
1¼	1.9	0.1
2½	2.6	0.3
5	3.9	0.6
7½	5.0	0.9
10	6.0	1.2
15	7.3	1.7
20	8.2	1.9
30	8.7	2.3
40	8.4	2.5
50	7.8	2.3
60	6.8	1.8
70	5.6	1.2
80	4.1	0.6
90	2.4	0.3
95	1.6	0.2
100	0.7	0.0

REFERENCE NO. 4



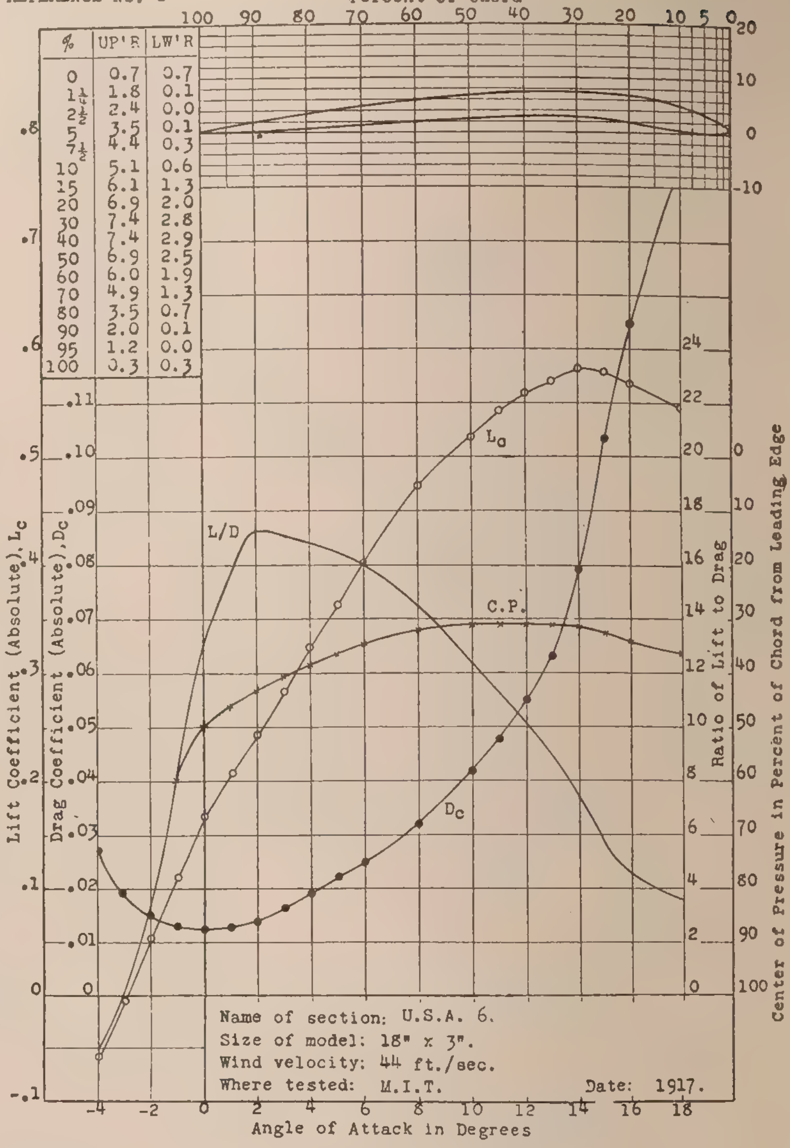
%	UP'R	LW'R
0	0.8	0.0
1¼	2.4	0.3
2½	3.4	0.5
5	4.9	1.1
7½	6.0	1.5
10	6.8	1.8
15	7.9	2.4
20	8.6	2.6
30	8.9	2.8
40	8.6	2.9
50	7.8	2.6
60	6.9	2.0
70	5.6	1.2
80	4.1	0.6
90	2.3	0.3
95	1.5	0.2
100	0.7	0.0

REFERENCE NO. 5



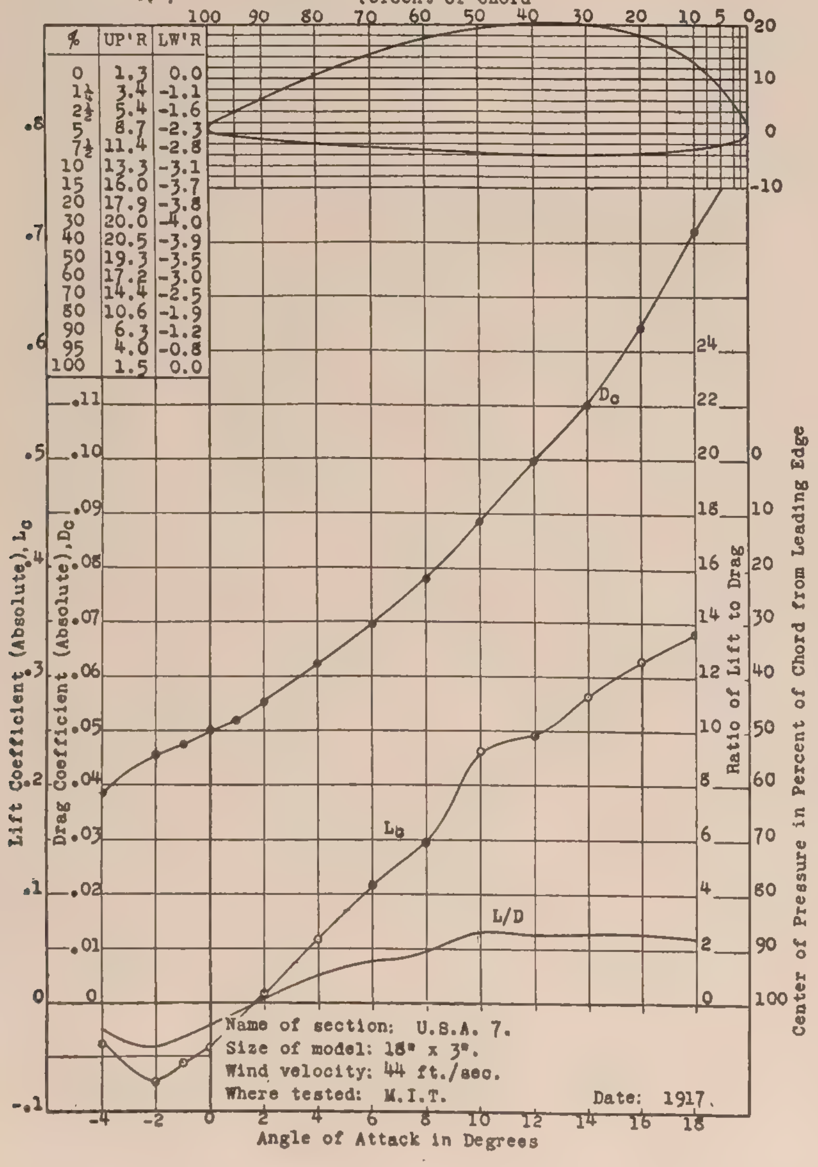
%	UP'R	LW'R
0	0.7	0.3
1¼	2.1	0.2
2½	3.0	0.0
5	4.4	0.0
7½	5.4	0.3
10	6.2	0.6
15	7.2	1.1
20	7.9	1.6
30	8.3	2.0
40	8.1	2.2
50	7.6	2.0
60	6.8	1.6
70	5.6	1.2
80	4.2	0.8
90	2.5	0.6
95	1.5	0.4
100	0.5	0.0

REFERENCE NO. 6



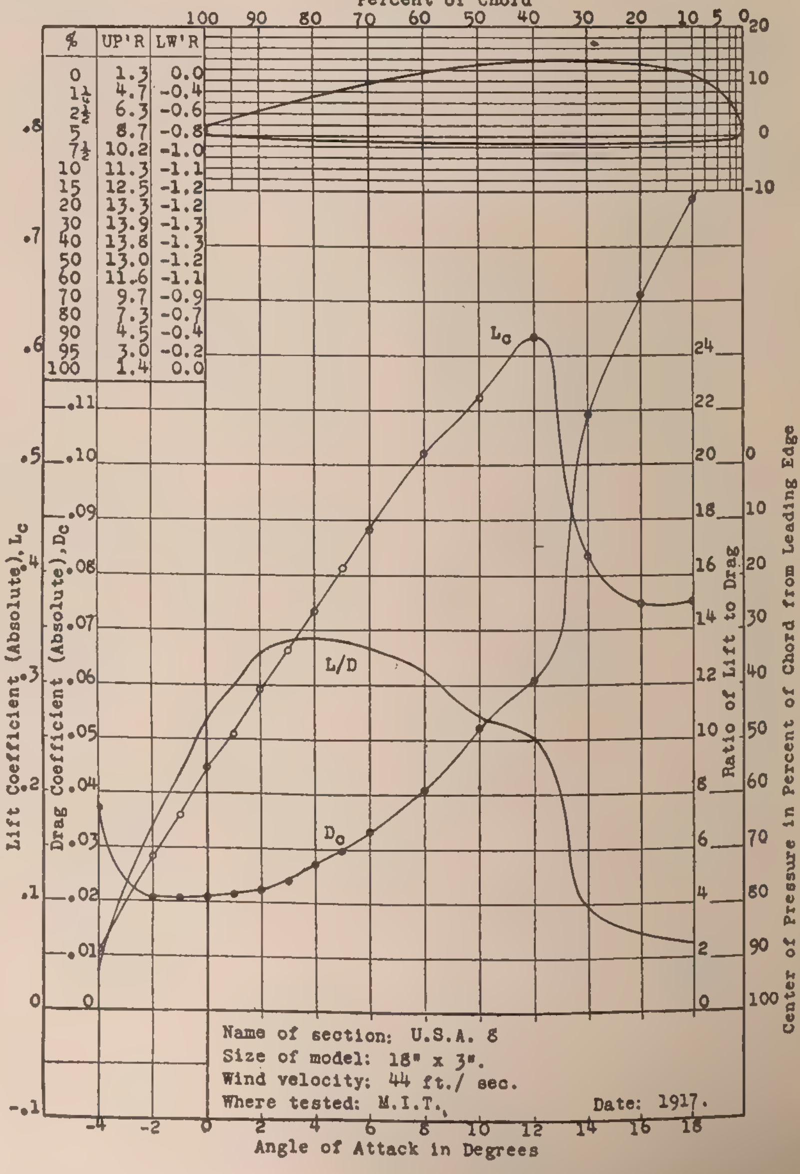
%	UP'R	LW'R
0	0.7	0.7
1¼	1.8	0.1
2½	2.4	0.0
5	3.5	0.1
7½	4.4	0.3
10	5.1	0.6
15	6.1	1.3
20	6.9	2.0
30	7.4	2.8
40	7.4	2.9
50	6.9	2.5
60	6.0	1.9
70	4.9	1.3
80	3.5	0.7
90	2.0	0.1
95	1.2	0.0
100	0.3	0.3

REFERENCE NO. 7



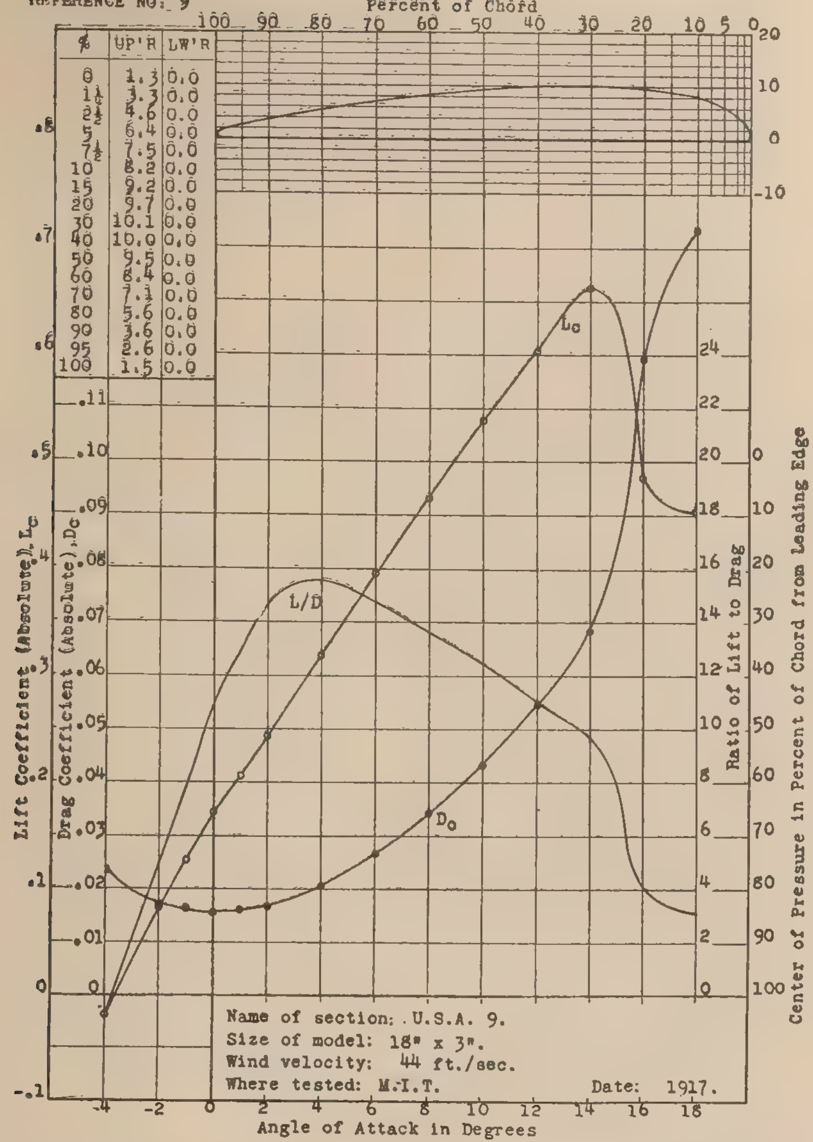
%	UP'R	LW'R
0	1.3	0.0
1¼	3.4	-1.1
2½	5.4	-1.6
5	8.7	-2.3
7½	11.4	-2.8
10	13.3	-3.1
15	16.0	-3.7
20	17.9	-3.8
30	20.0	-4.0
40	20.5	-3.9
50	19.3	-3.5
60	17.2	-3.0
70	14.4	-2.5
80	10.6	-1.9
90	6.3	-1.2
95	4.0	-0.8
100	1.5	0.0

REFERENCE NO. 8

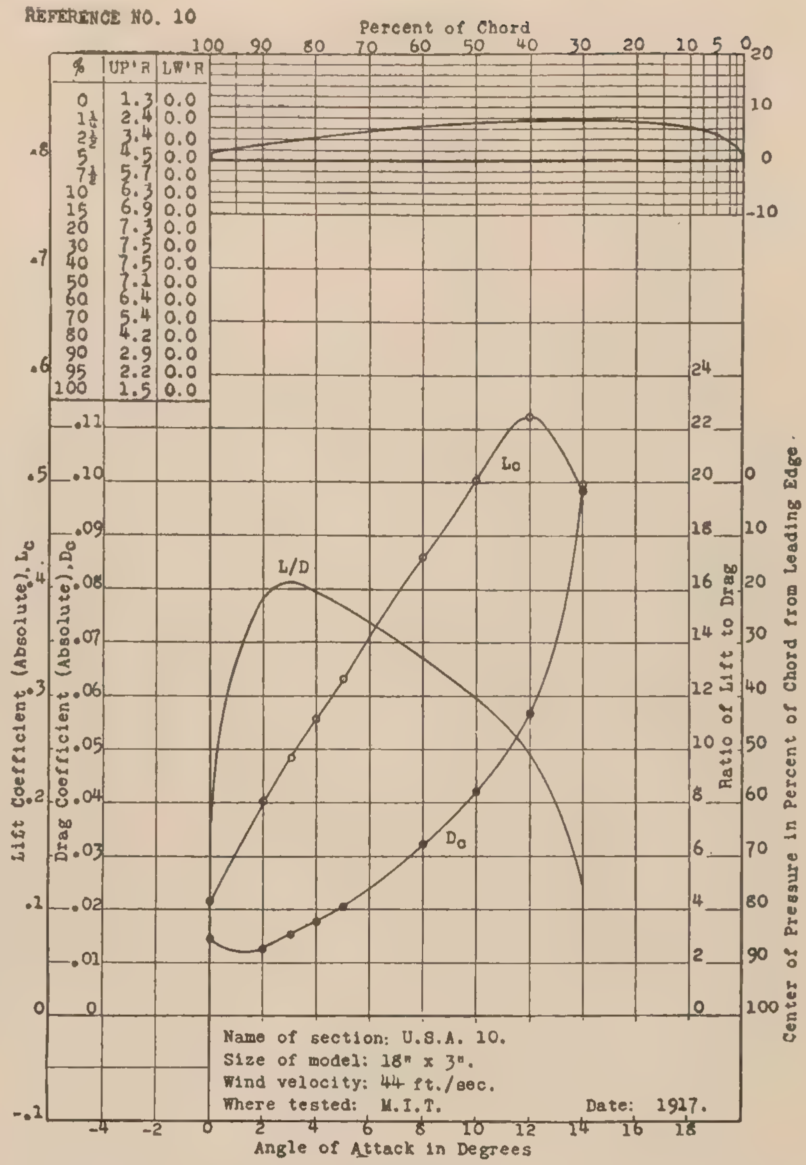


%	UP'R	LW'R
0	1.3	0.0
1¼	4.7	-0.4
2½	6.3	-0.6
5	8.7	-0.8
7½	10.2	-1.0
10	11.3	-1.1
15	12.5	-1.2
20	13.3	-1.2
30	13.9	-1.3
40	13.8	-1.3
50	13.0	-1.2
60	11.6	-1.1
70	9.7	-0.9
80	7.3	-0.7
90	4.5	-0.4
95	3.0	-0.2
100	1.4	0.0

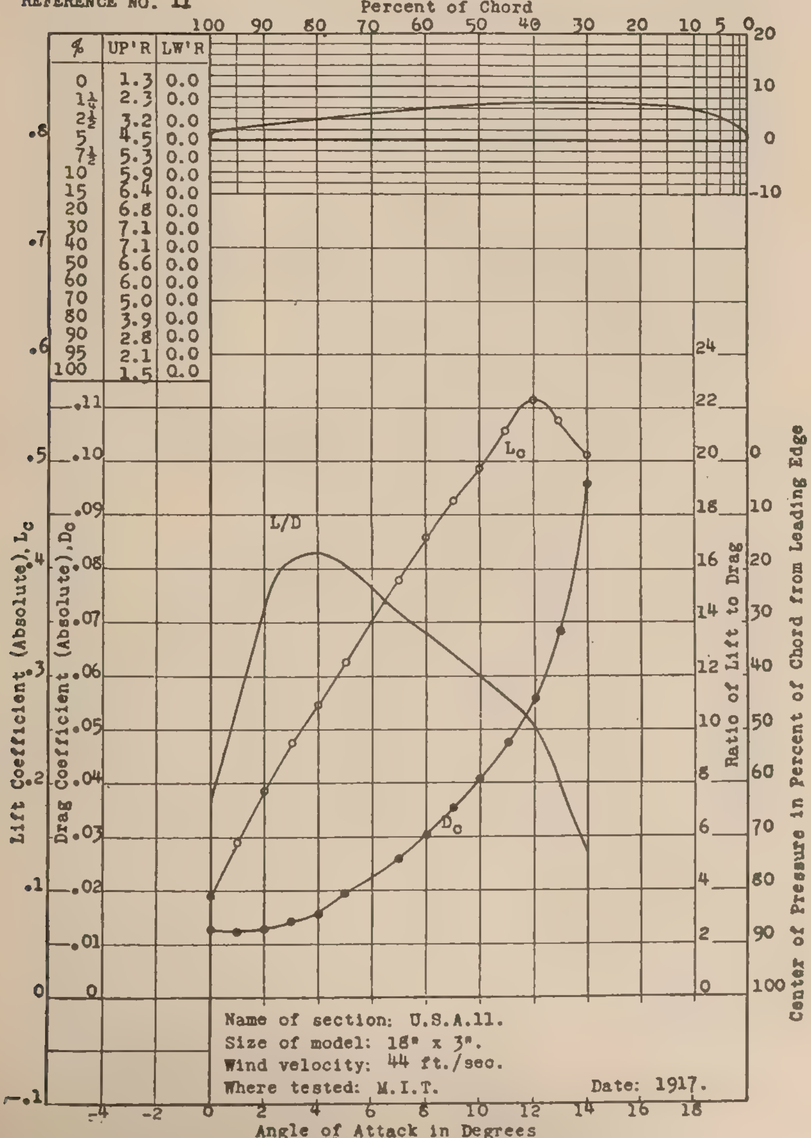
REFERENCE NO. 9



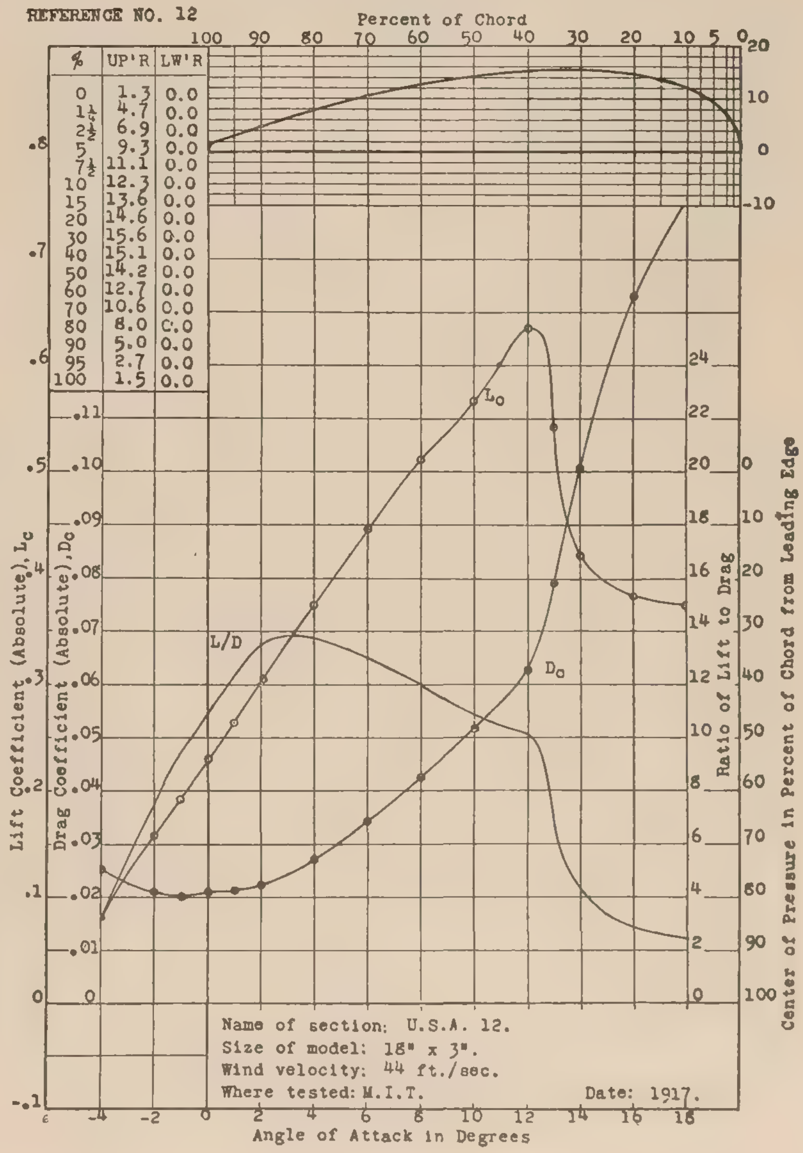
REFERENCE NO. 10



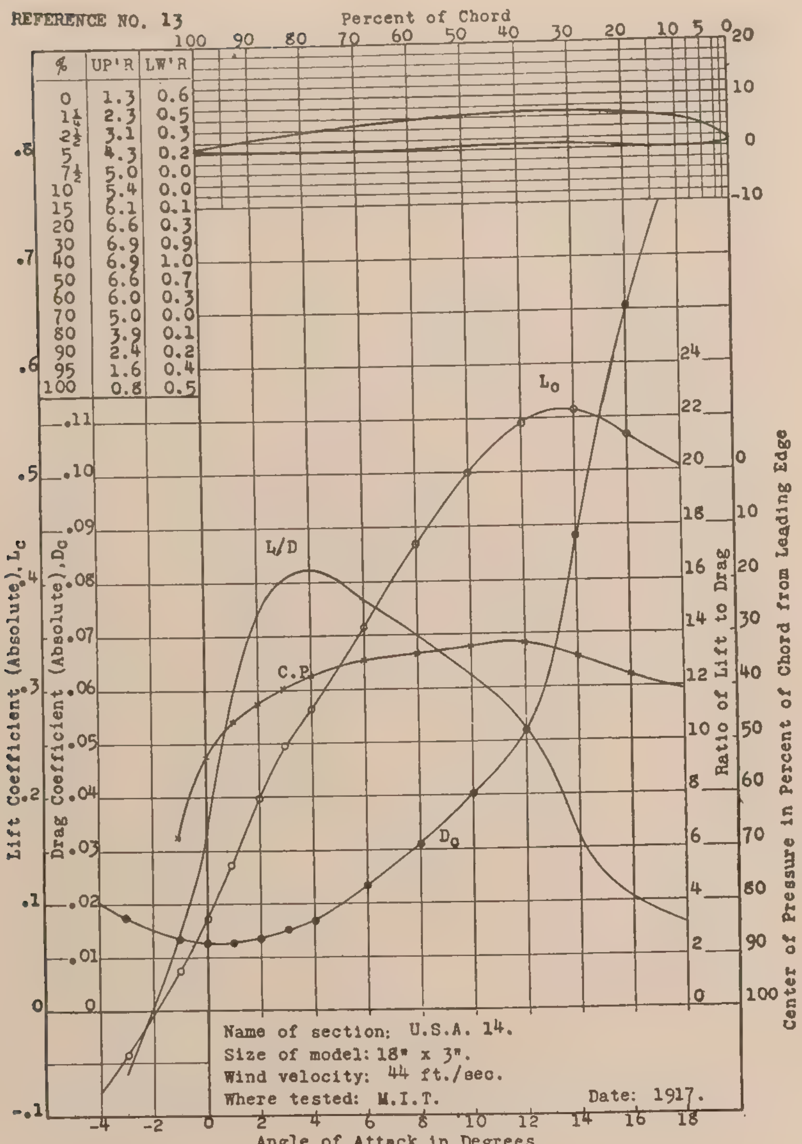
REFERENCE NO. 11



REFERENCE NO. 12

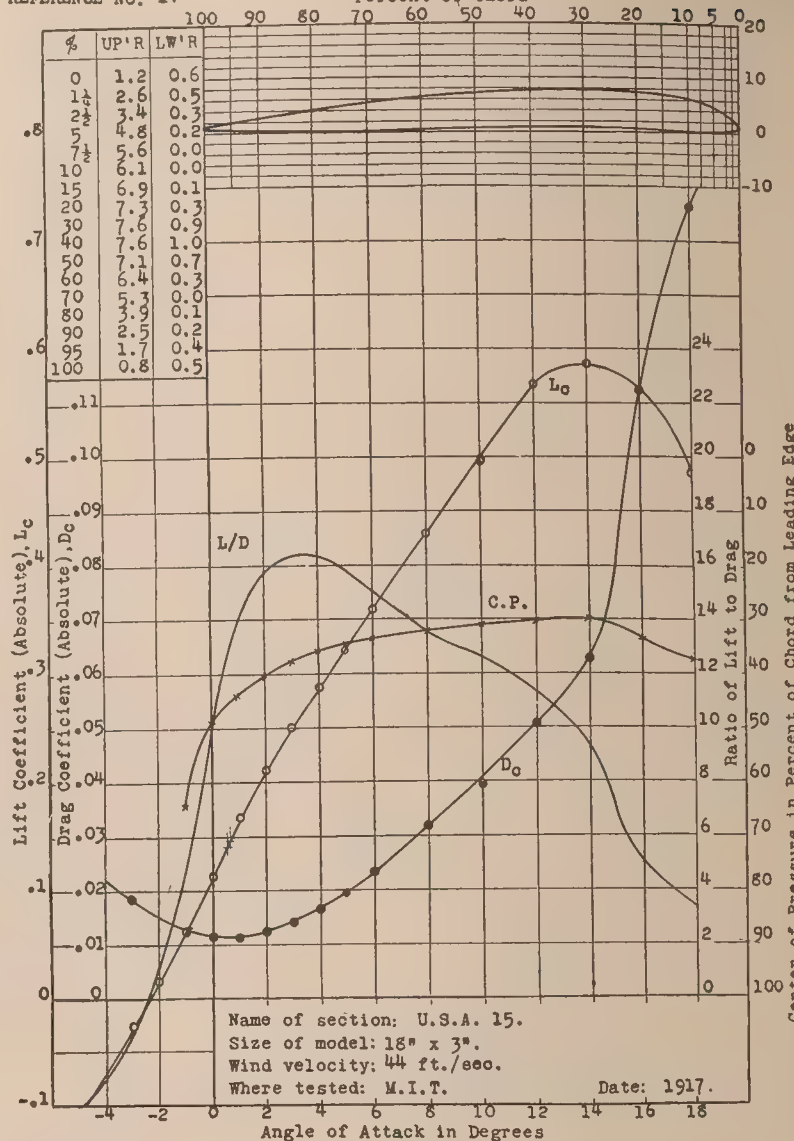


REFERENCE NO. 13



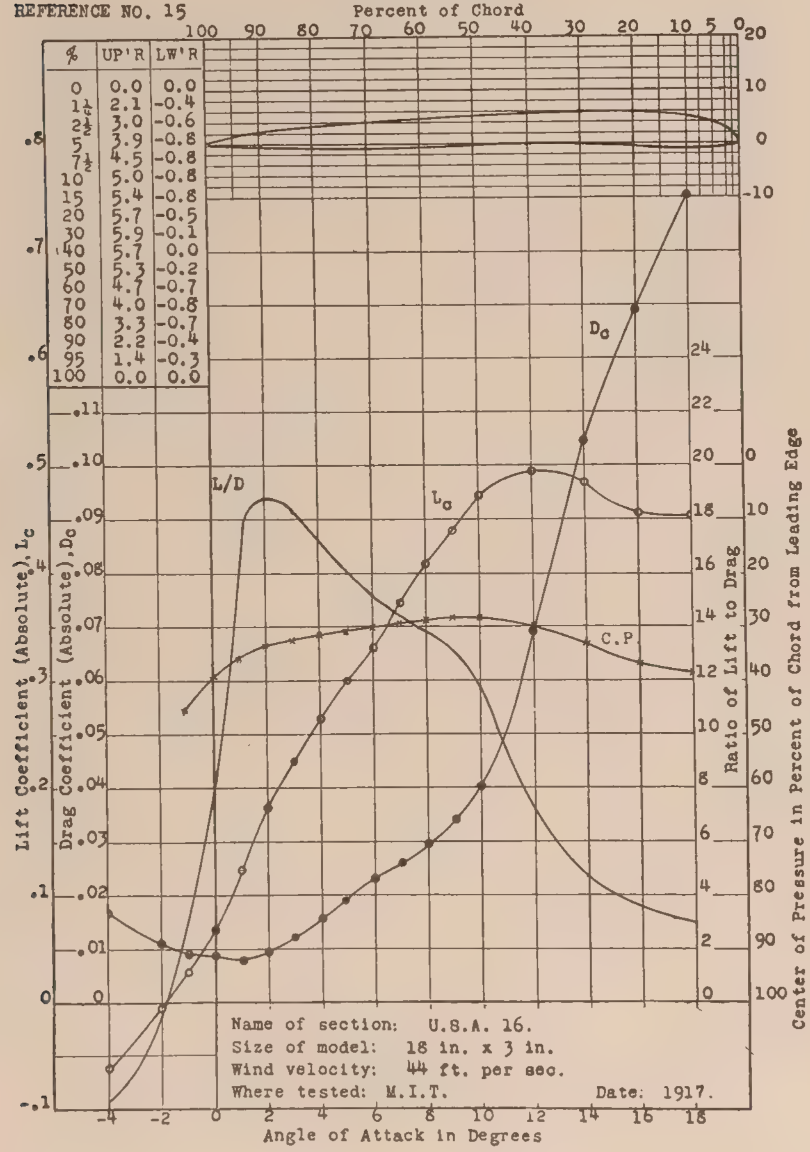
%	UP'R	LW'R
0	1.3	0.6
1¼	2.3	0.5
2½	3.1	0.3
5	4.3	0.2
7½	5.0	0.0
10	5.4	0.0
15	6.1	0.1
20	6.6	0.3
30	6.9	0.9
40	6.9	1.0
50	6.6	0.7
60	6.0	0.3
70	5.0	0.0
80	3.9	0.1
90	2.4	0.2
95	1.6	0.4
100	0.8	0.5

REFERENCE NO. 14



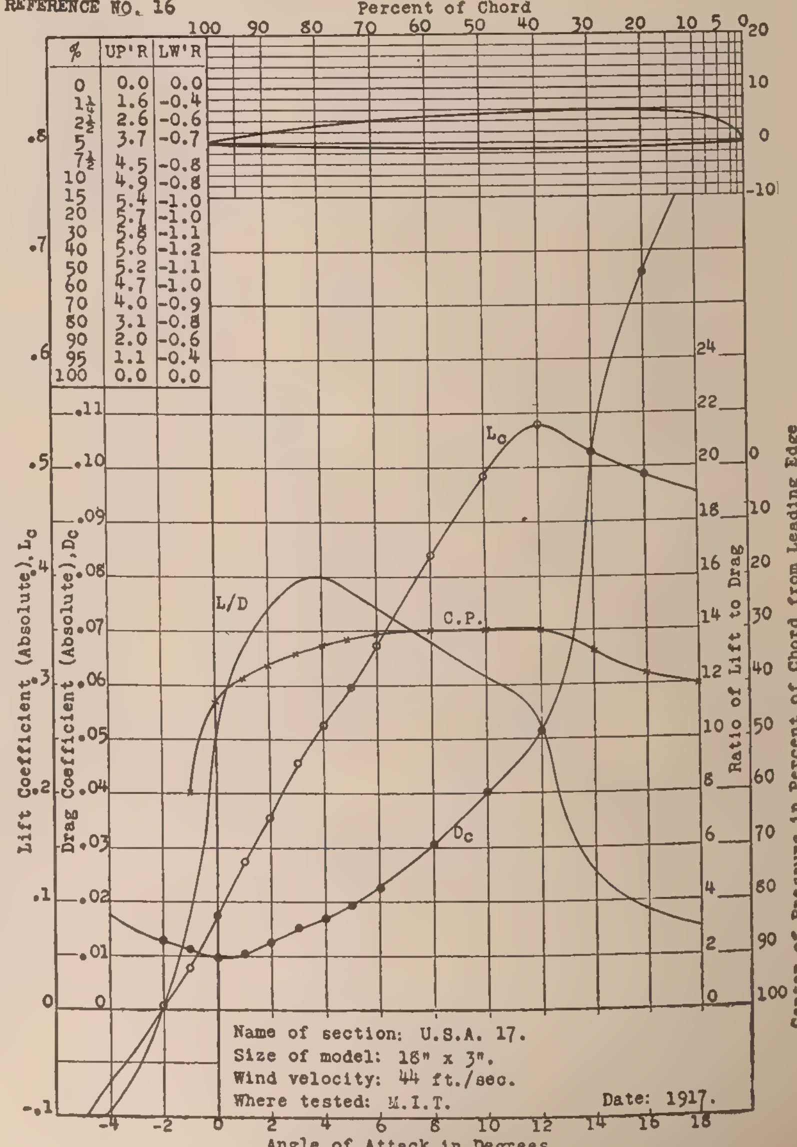
%	UP'R	LW'R
0	1.2	0.6
1¼	2.6	0.5
2½	3.4	0.3
5	4.8	0.2
7½	5.6	0.0
10	6.1	0.0
15	6.9	0.1
20	7.3	0.3
30	7.6	0.9
40	7.6	1.0
50	7.1	0.7
60	6.4	0.3
70	5.3	0.0
80	3.9	0.1
90	2.5	0.2
95	1.7	0.4
100	0.8	0.5

REFERENCE NO. 15



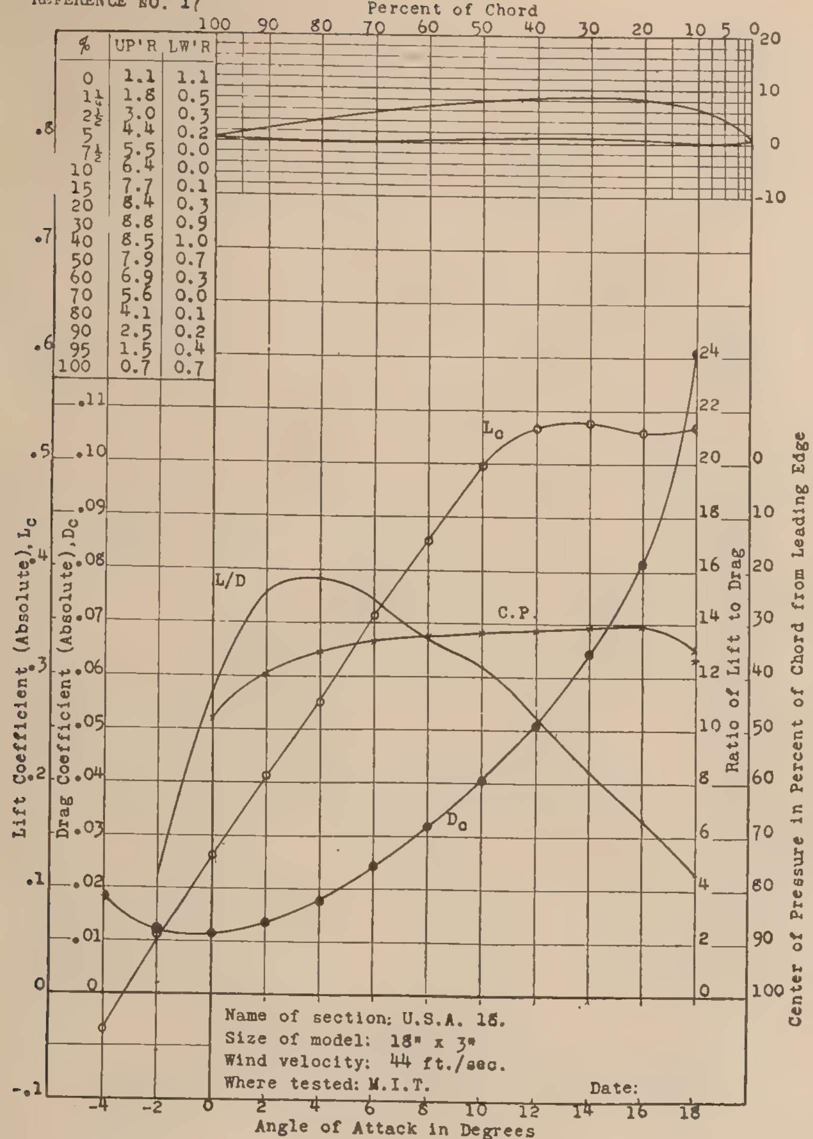
%	UP'R	LW'R
0	0.0	0.0
1¼	2.1	-0.4
2½	3.0	-0.6
5	3.9	-0.8
7½	4.5	-0.8
10	5.0	-0.8
15	5.4	-0.8
20	5.7	-0.5
30	5.9	-0.1
40	5.7	0.0
50	5.3	-0.2
60	4.7	-0.7
70	4.0	-0.8
80	3.3	-0.7
90	2.2	-0.4
95	1.4	-0.3
100	0.0	0.0

REFERENCE NO. 16



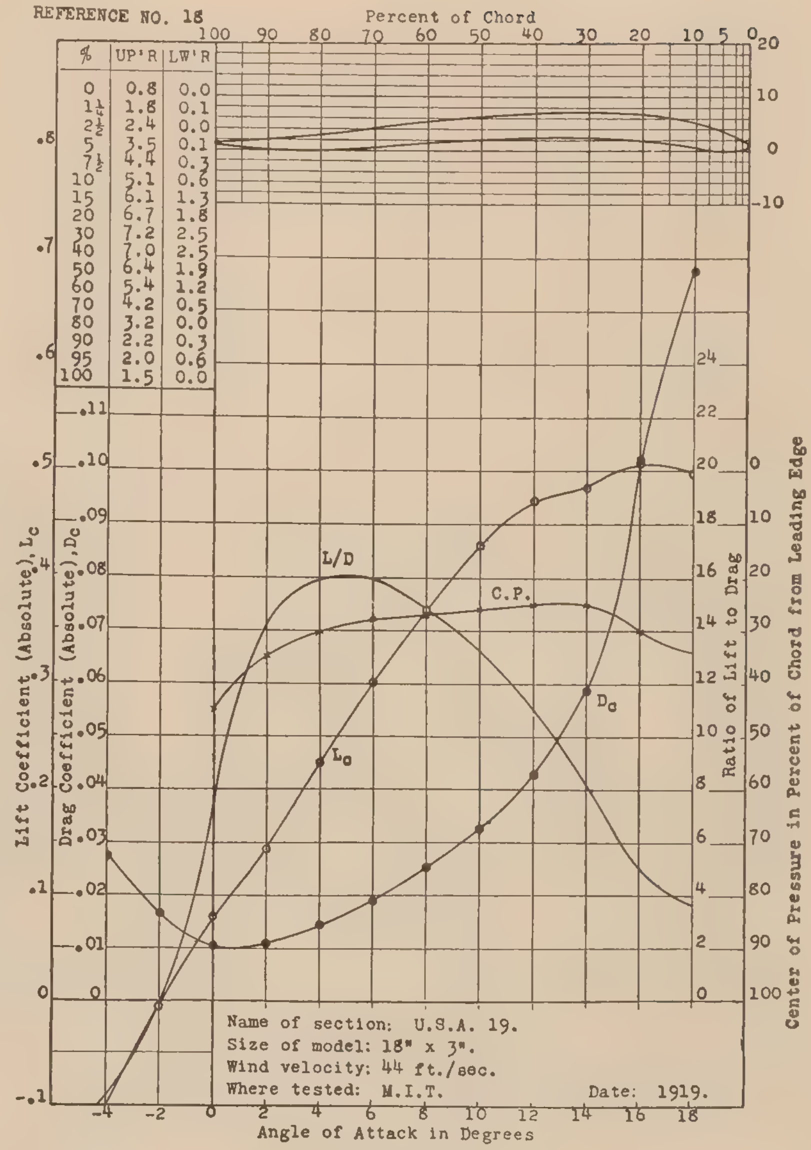
%	UP'R	LW'R
0	0.0	0.0
1¼	1.6	-0.4
2½	2.6	-0.6
5	3.7	-0.7
7½	4.5	-0.8
10	4.9	-0.8
15	5.4	-1.0
20	5.7	-1.0
30	5.8	-1.1
40	5.6	-1.2
50	5.2	-1.1
60	4.7	-1.0
70	4.0	-0.9
80	3.1	-0.8
90	2.0	-0.6
95	1.1	-0.4
100	0.0	0.0

REFERENCE NO. 17



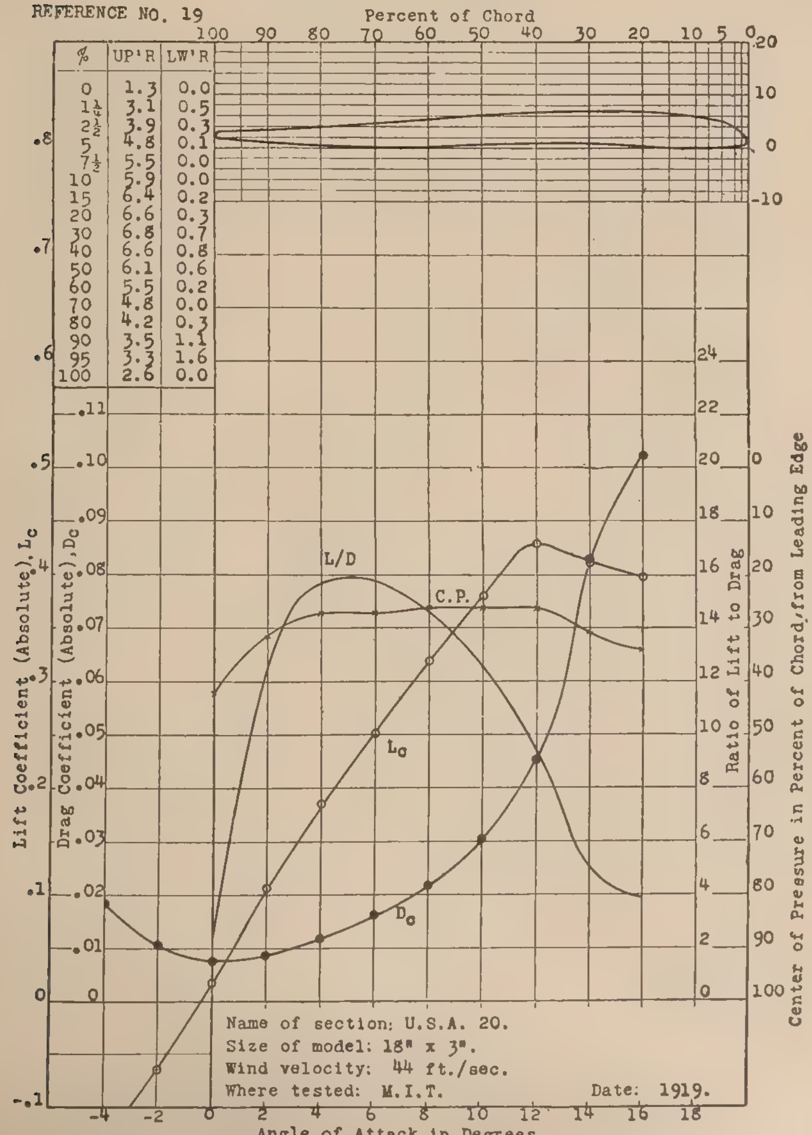
%	UP'R	LW'R
0	1.1	1.1
1¼	1.8	0.5
2½	3.0	0.3
5	4.4	0.2
7½	5.5	0.0
10	6.4	0.0
15	7.7	0.1
20	8.4	0.3
30	8.8	0.9
40	8.5	1.0
50	7.9	0.7
60	6.9	0.3
70	5.6	0.0
80	4.1	0.1
90	2.5	0.2
95	1.5	0.4
100	0.7	0.7

REFERENCE NO. 18



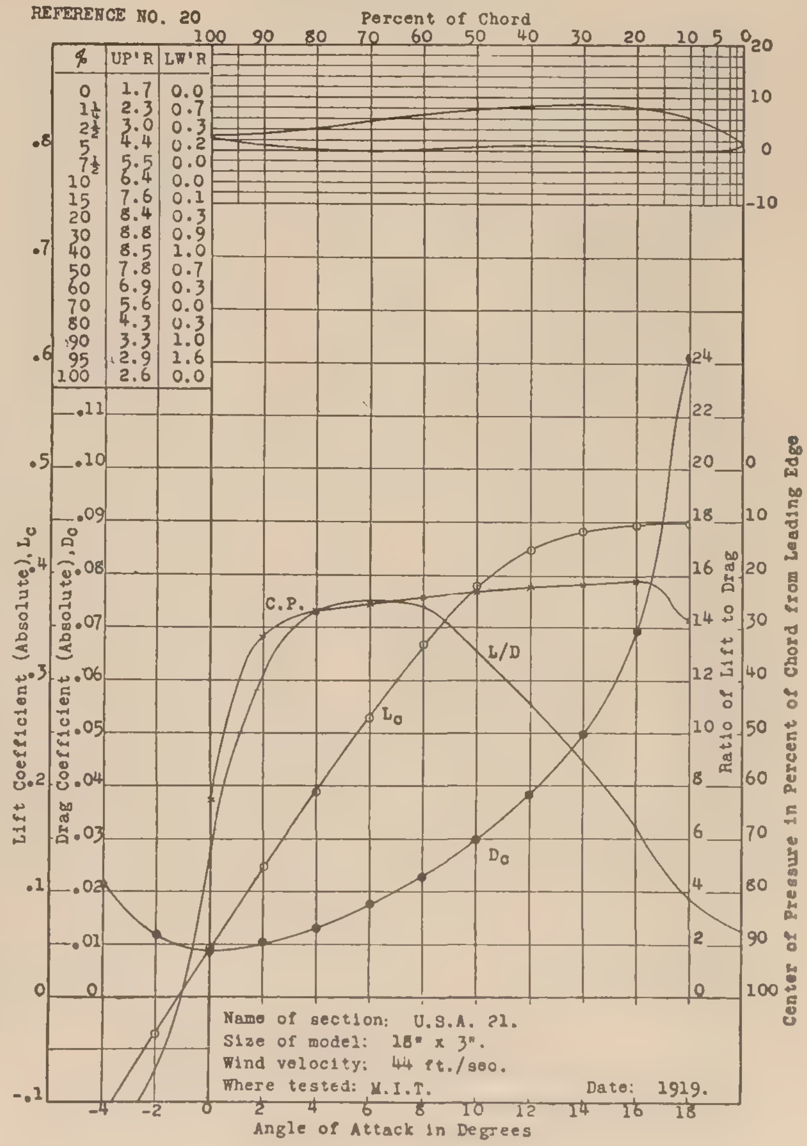
%	UP'R	LW'R
0	0.8	0.0
1¼	1.8	0.1
2½	2.4	0.0
5	3.5	0.1
7½	4.4	0.3
10	5.1	0.6
15	6.1	1.3
20	6.7	1.8
30	7.2	2.5
40	7.0	2.5
50	6.4	1.9
60	5.4	1.2
70	4.2	0.5
80	3.2	0.0
90	2.2	0.3
95	2.0	0.6
100	1.5	0.0

REFERENCE NO. 19



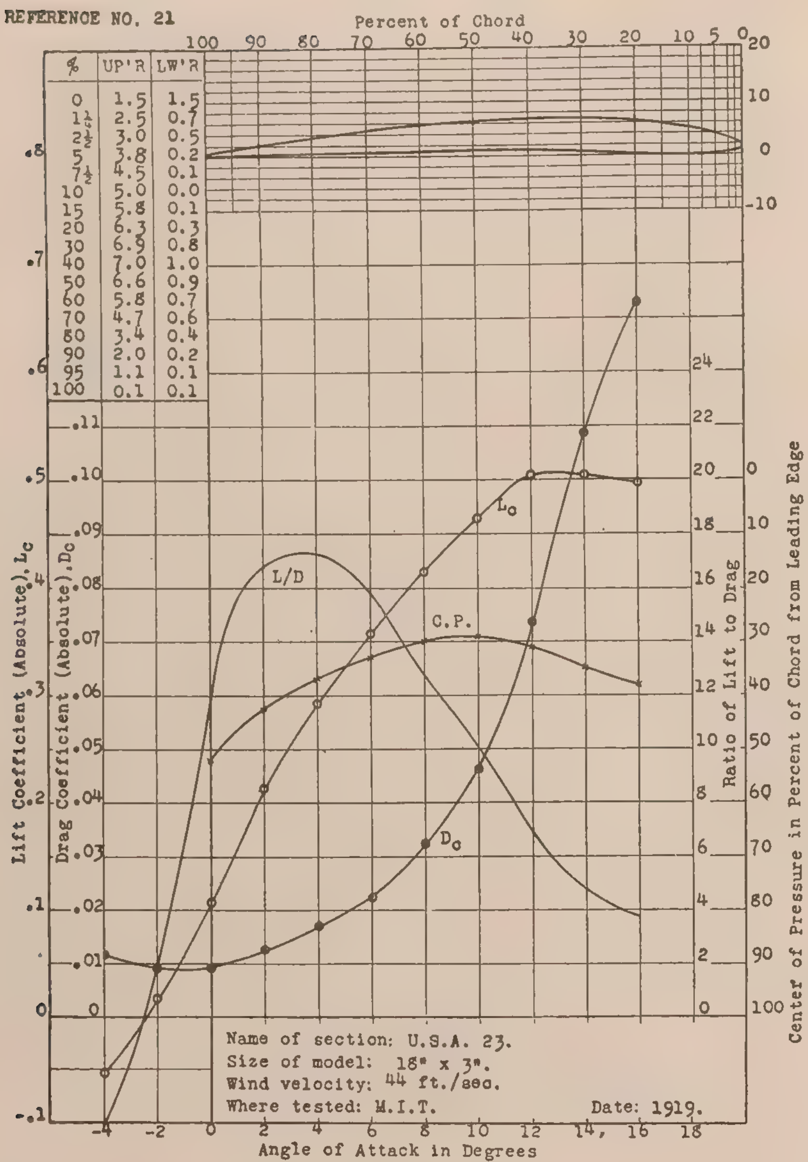
%	UP'R	LW'R
0	1.3	0.0
1¼	3.1	0.5
2½	3.9	0.3
5	4.8	0.1
7½	5.5	0.0
10	5.9	0.0
15	6.4	0.2
20	6.6	0.3
30	6.8	0.7
40	6.6	0.8
50	6.1	0.6
60	5.5	0.2
70	4.8	0.0
80	4.2	0.3
90	3.5	1.1
95	3.3	1.6
100	2.6	0.0

REFERENCE NO. 20



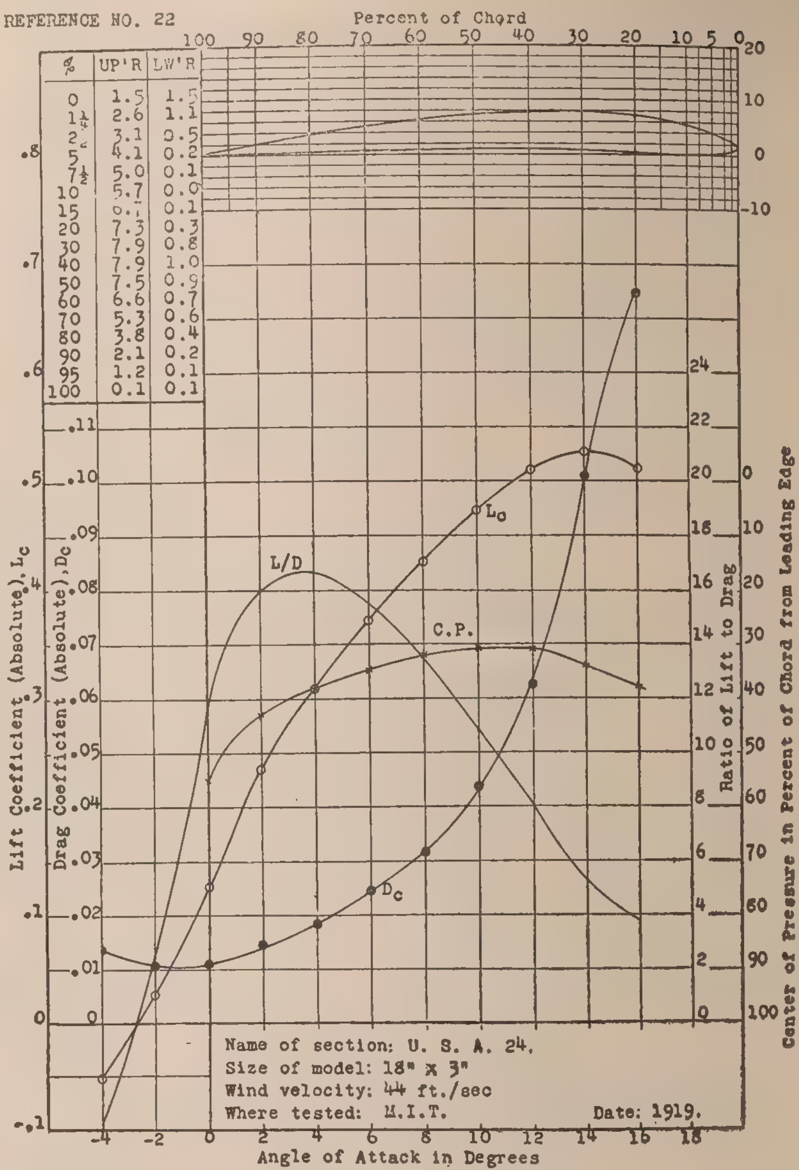
%	UP'R	LW'R
0	1.7	0.0
1¼	2.3	0.7
2½	3.0	0.3
5	4.4	0.2
7½	5.5	0.0
10	6.4	0.0
15	7.6	0.1
20	8.4	0.3
30	8.8	0.9
40	8.5	1.0
50	7.8	0.7
60	6.9	0.3
70	5.6	0.0
80	4.3	0.3
90	3.3	1.0
95	2.9	1.6
100	2.6	0.0

REFERENCE NO. 21



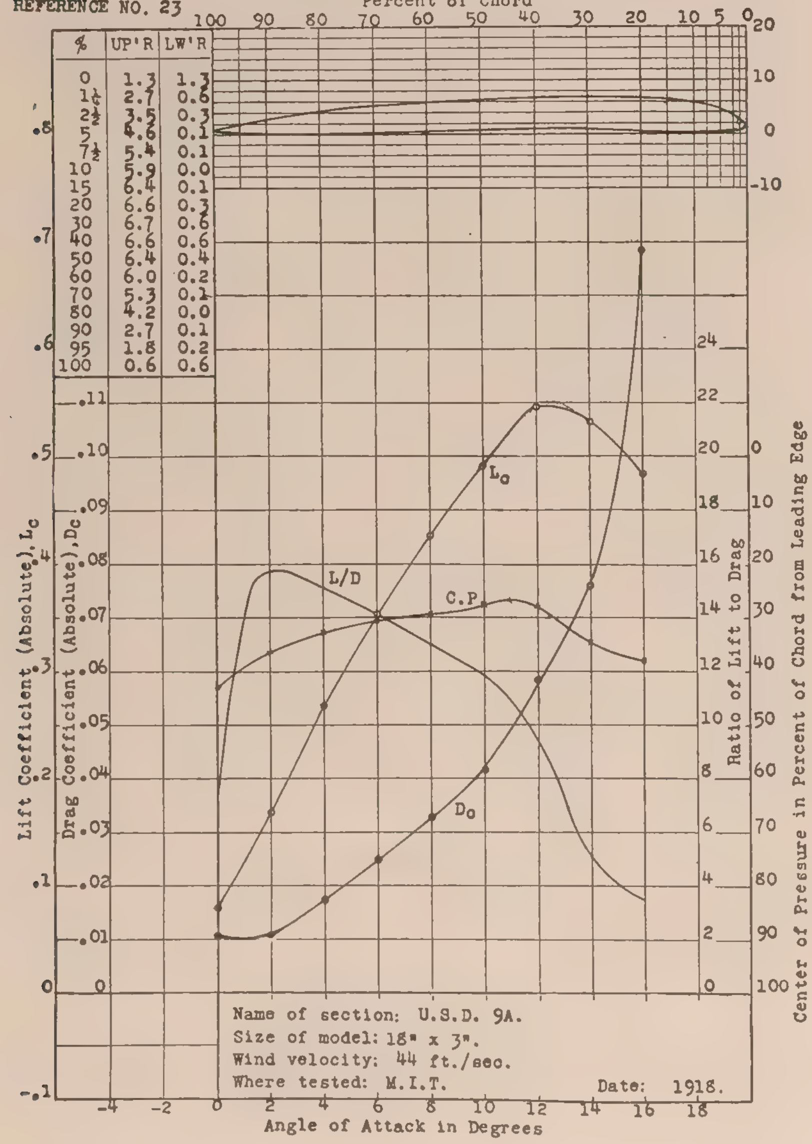
%	UP'R	LW'R
0	1.5	1.5
1¼	2.5	0.7
2½	3.0	0.5
5	3.8	0.2
7½	4.5	0.1
10	5.0	0.0
15	5.8	0.1
20	6.3	0.3
30	6.9	0.8
40	7.0	1.0
50	6.6	0.9
60	5.8	0.7
70	4.7	0.6
80	3.4	0.4
90	2.0	0.2
95	1.1	0.1
100	0.1	0.1

REFERENCE NO. 22



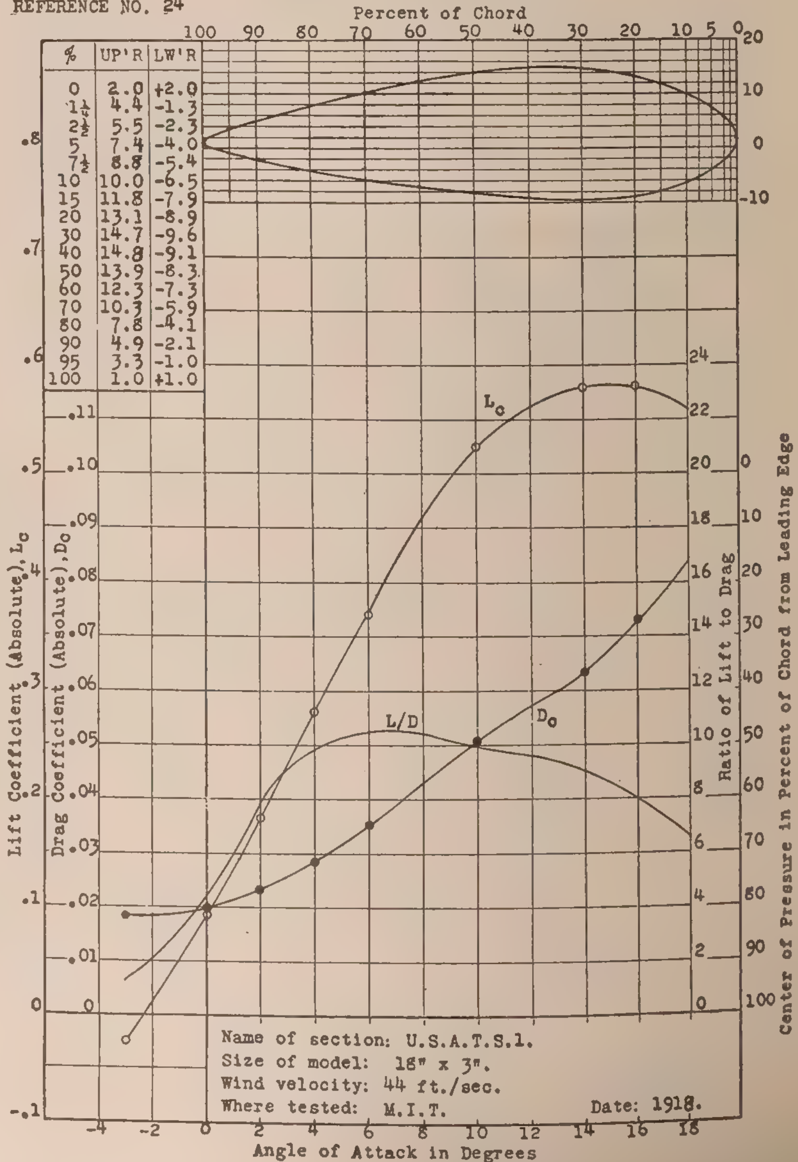
%	UP'R	LW'R
0	1.5	1.5
1¼	2.6	1.1
2½	3.1	0.5
5	4.1	0.2
7½	5.0	0.1
10	5.7	0.0
15	[illegible]	0.1
20	7.3	0.3
30	7.9	0.8
40	7.9	1.0
50	7.5	0.9
60	6.6	0.7
70	5.3	0.6
80	3.8	0.4
90	2.1	0.2
95	1.2	0.1
100	0.1	0.1

REFERENCE NO. 23



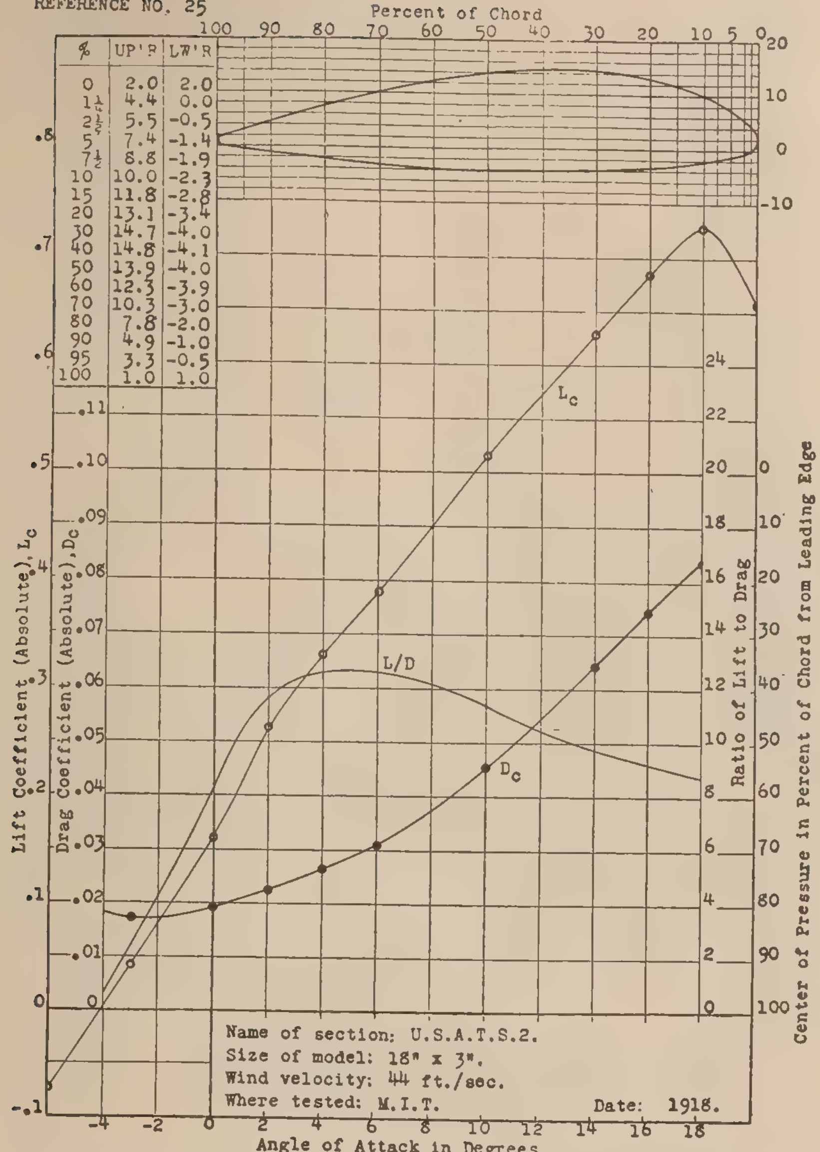
%	UP'R	LW'R
0	1.3	1.3
1¼	2.7	0.6
2½	3.5	0.3
5	4.6	0.1
7½	5.4	0.1
10	5.9	0.0
15	6.4	0.1
20	6.6	0.3
30	6.7	0.6
40	6.6	0.6
50	6.4	0.4
60	6.0	0.2
70	5.3	0.1
80	4.2	0.0
90	2.7	0.1
95	1.8	0.2
100	0.6	0.6

REFERENCE NO. 24



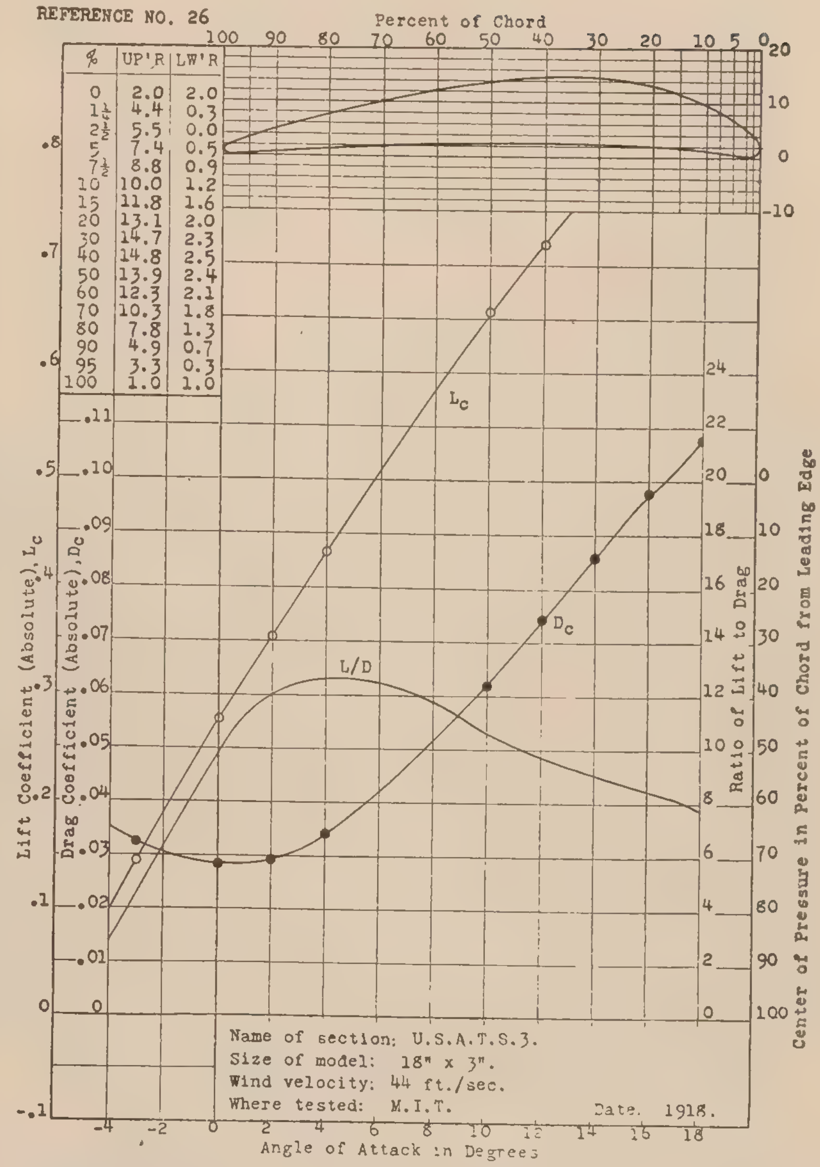
%	UP'R	LW'R
0	2.0	+2.0
1¼	4.4	-1.3
2½	5.5	-2.3
5	7.4	-4.0
7½	8.8	-5.4
10	10.0	-6.5
15	11.8	-7.9
20	13.1	-8.9
30	14.7	-9.6
40	14.8	-9.1
50	13.9	-8.3
60	12.3	-7.3
70	10.3	-5.9
80	7.8	-4.1
90	4.9	-2.1
95	3.3	-1.0
100	1.0	+1.0

REFERENCE NO. 25



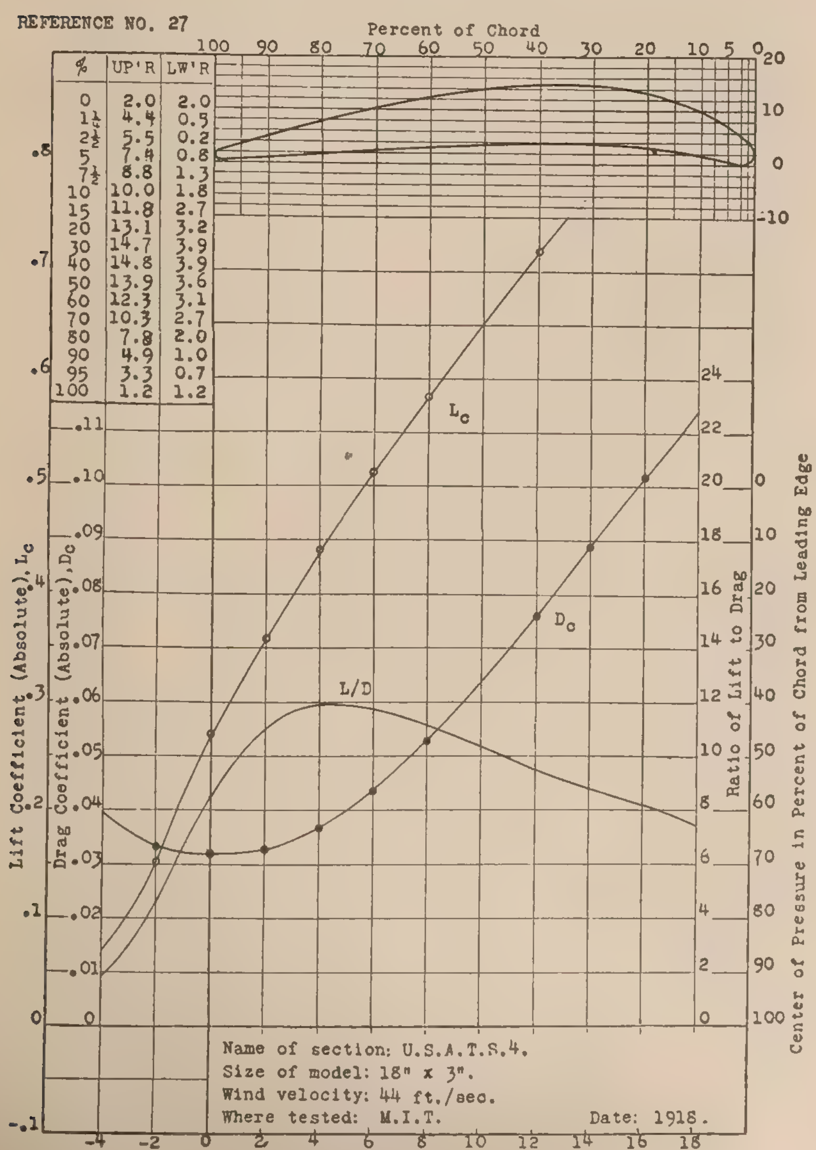
%	UP'R	LW'R
0	2.0	2.0
1½	4.4	0.0
2½	5.5	-0.5
5	7.4	-1.4
7½	8.8	-1.9
10	10.0	-2.3
15	11.8	-2.8
20	13.1	-3.4
30	14.7	-4.0
40	14.8	-4.1
50	13.9	-4.0
60	12.3	-3.9
70	10.3	-3.0
80	7.8	-2.0
90	4.9	-1.0
95	3.3	-0.5
100	1.0	1.0

REFERENCE NO. 26



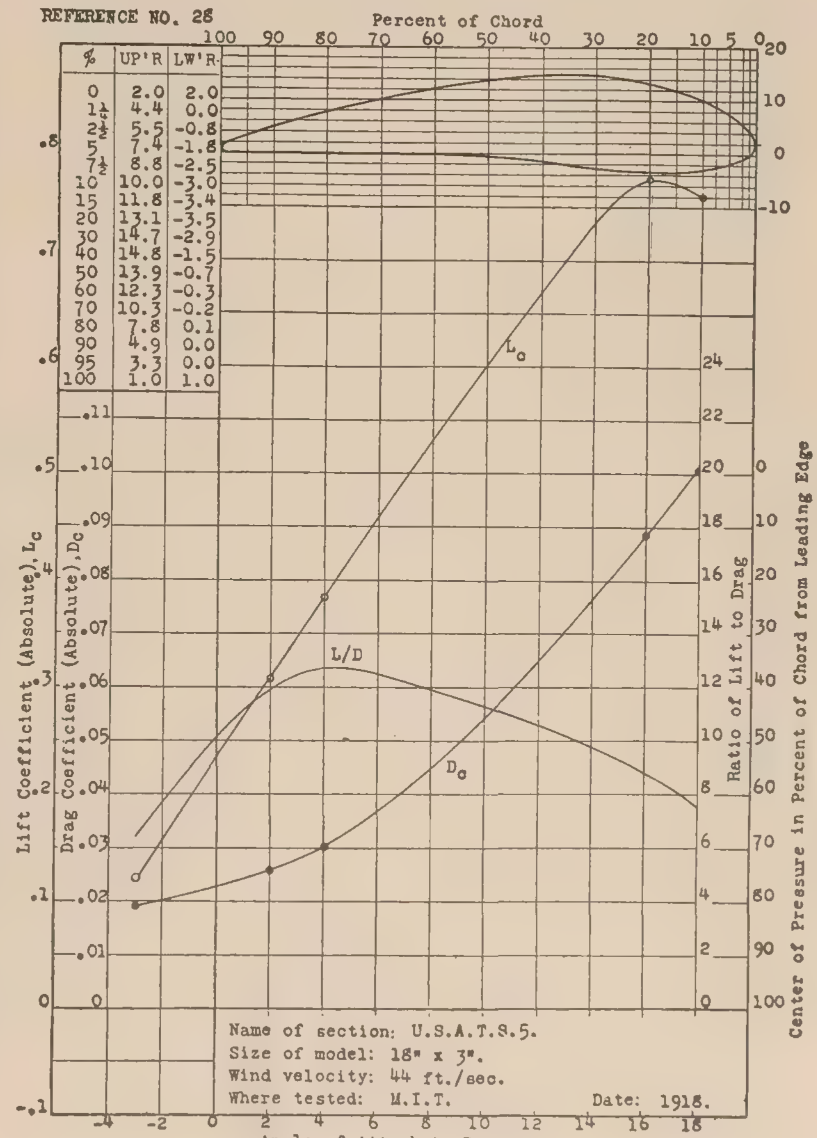
%	UP'R	LW'R
0	2.0	2.0
1½	4.4	0.3
2½	5.5	0.0
5	7.4	0.5
7½	8.8	0.9
10	10.0	1.2
15	11.8	1.6
20	13.1	2.0
30	14.7	2.3
40	14.8	2.5
50	13.9	2.4
60	12.3	2.1
70	10.3	1.8
80	7.8	1.3
90	4.9	0.7
95	3.3	0.3
100	1.0	1.0

REFERENCE NO. 27



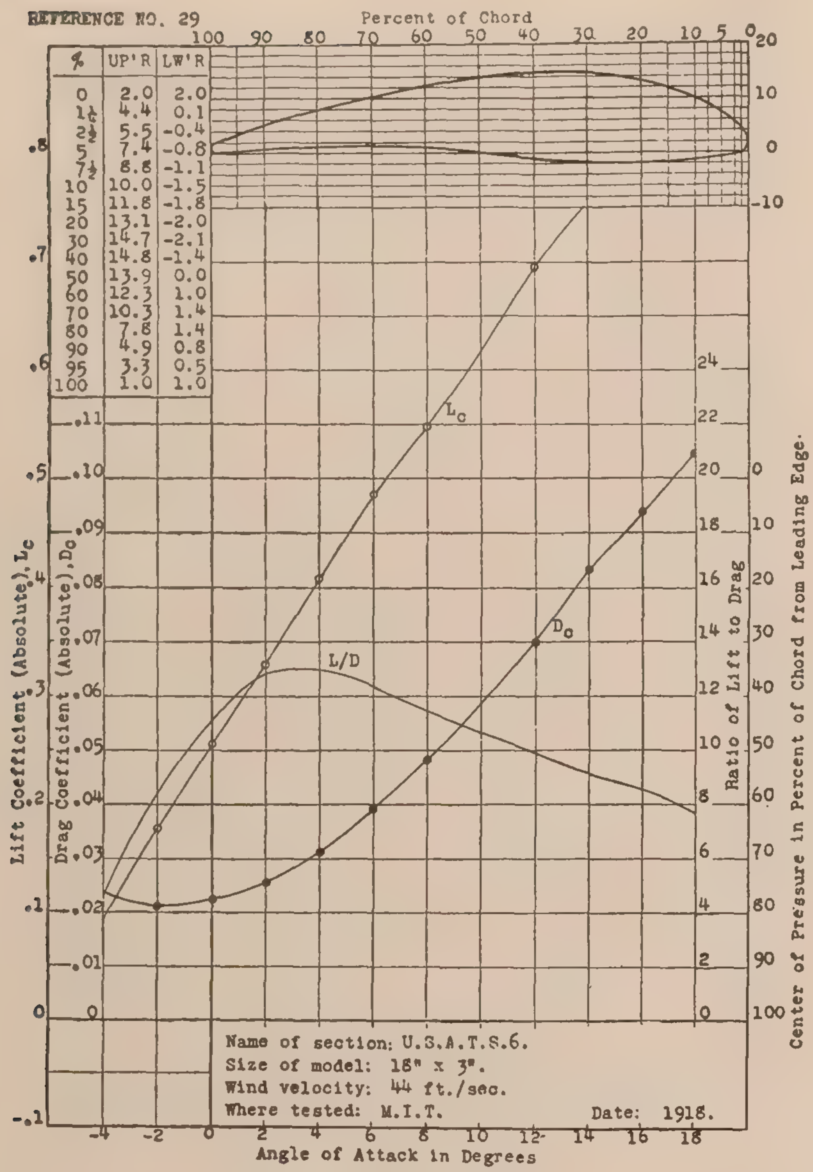
%	UP'R	LW'R
0	2.0	2.0
1½	4.4	0.5
2½	5.5	0.2
5	7.4	0.8
7½	8.8	1.3
10	10.0	1.8
15	11.8	2.7
20	13.1	3.2
30	14.7	3.9
40	14.8	3.9
50	13.9	3.6
60	12.3	3.1
70	10.3	2.7
80	7.8	2.0
90	4.9	1.0
95	3.3	0.7
100	1.2	1.2

REFERENCE NO. 28



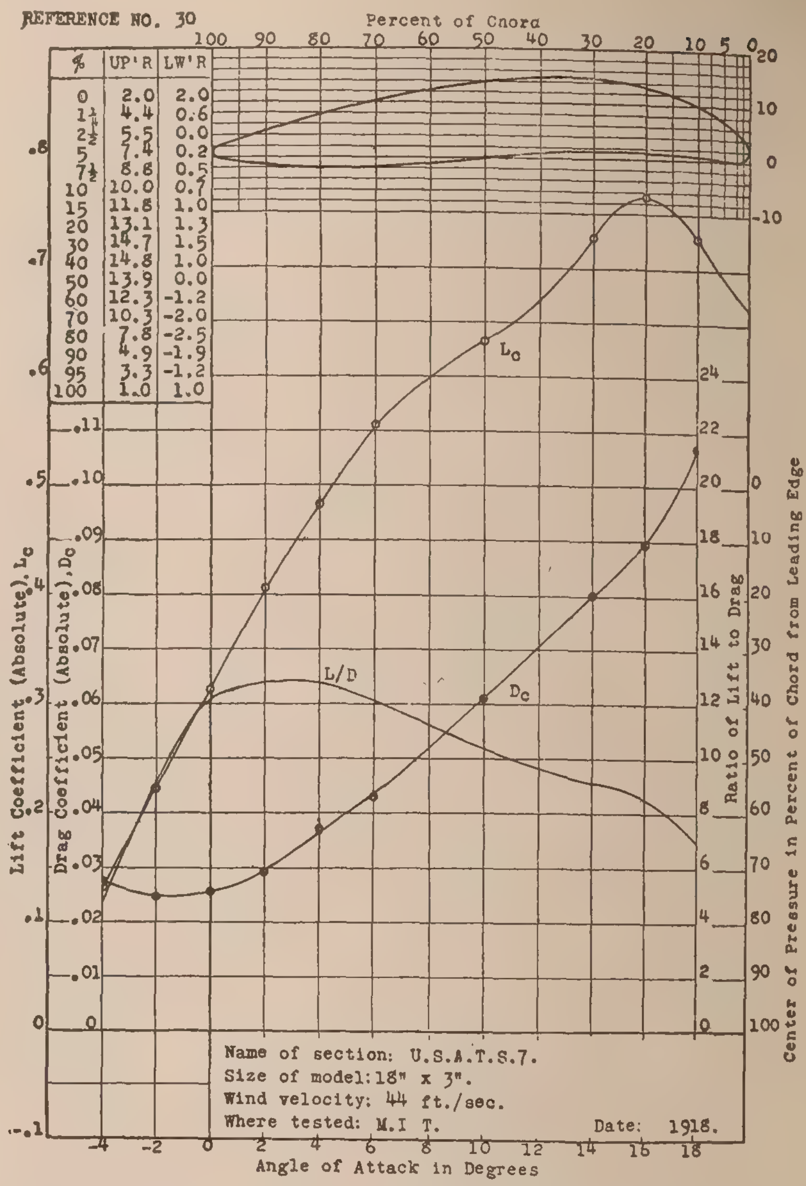
%	UP'R	LW'R
0	2.0	2.0
1½	4.4	0.0
2½	5.5	-0.8
5	7.4	-1.8
7½	8.8	-2.5
10	10.0	-3.0
15	11.8	-3.4
20	13.1	-3.5
30	14.7	-2.9
40	14.8	-1.5
50	13.9	-0.7
60	12.3	-0.3
70	10.3	-0.2
80	7.8	0.1
90	4.9	0.0
95	3.3	0.0
100	1.0	1.0

REFERENCE NO. 29



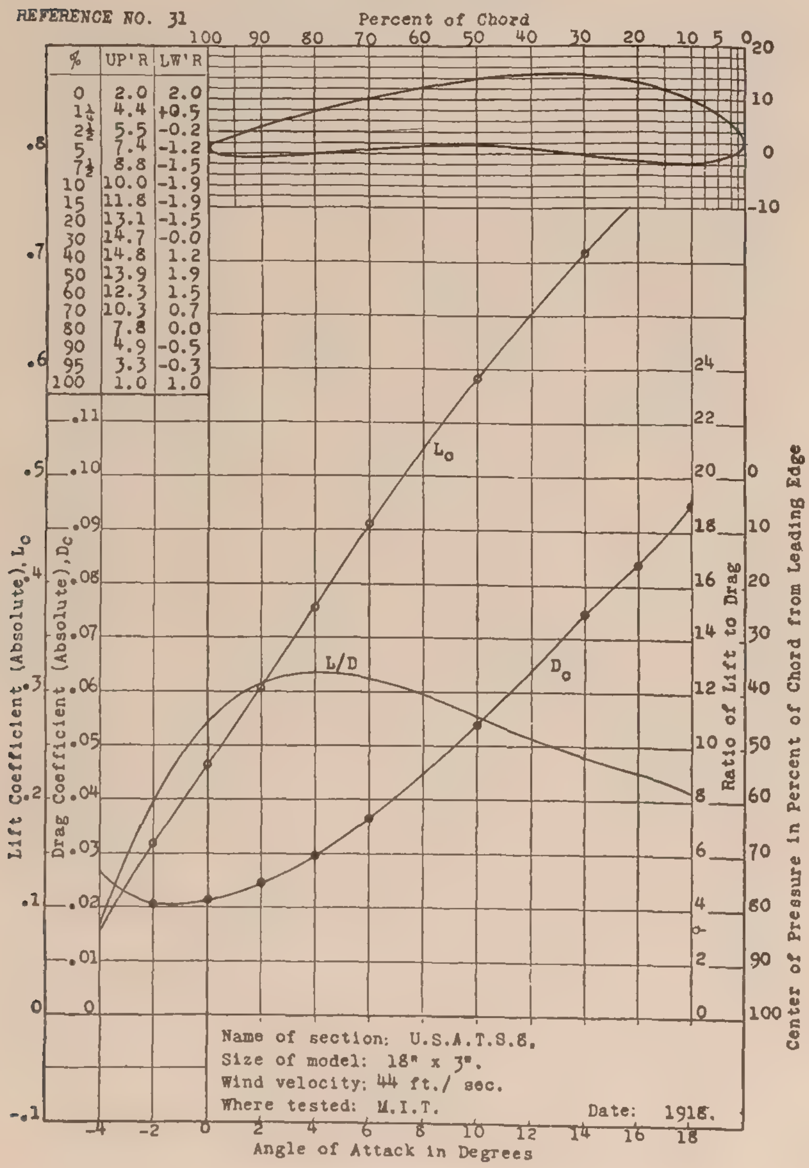
%	UP'R	LW'R
0	2.0	2.0
1¼	4.4	0.1
2½	5.5	-0.4
5	7.4	-0.8
7½	8.8	-1.1
10	10.0	-1.5
15	11.8	-1.8
20	13.1	-2.0
30	14.7	-2.1
40	14.8	-1.4
50	13.9	0.0
60	12.3	1.0
70	10.3	1.4
80	7.8	1.4
90	4.9	0.8
95	3.3	0.5
100	1.0	1.0

REFERENCE NO. 30



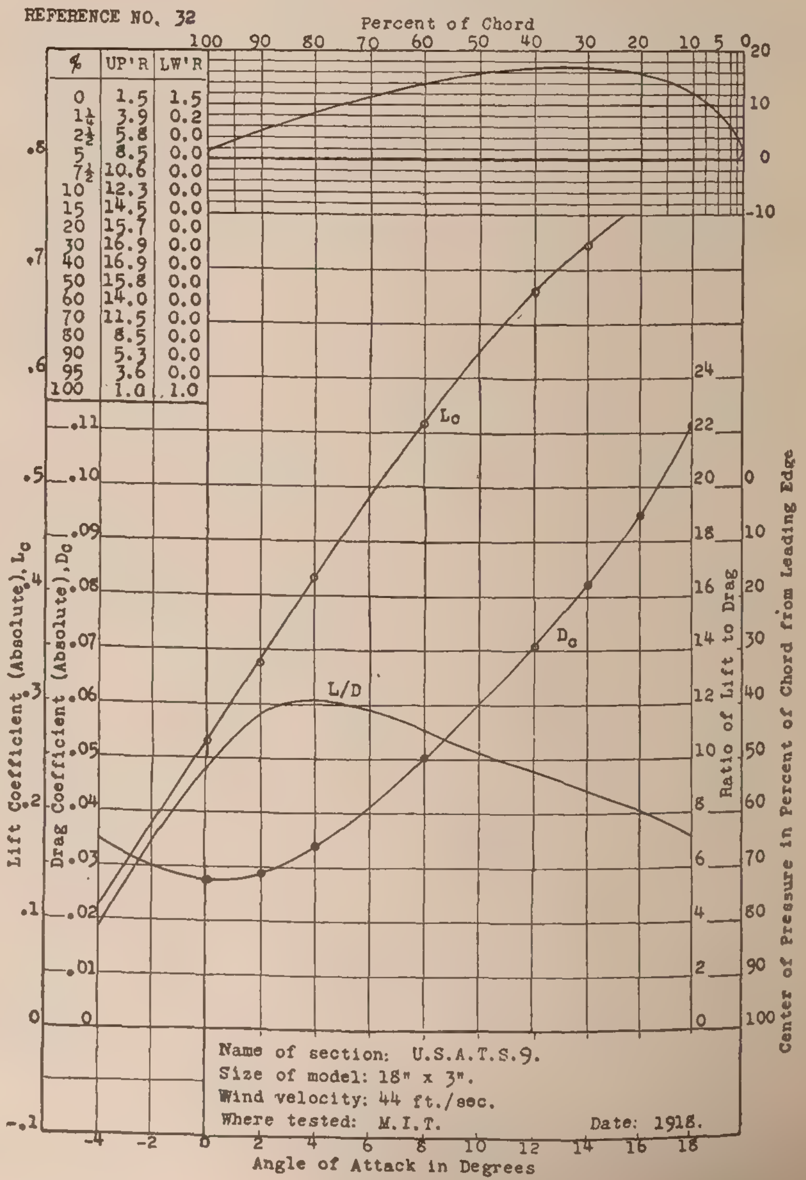
%	UP'R	LW'R
0	2.0	2.0
1¼	4.4	0.6
2½	5.5	0.0
5	7.4	0.2
7½	8.8	0.5
10	10.0	0.7
15	11.8	1.0
20	13.1	1.3
30	14.7	1.5
40	14.8	1.0
50	13.9	0.0
60	12.3	-1.2
70	10.3	-2.0
80	7.8	-2.5
90	4.9	-1.9
95	3.3	-1.2
100	1.0	1.0

REFERENCE NO. 31



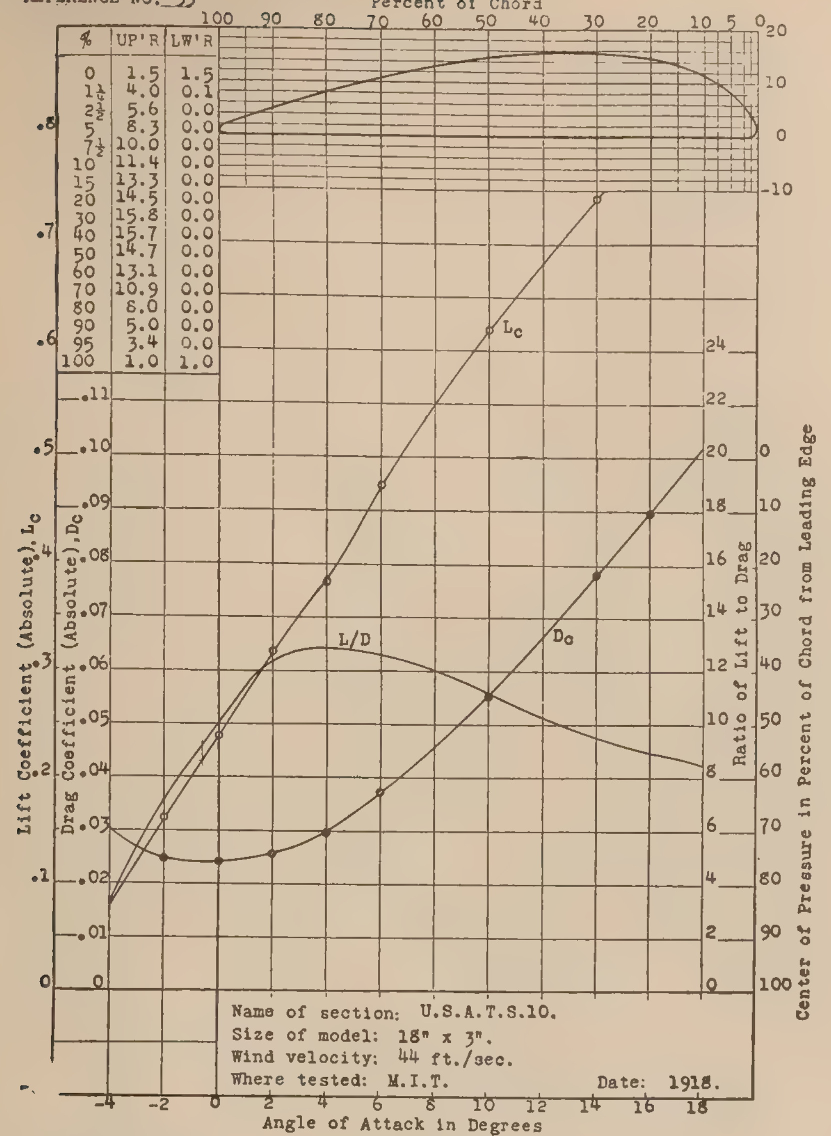
%	UP'R	LW'R
0	2.0	2.0
1¼	4.4	+0.5
2½	5.5	-0.2
5	7.4	-1.2
7½	8.8	-1.5
10	10.0	-1.9
15	11.8	-1.9
20	13.1	-1.5
30	14.7	-0.0
40	14.8	1.2
50	13.9	1.9
60	12.3	1.5
70	10.3	0.7
80	7.8	0.0
90	4.9	-0.5
95	3.3	-0.3
100	1.0	1.0

REFERENCE NO. 32



%	UP'R	LW'R
0	1.5	1.5
1¼	3.9	0.2
2½	5.8	0.0
5	8.5	0.0
7½	10.6	0.0
10	12.3	0.0
15	14.5	0.0
20	15.7	0.0
30	16.9	0.0
40	16.9	0.0
50	15.8	0.0
60	14.0	0.0
70	11.5	0.0
80	8.5	0.0
90	5.3	0.0
95	3.6	0.0
100	1.0	1.0

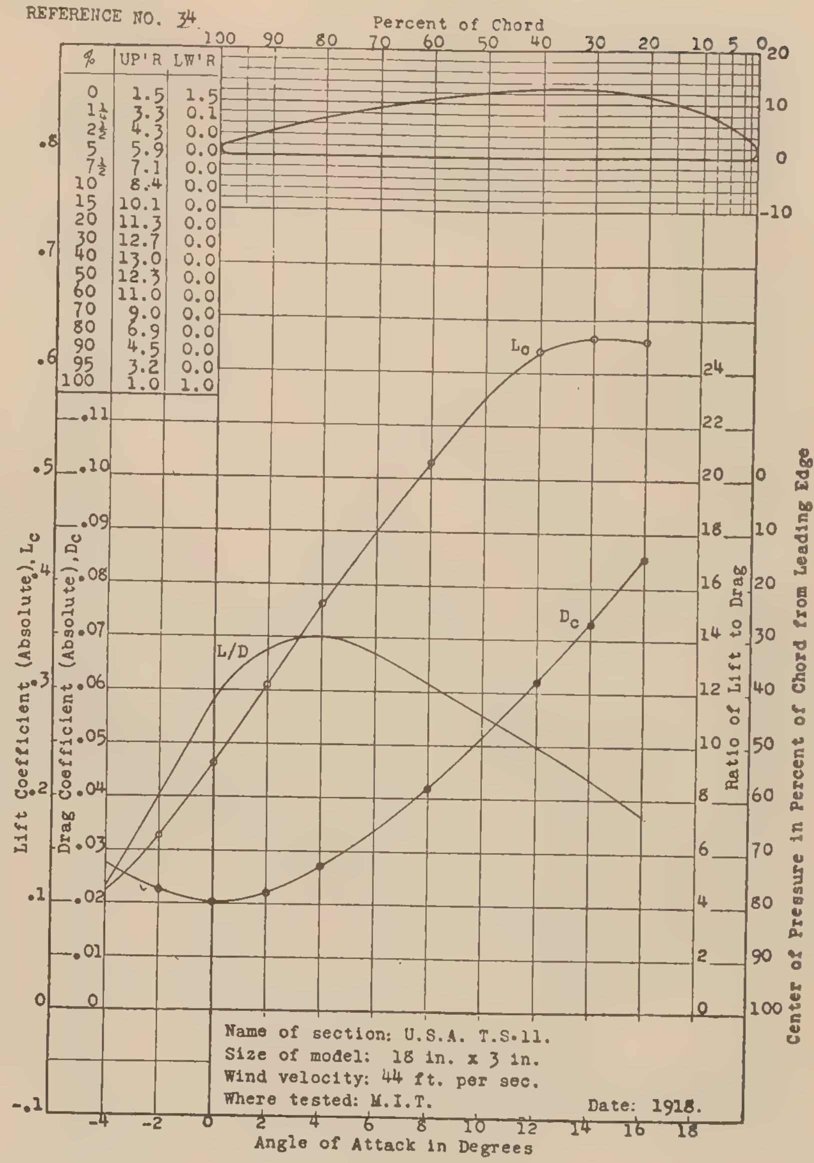
REFERENCE NO. 33



%	UP'R	LW'R
0	1.5	1.5
1¼	4.0	0.1
2½	5.6	0.0
5	8.3	0.0
7½	10.0	0.0
10	11.4	0.0
15	13.3	0.0
20	14.5	0.0
30	15.8	0.0
40	15.7	0.0
50	14.7	0.0
60	13.1	0.0
70	10.9	0.0
80	8.0	0.0
90	5.0	0.0
95	3.4	0.0
100	1.0	1.0

Name of section: U.S.A.T.S.10.
Size of model: 18" x 3".
Wind velocity: 44 ft./sec.
Where tested: M.I.T. Date: 1918.

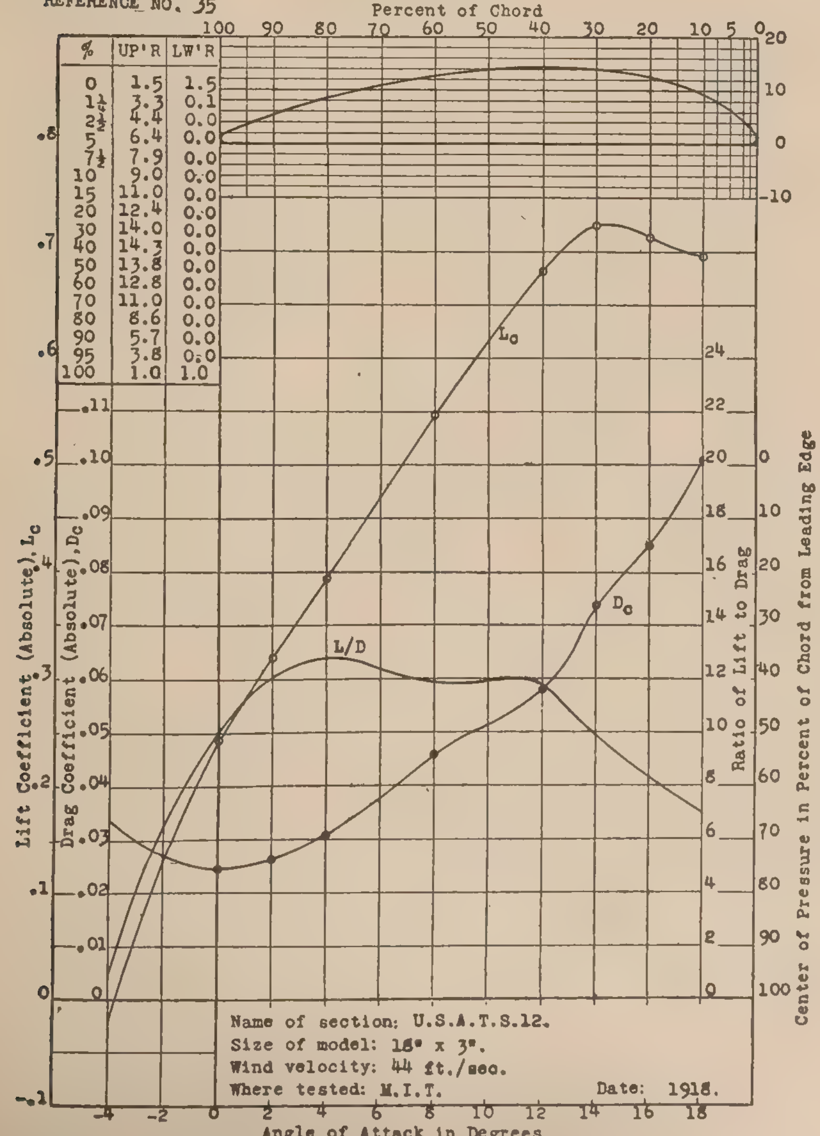
REFERENCE NO. 34



%	UP'R	LW'R
0	1.5	1.5
1¼	3.3	0.1
2½	4.3	0.0
5	5.9	0.0
7½	7.1	0.0
10	8.4	0.0
15	10.1	0.0
20	11.3	0.0
30	12.7	0.0
40	13.0	0.0
50	12.3	0.0
60	11.0	0.0
70	9.0	0.0
80	6.9	0.0
90	4.5	0.0
95	3.2	0.0
100	1.0	1.0

Name of section: U.S.A. T.S.11.
Size of model: 18 in. x 3 in.
Wind velocity: 44 ft. per sec.
Where tested: M.I.T. Date: 1918.

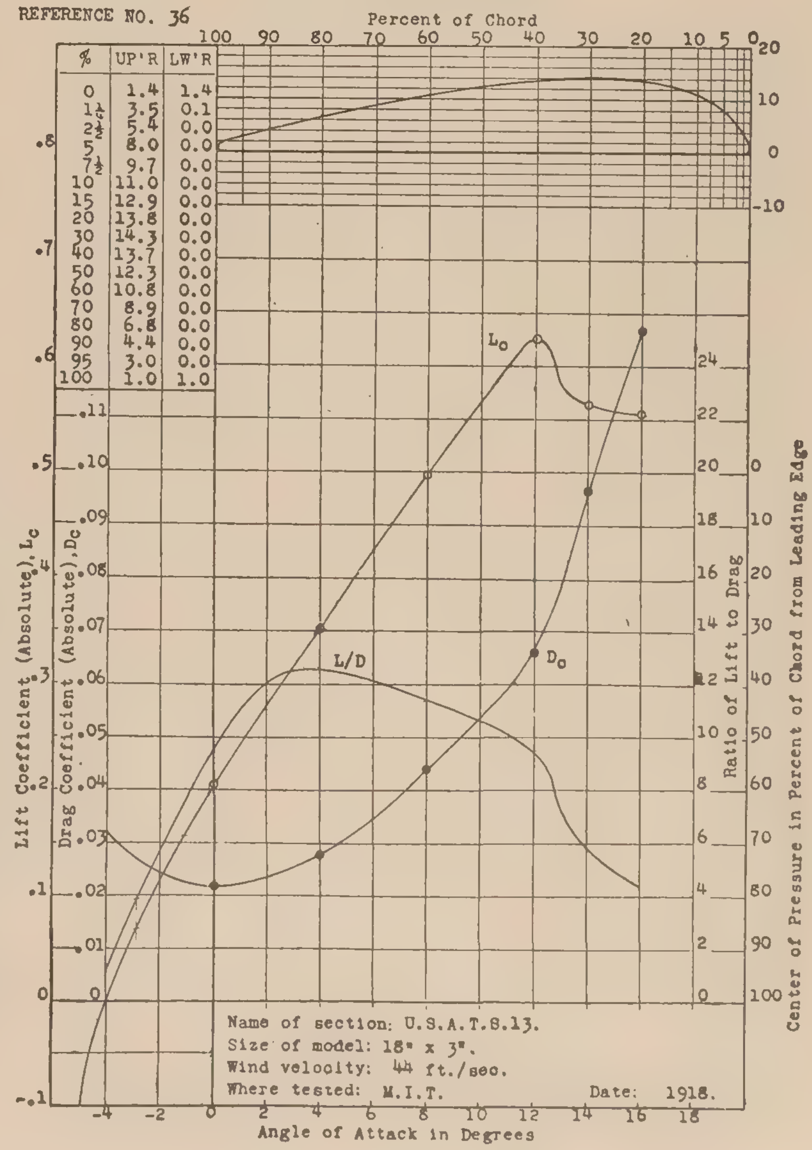
REFERENCE NO. 35



%	UP'R	LW'R
0	1.5	1.5
1¼	3.3	0.1
2½	4.4	0.0
5	6.4	0.0
7½	7.9	0.0
10	9.0	0.0
15	11.0	0.0
20	12.4	0.0
30	14.0	0.0
40	14.3	0.0
50	13.8	0.0
60	12.8	0.0
70	11.0	0.0
80	8.6	0.0
90	5.7	0.0
95	3.8	0.0
100	1.0	1.0

Name of section: U.S.A.T.S.12.
Size of model: 18" x 3".
Wind velocity: 44 ft./sec.
Where tested: M.I.T. Date: 1918.

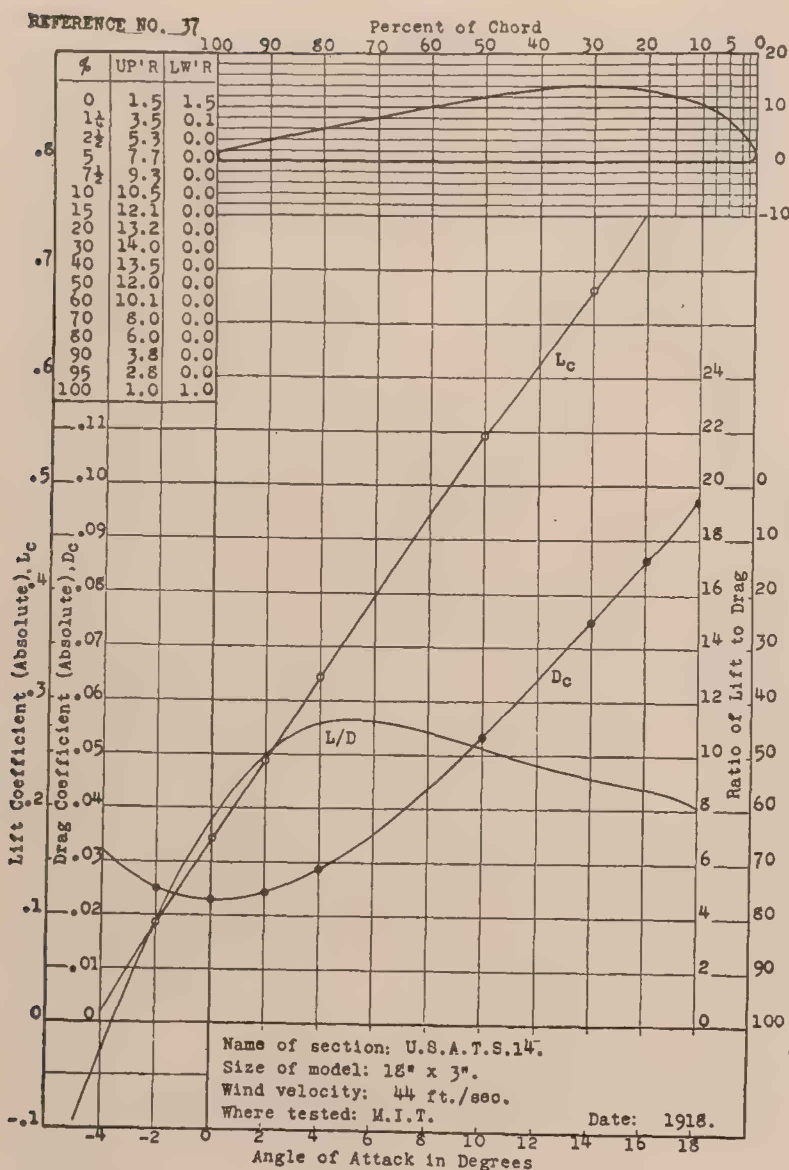
REFERENCE NO. 36



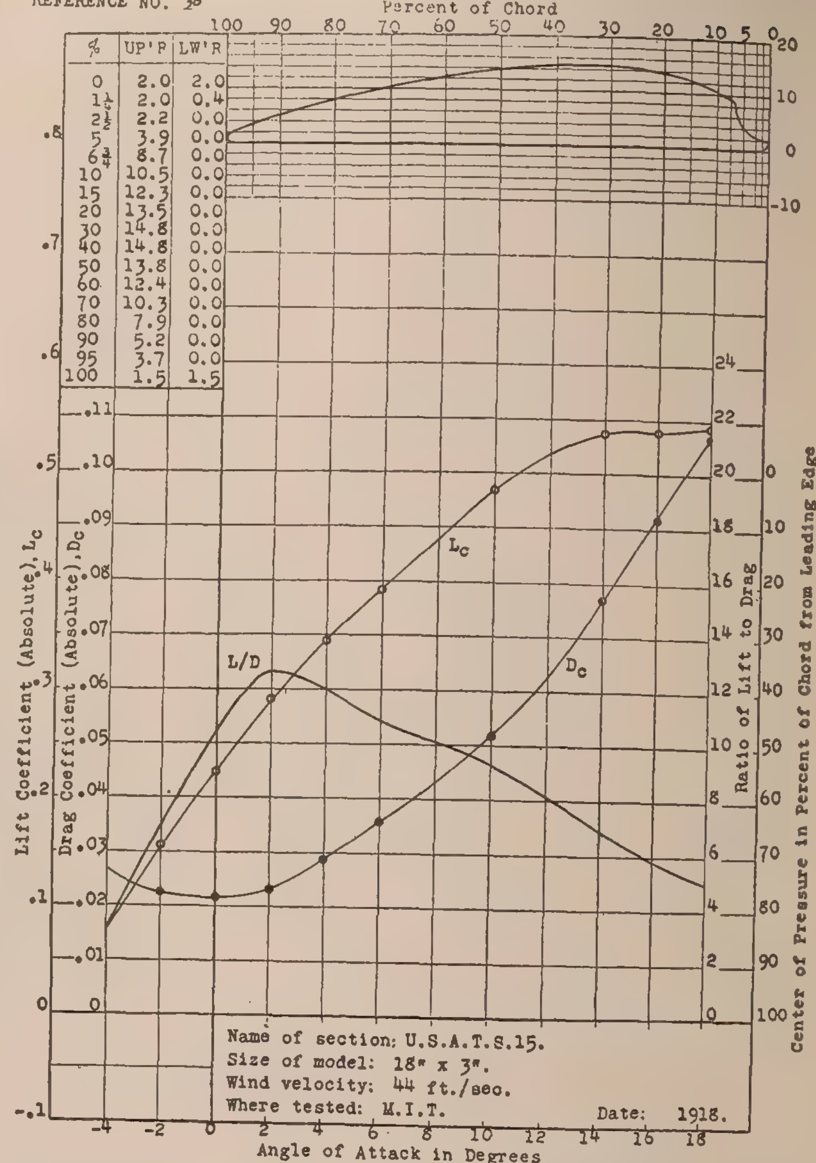
%	UP'R	LW'R
0	1.4	1.4
1¼	3.5	0.1
2½	5.4	0.0
5	8.0	0.0
7½	9.7	0.0
10	11.0	0.0
15	12.9	0.0
20	13.8	0.0
30	14.3	0.0
40	13.7	0.0
50	12.3	0.0
60	10.8	0.0
70	8.9	0.0
80	6.8	0.0
90	4.4	0.0
95	3.0	0.0
100	1.0	1.0

Name of section: U.S.A.T.S.13.
Size of model: 18" x 3".
Wind velocity: 44 ft./sec.
Where tested: M.I.T. Date: 1918.

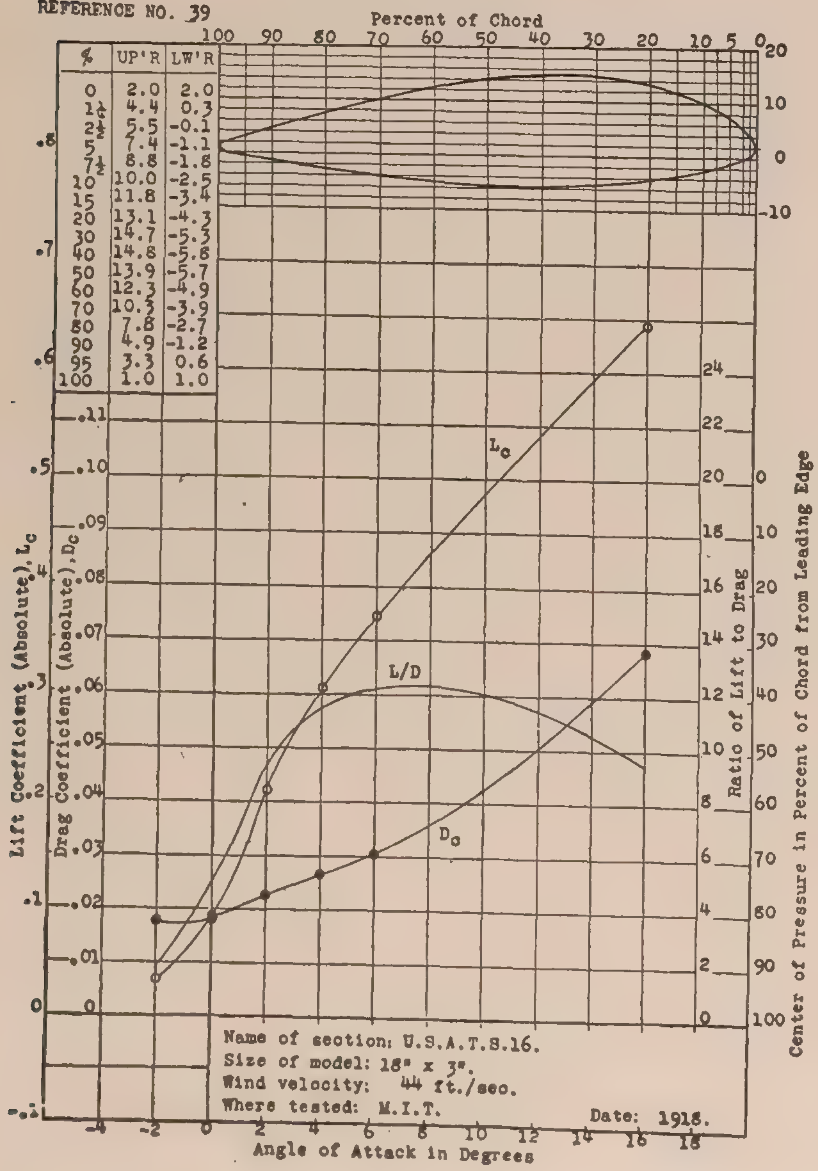
REFERENCE NO. 37



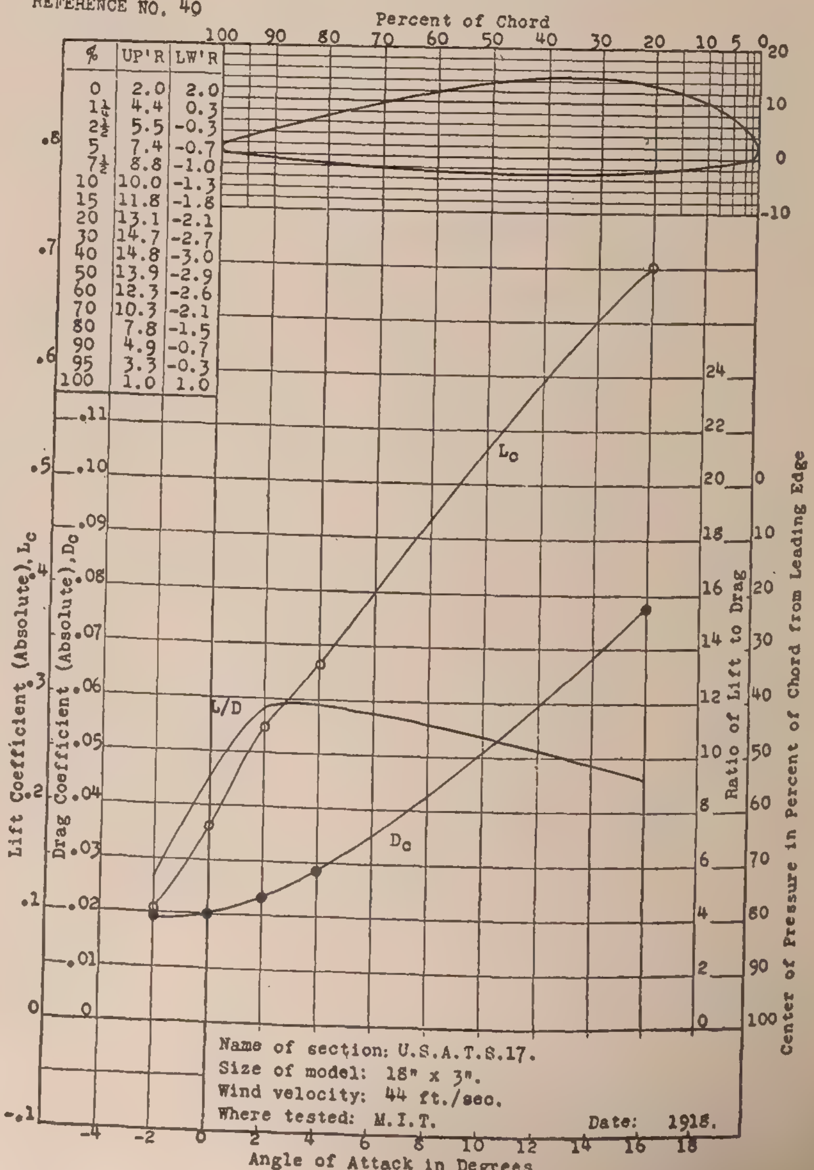
REFERENCE NO. 38



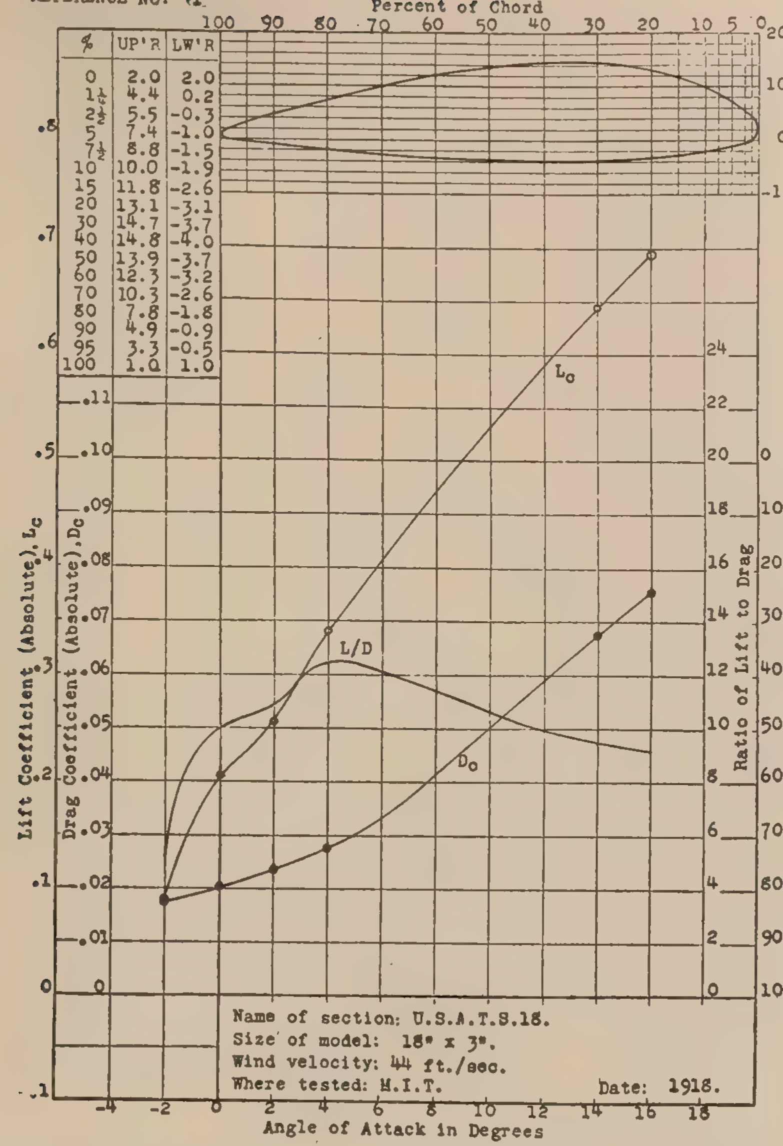
REFERENCE NO. 39



REFERENCE NO. 40

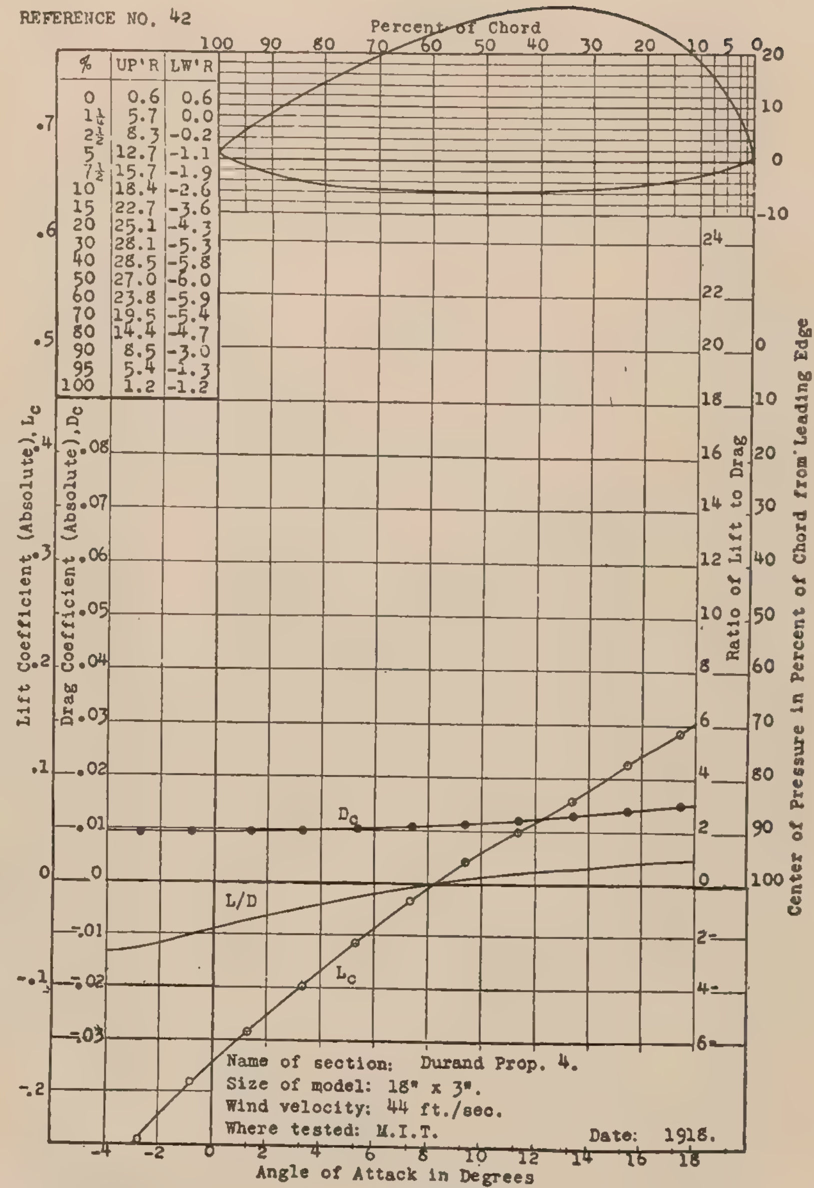


REFERENCE NO. 41



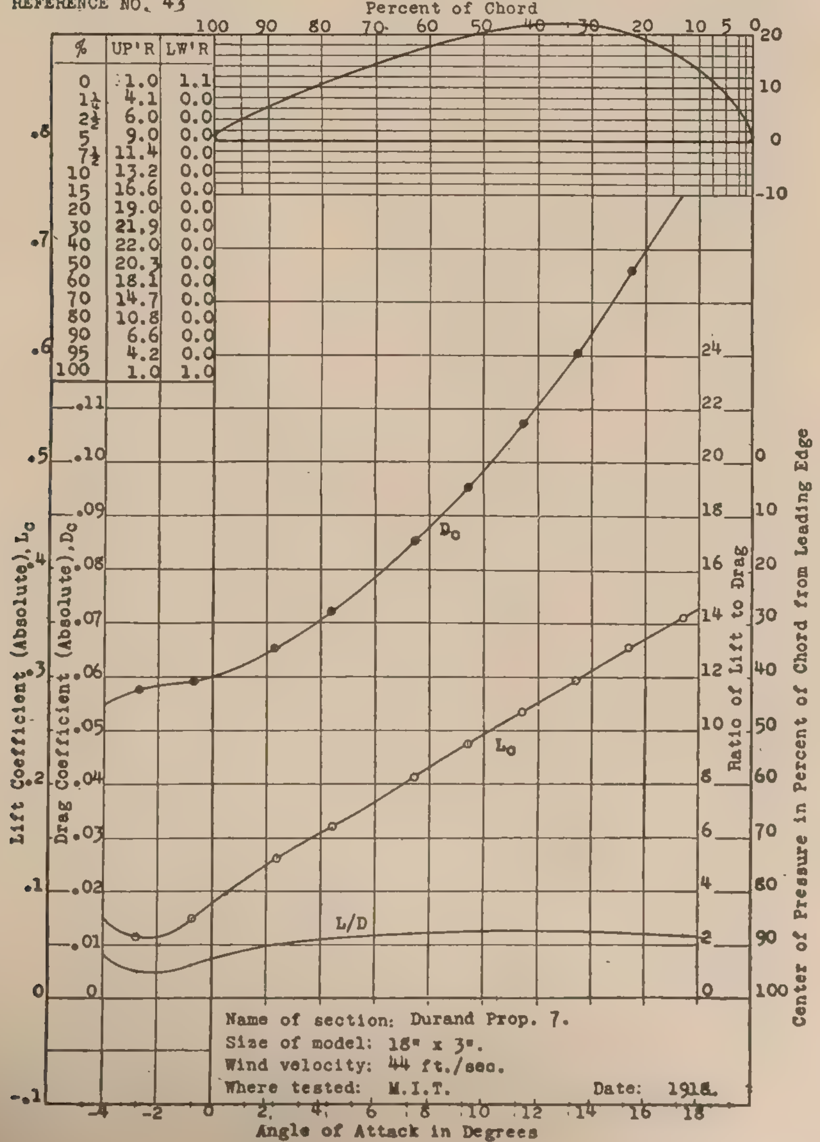
%	UP'R	LW'R
0	2.0	2.0
1¼	4.4	0.2
2½	5.5	-0.3
5	7.4	-1.0
7½	8.8	-1.5
10	10.0	-1.9
15	11.8	-2.6
20	13.1	-3.1
30	14.7	-3.7
40	14.8	-4.0
50	13.9	-3.7
60	12.3	-3.2
70	10.3	-2.6
80	7.8	-1.8
90	4.9	-0.9
95	3.3	-0.5
100	1.0	1.0

REFERENCE NO. 42



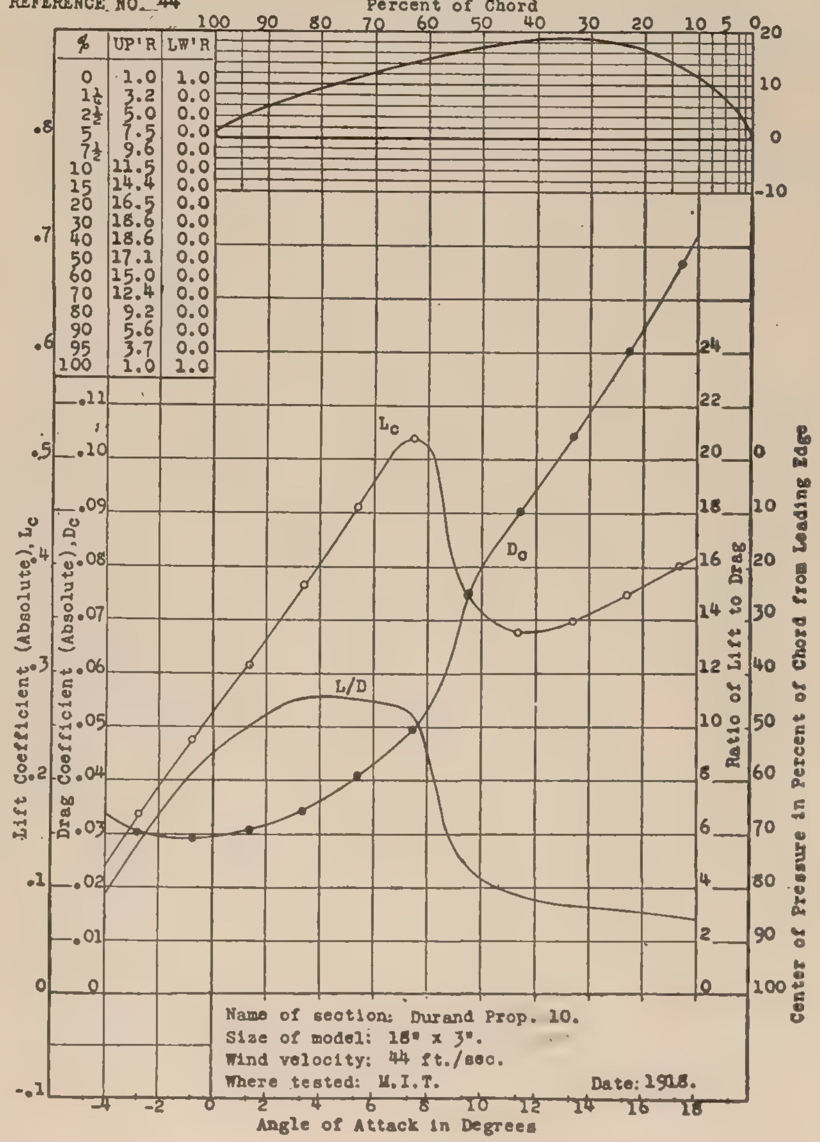
%	UP'R	LW'R
0	0.6	0.6
1¼	5.7	0.0
2½	8.3	-0.2
5	12.7	-1.1
7½	15.7	-1.9
10	18.4	-2.6
15	22.7	-3.6
20	25.1	-4.3
30	28.1	-5.3
40	28.5	-5.8
50	27.0	-6.0
60	23.8	-5.9
70	19.5	-5.4
80	14.4	-4.7
90	8.5	-3.0
95	5.4	-1.3
100	1.2	-1.2

REFERENCE NO. 43



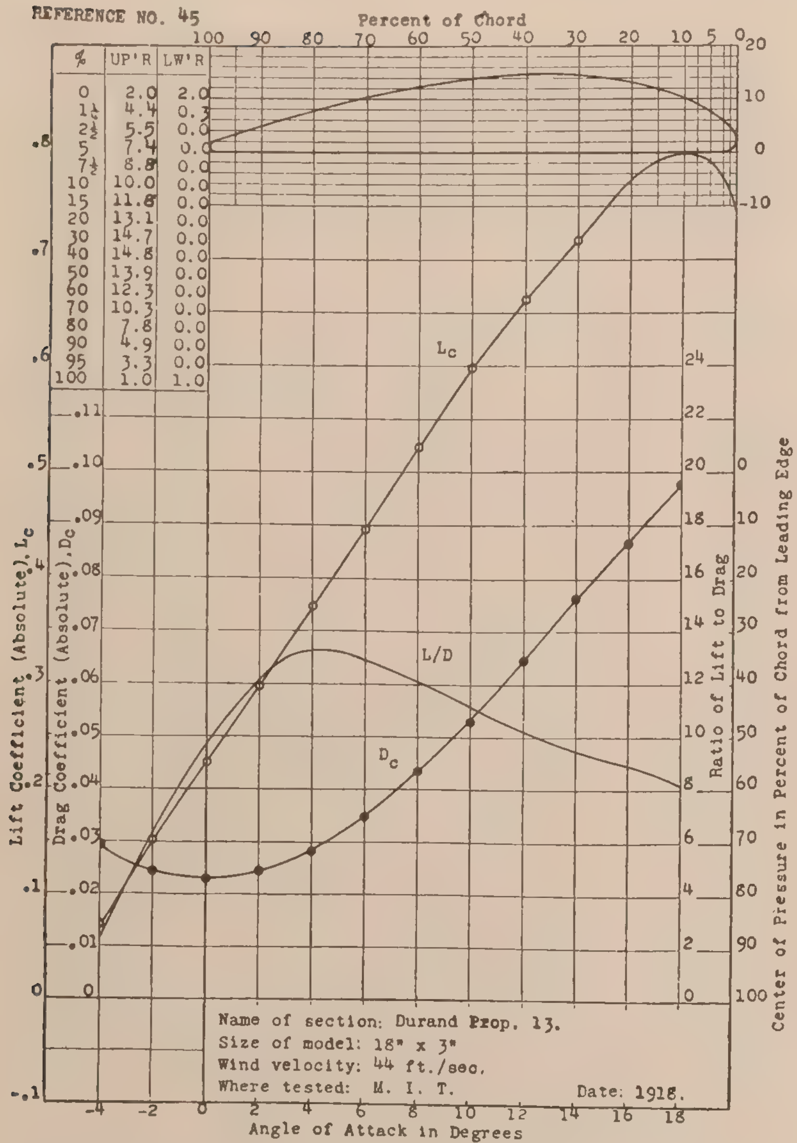
%	UP'R	LW'R
0	1.0	1.1
1¼	4.1	0.0
2½	6.0	0.0
5	9.0	0.0
7½	11.4	0.0
10	13.2	0.0
15	16.6	0.0
20	19.0	0.0
30	21.9	0.0
40	22.0	0.0
50	20.3	0.0
60	18.1	0.0
70	14.7	0.0
80	10.8	0.0
90	6.6	0.0
95	4.2	0.0
100	1.0	1.0

REFERENCE NO. 44



%	UP'R	LW'R
0	1.0	1.0
1¼	3.2	0.0
2½	5.0	0.0
5	7.5	0.0
7½	9.6	0.0
10	11.5	0.0
15	14.4	0.0
20	16.5	0.0
30	18.6	0.0
40	18.6	0.0
50	17.1	0.0
60	15.0	0.0
70	12.4	0.0
80	9.2	0.0
90	5.6	0.0
95	3.7	0.0
100	1.0	1.0

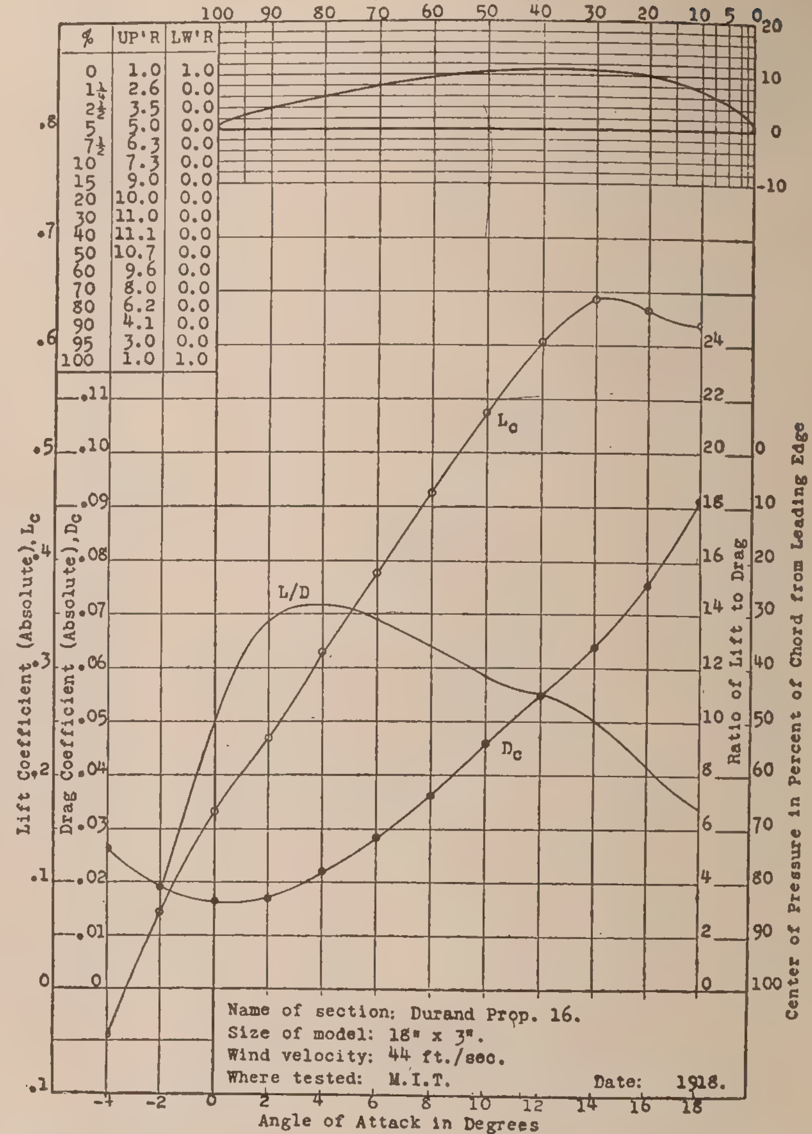
REFERENCE NO. 45



%	UP'R	LW'R
0	2.0	2.0
1¼	4.4	0.3
2½	5.5	0.0
5	7.4	0.0
7½	8.8	0.0
10	10.0	0.0
15	11.8	0.0
20	13.1	0.0
30	14.7	0.0
40	14.8	0.0
50	13.9	0.0
60	12.3	0.0
70	10.3	0.0
80	7.8	0.0
90	4.9	0.0
95	3.3	0.0
100	1.0	1.0

Name of section: Durand Prop. 13.
Size of model: 18" x 3"
Wind velocity: 44 ft./sec.
Where tested: M. I. T.
Date: 1918.

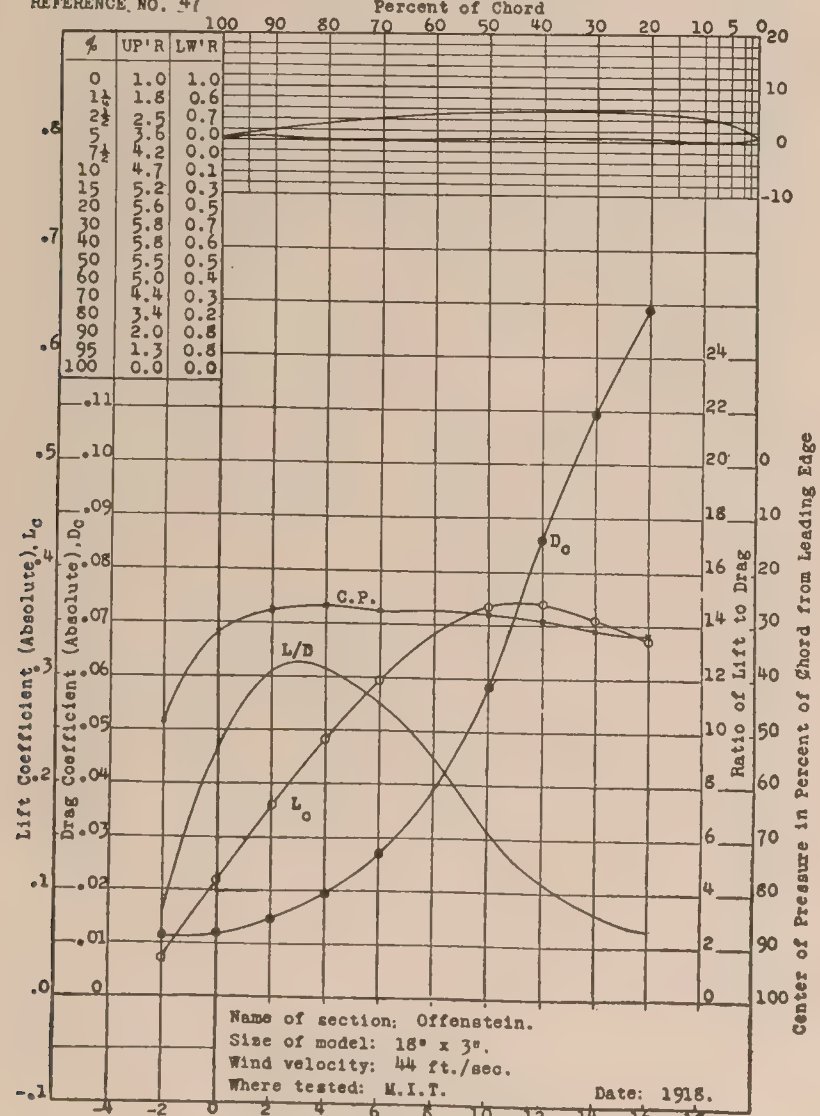
REFERENCE NO. 46



%	UP'R	LW'R
0	1.0	1.0
1¼	2.6	0.0
2½	3.5	0.0
5	5.0	0.0
7½	6.3	0.0
10	7.3	0.0
15	9.0	0.0
20	10.0	0.0
30	11.0	0.0
40	11.1	0.0
50	10.7	0.0
60	9.6	0.0
70	8.0	0.0
80	6.2	0.0
90	4.1	0.0
95	3.0	0.0
100	1.0	1.0

Name of section: Durand Prop. 16.
Size of model: 18" x 3".
Wind velocity: 44 ft./sec.
Where tested: M.I.T.
Date: 1918.

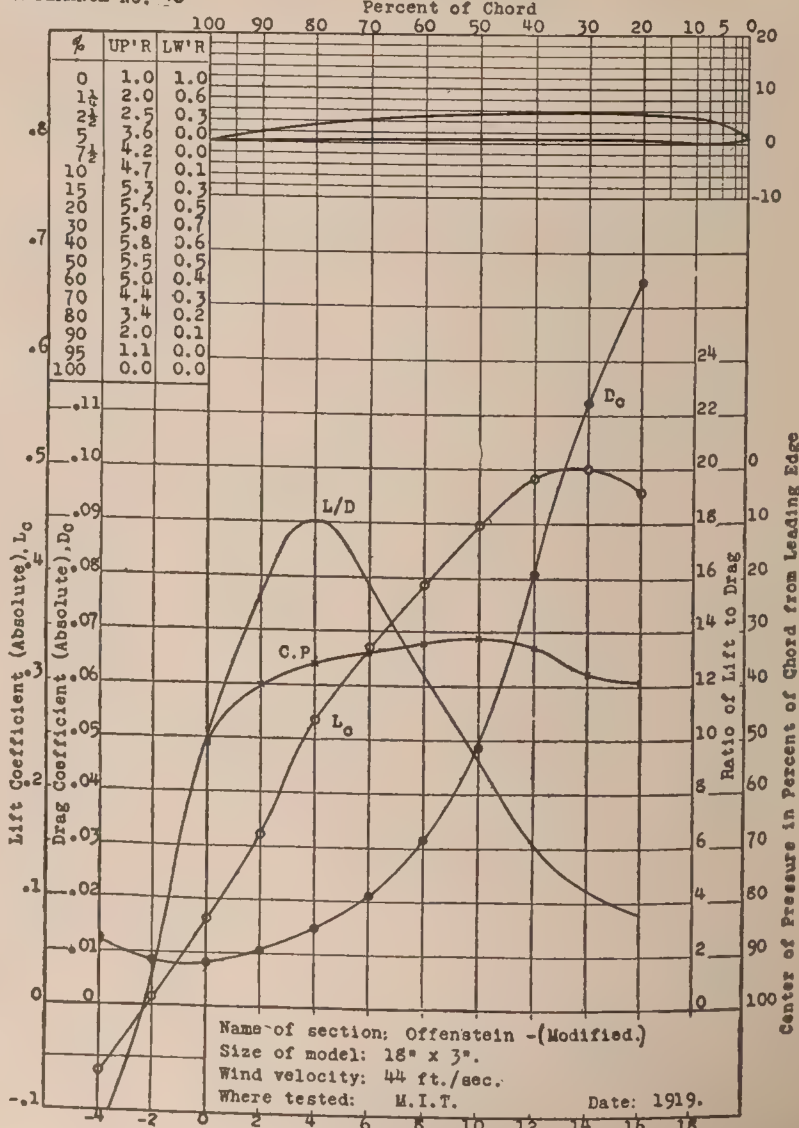
REFERENCE NO. 47



%	UP'R	LW'R
0	1.0	1.0
1¼	1.8	0.6
2½	2.5	0.7
5	3.6	0.0
7½	4.2	0.0
10	4.7	0.1
15	5.2	0.3
20	5.6	0.5
30	5.8	0.7
40	5.8	0.6
50	5.5	0.5
60	5.0	0.4
70	4.4	0.3
80	3.4	0.2
90	2.0	0.8
95	1.3	0.8
100	0.0	0.0

Name of section: Offenstein.
Size of model: 18" x 3".
Wind velocity: 44 ft./sec.
Where tested: M.I.T.
Date: 1918.

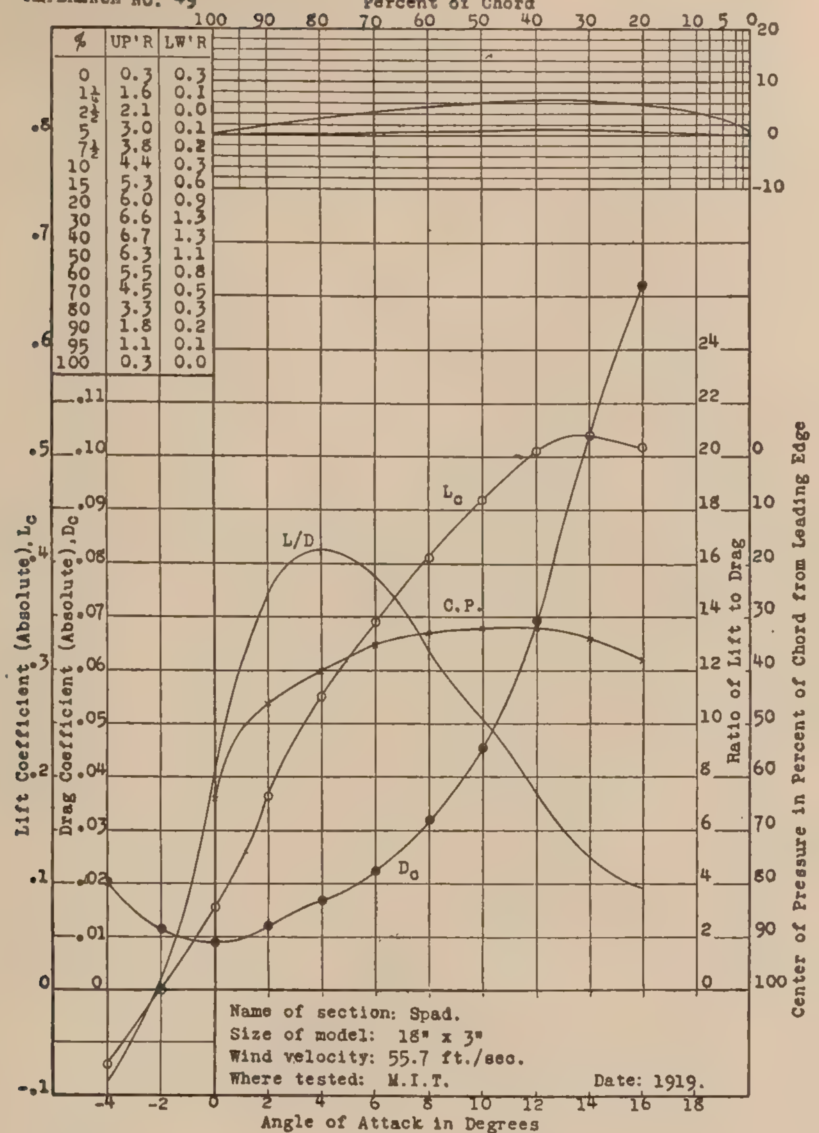
REFERENCE NO. 48



%	UP'R	LW'R
0	1.0	1.0
1¼	2.0	0.6
2½	2.5	0.3
5	3.6	0.0
7½	4.2	0.0
10	4.7	0.1
15	5.3	0.3
20	5.5	0.5
30	5.8	0.7
40	5.8	0.6
50	5.5	0.5
60	5.0	0.4
70	4.4	0.3
80	3.4	0.2
90	2.0	0.1
95	1.1	0.0
100	0.0	0.0

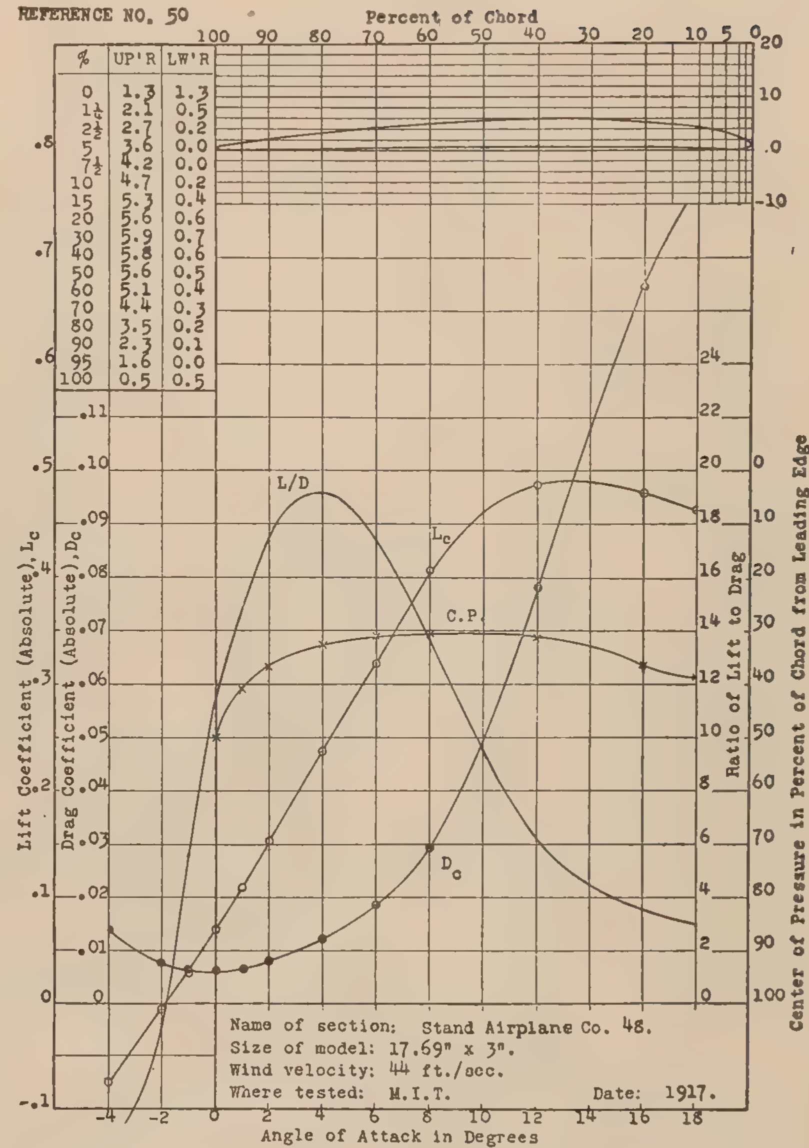
Name of section: Offenstein -(Modified.)
Size of model: 18" x 3".
Wind velocity: 44 ft./sec.
Where tested: M.I.T.
Date: 1919.

REFERENCE NO. 49



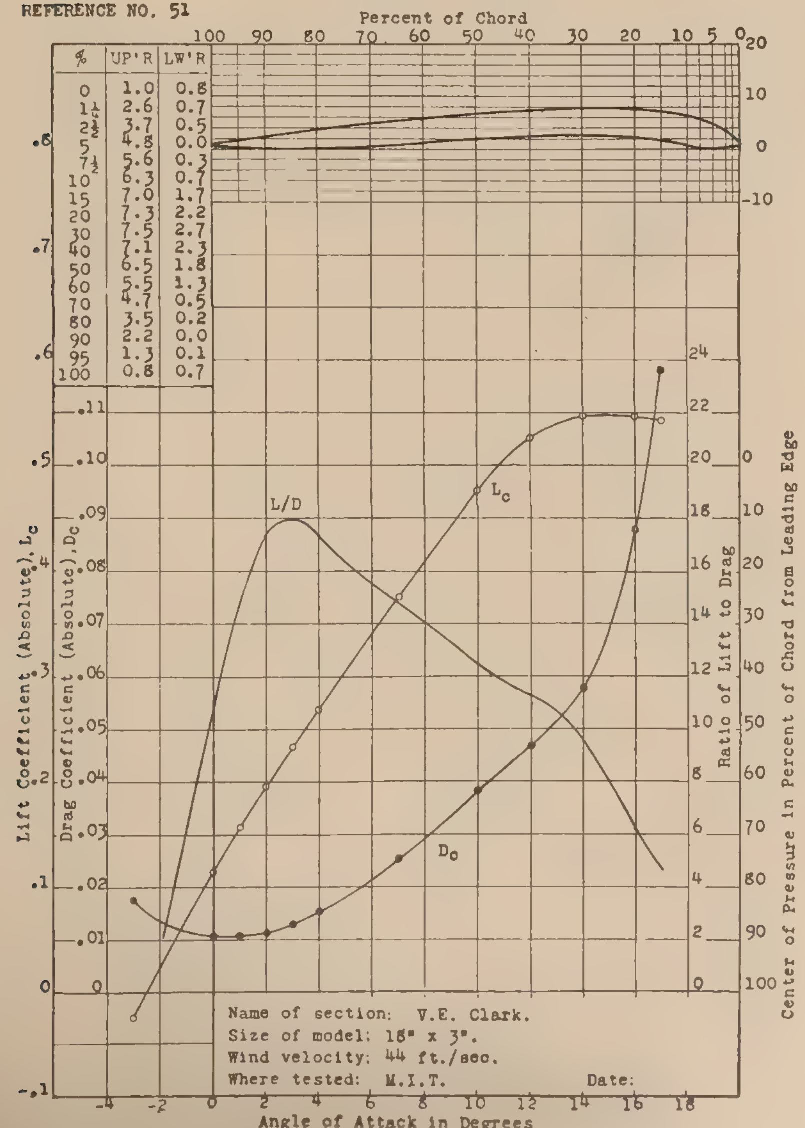
%	UP'R	LW'R
0	0.3	0.3
1 1/4	1.6	0.1
2 1/2	2.1	0.0
5	3.0	0.1
7 1/2	3.8	0.2
10	4.4	0.3
15	5.3	0.6
20	6.0	0.9
30	6.6	1.3
40	6.7	1.3
50	6.3	1.1
60	5.5	0.8
70	4.5	0.5
80	3.3	0.3
90	1.8	0.2
95	1.1	0.1
100	0.3	0.0

REFERENCE NO. 50



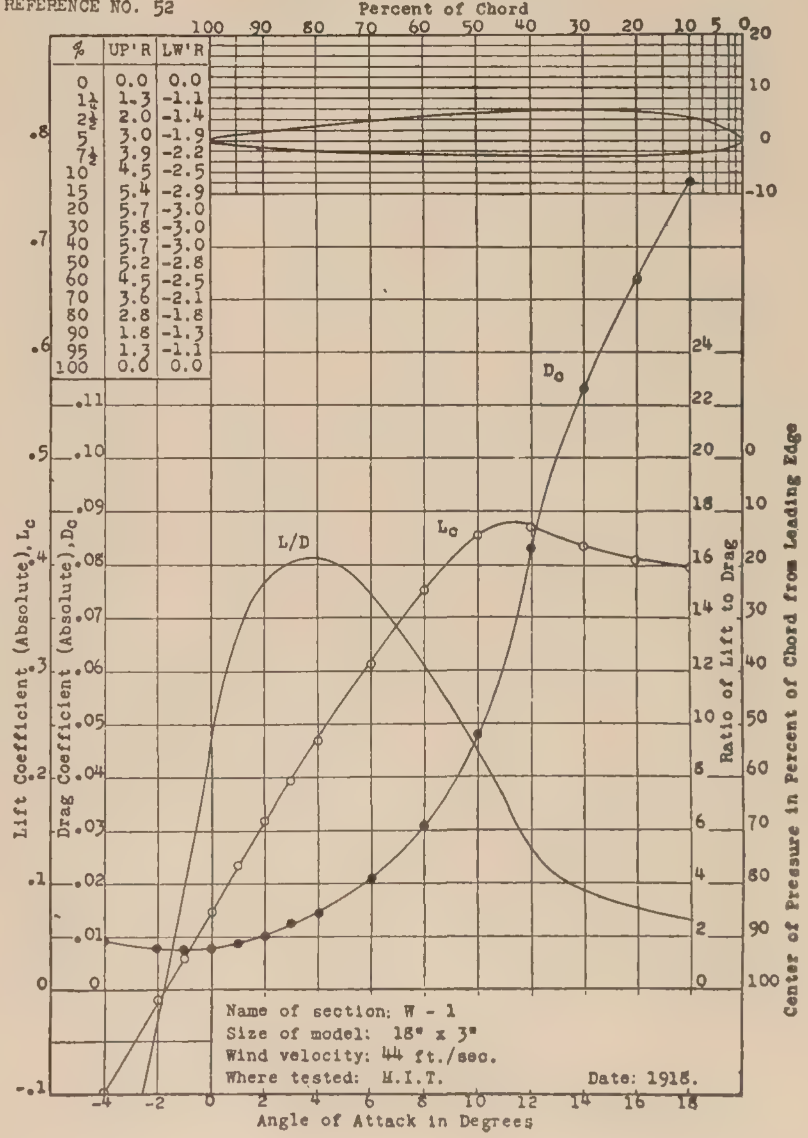
%	UP'R	LW'R
0	1.3	1.3
1 1/4	2.1	0.5
2 1/2	2.7	0.2
5	3.6	0.0
7 1/2	4.2	0.0
10	4.7	0.2
15	5.3	0.4
20	5.6	0.6
30	5.9	0.7
40	5.8	0.6
50	5.6	0.5
60	5.1	0.4
70	4.4	0.3
80	3.5	0.2
90	2.3	0.1
95	1.6	0.0
100	0.5	0.5

REFERENCE NO. 51



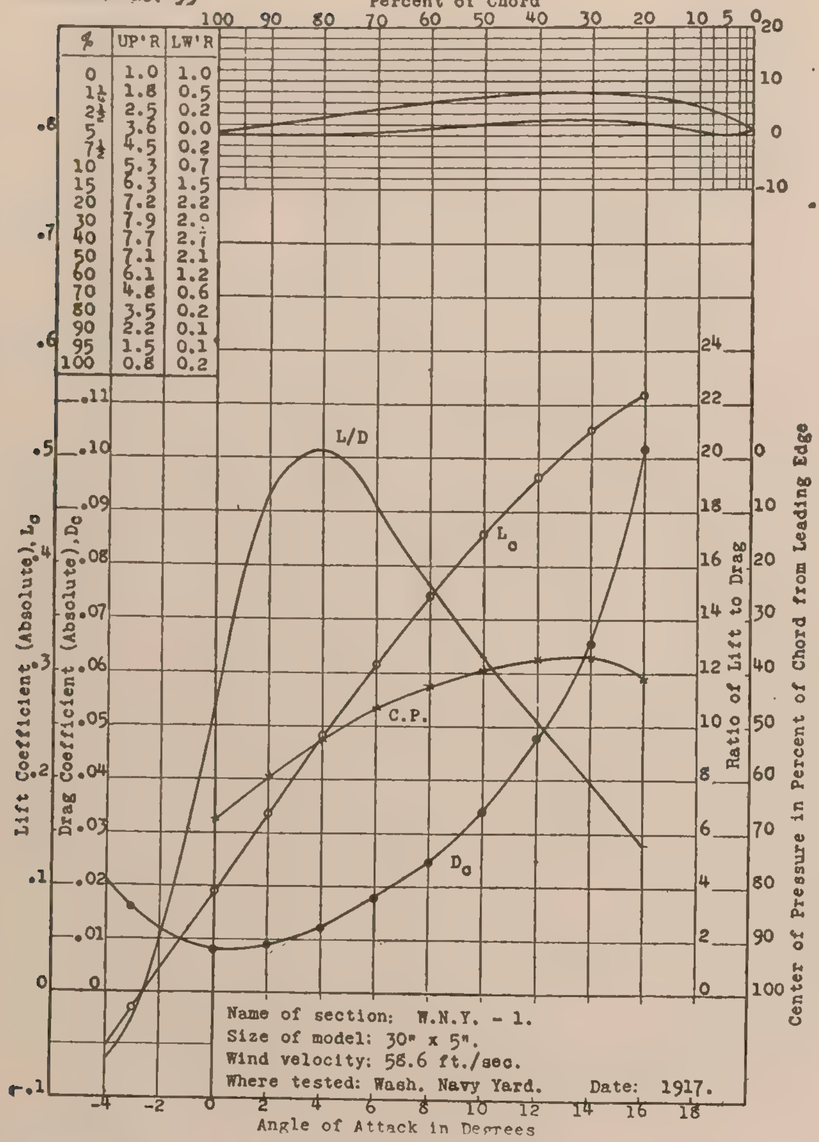
%	UP'R	LW'R
0	1.0	0.8
1 1/4	2.6	0.7
2 1/2	3.7	0.5
5	4.8	0.0
7 1/2	5.6	0.3
10	6.3	0.7
15	7.0	1.7
20	7.3	2.2
30	7.5	2.7
40	7.1	2.3
50	6.5	1.8
60	5.5	1.3
70	4.7	0.5
80	3.5	0.2
90	2.2	0.0
95	1.3	0.1
100	0.8	0.7

REFERENCE NO. 52

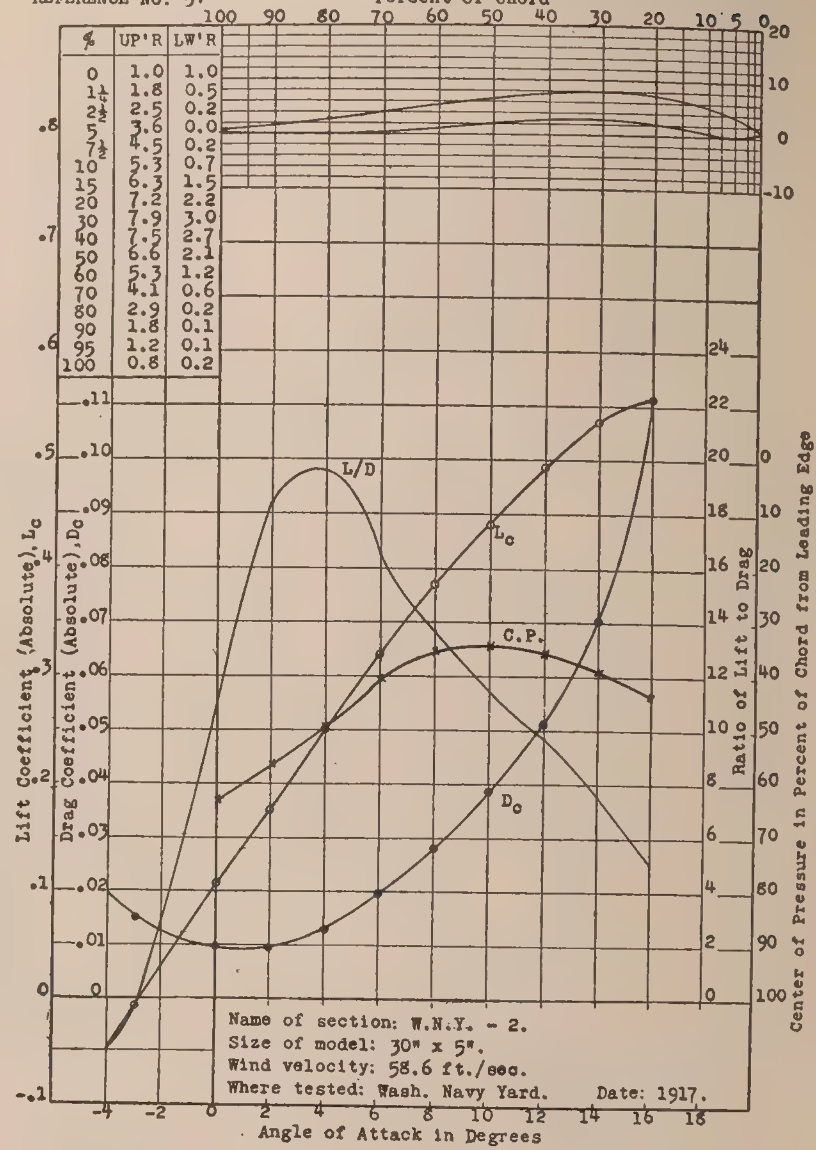


%	UP'R	LW'R
0	0.0	0.0
1 1/4	1.3	-1.1
2 1/2	2.0	-1.4
5	3.0	-1.9
7 1/2	3.9	-2.2
10	4.5	-2.5
15	5.4	-2.9
20	5.7	-3.0
30	5.8	-3.0
40	5.7	-3.0
50	5.2	-2.8
60	4.5	-2.5
70	3.6	-2.1
80	2.8	-1.8
90	1.8	-1.3
95	1.3	-1.1
100	0.0	0.0

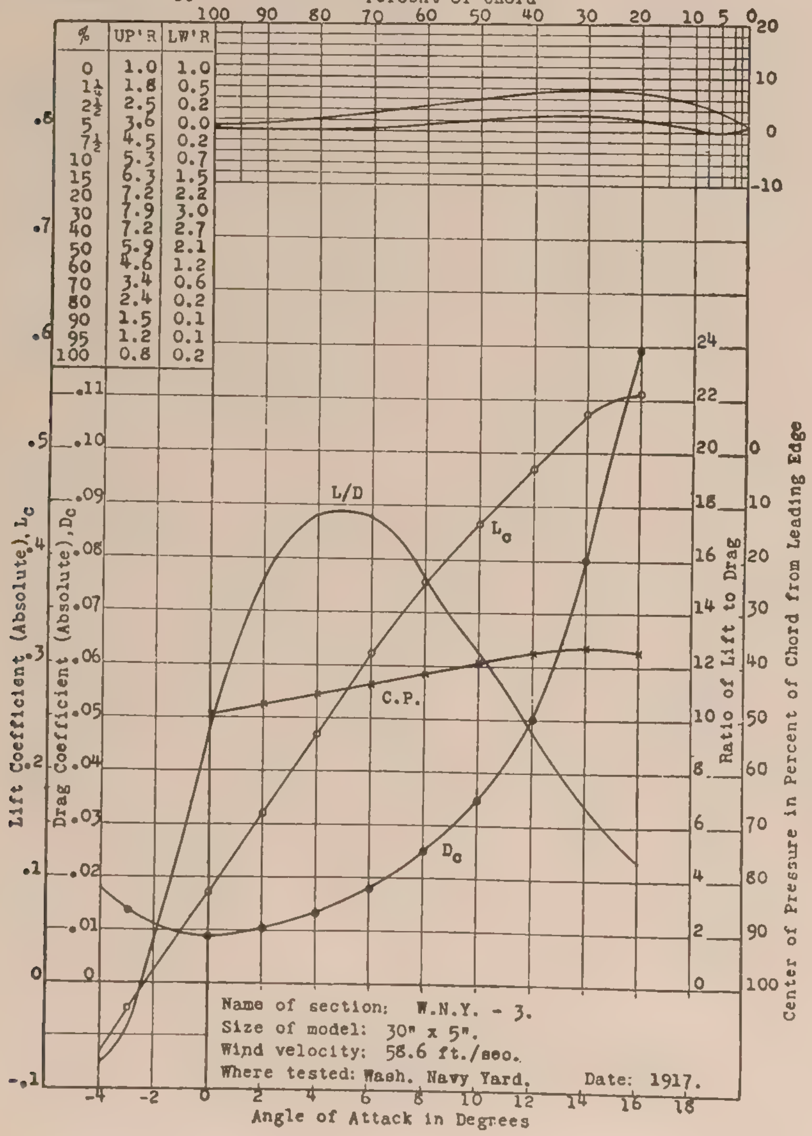
REFERENCE NO. 53



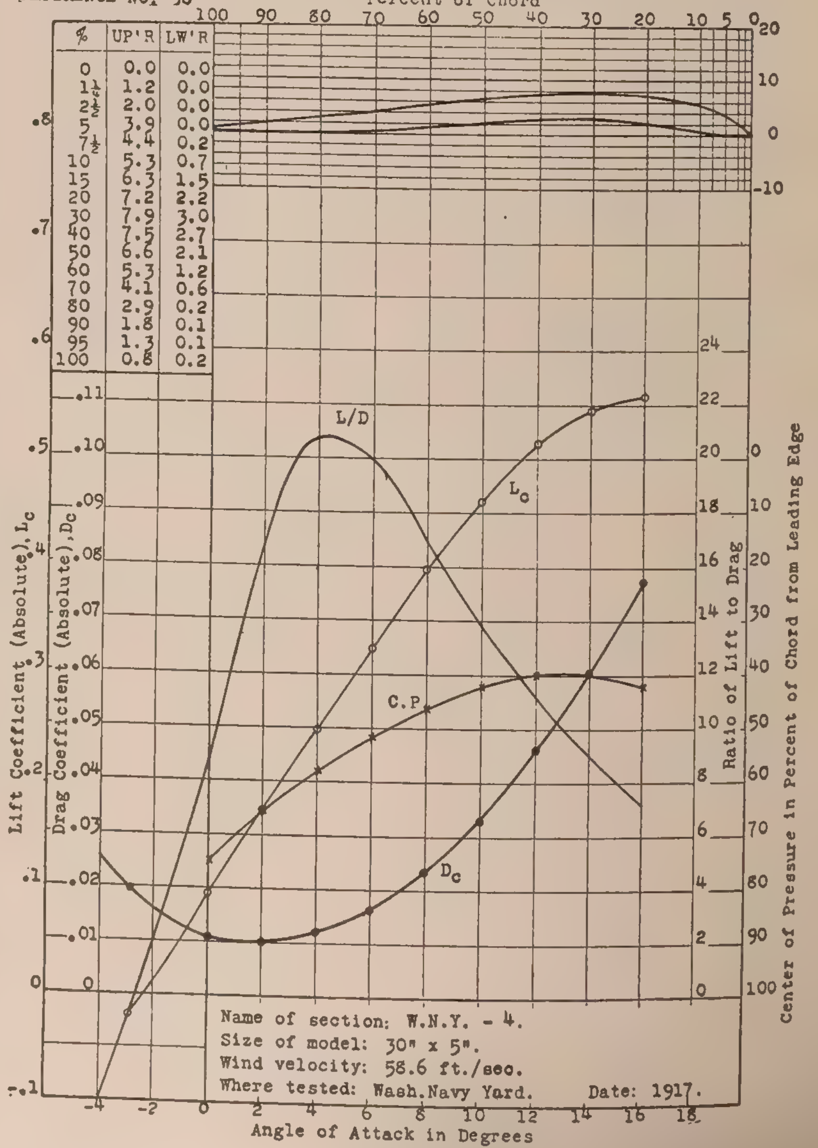
REFERENCE NO. 54



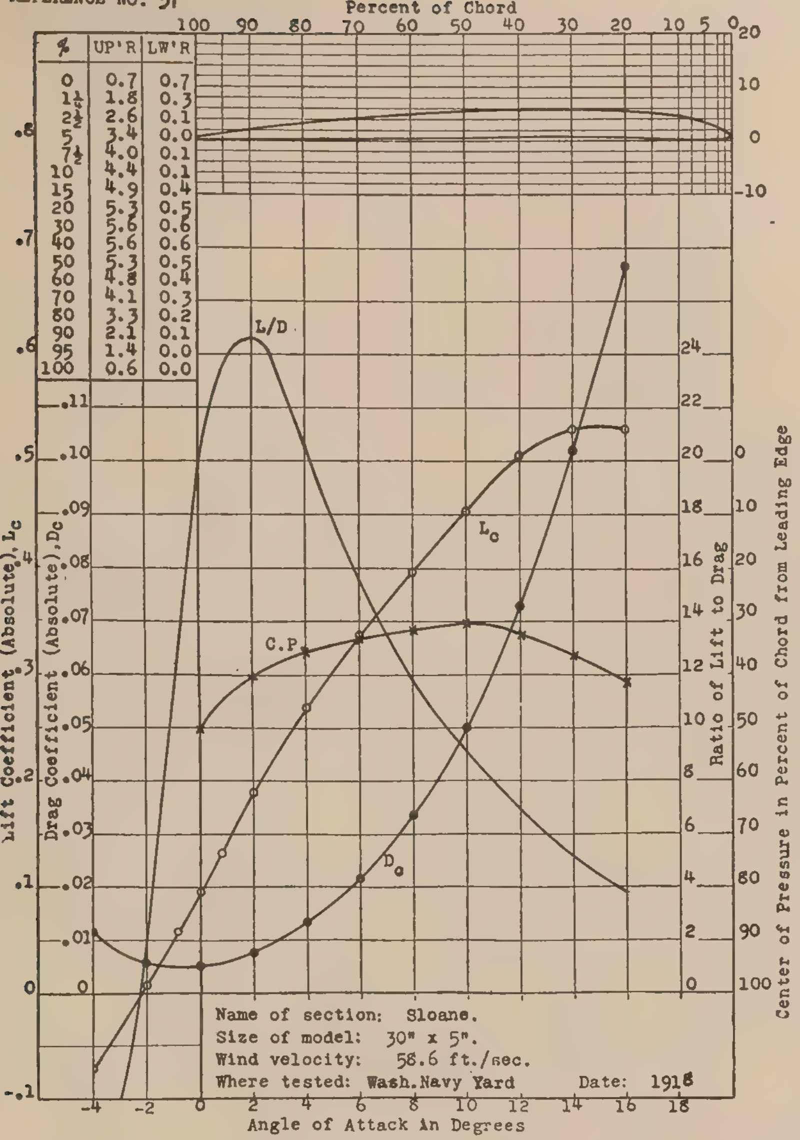
REFERENCE NO. 55



REFERENCE NO. 56

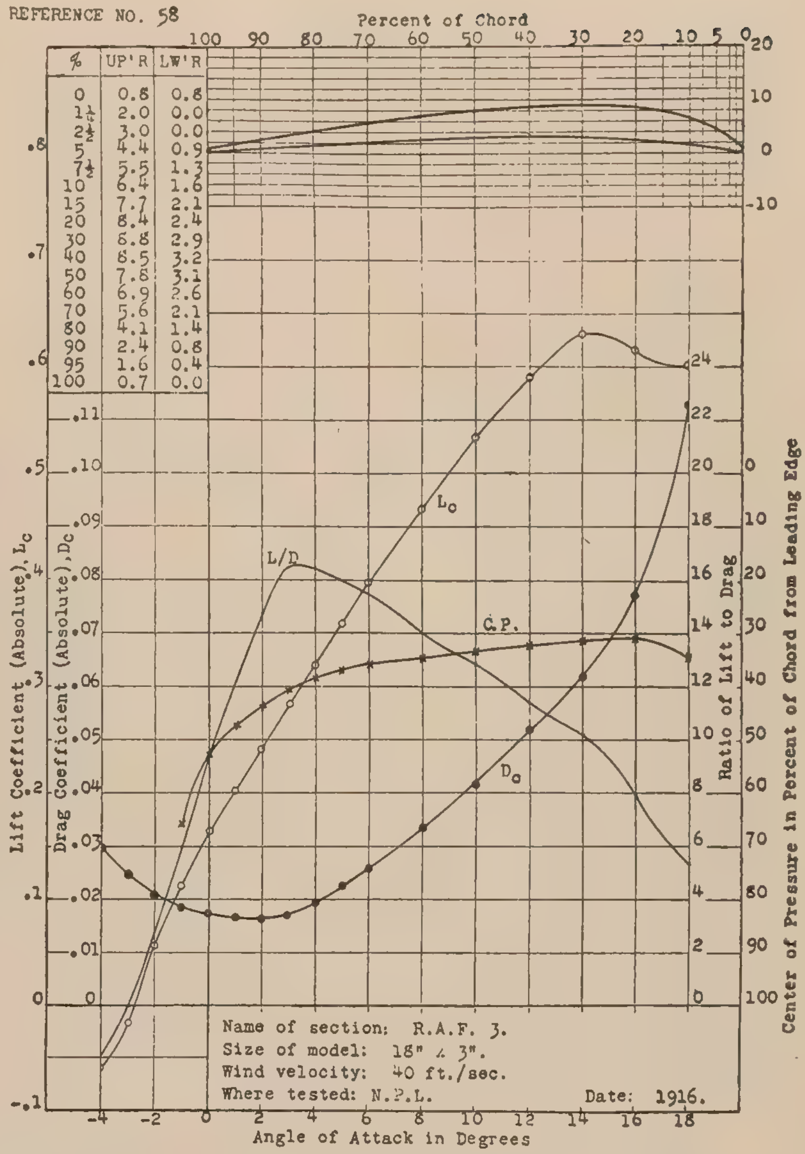


REFERENCE NO. 57



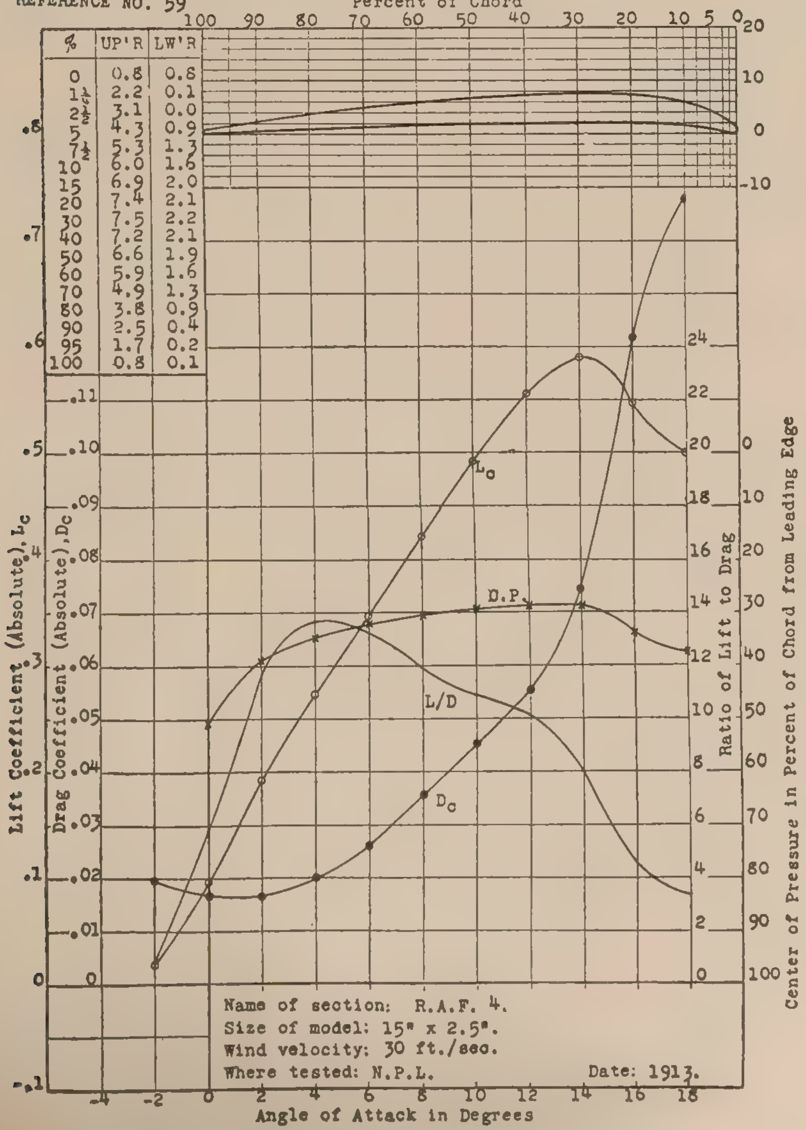
%	UP'R	LW'R
0	0.7	0.7
1½	1.8	0.3
2½	2.6	0.1
5	3.4	0.0
7½	4.0	0.1
10	4.4	0.1
15	4.9	0.4
20	5.3	0.5
30	5.6	0.6
40	5.6	0.6
50	5.3	0.5
60	4.8	0.4
70	4.1	0.3
80	3.3	0.2
90	2.1	0.1
95	1.4	0.0
100	0.6	0.0

REFERENCE NO. 58



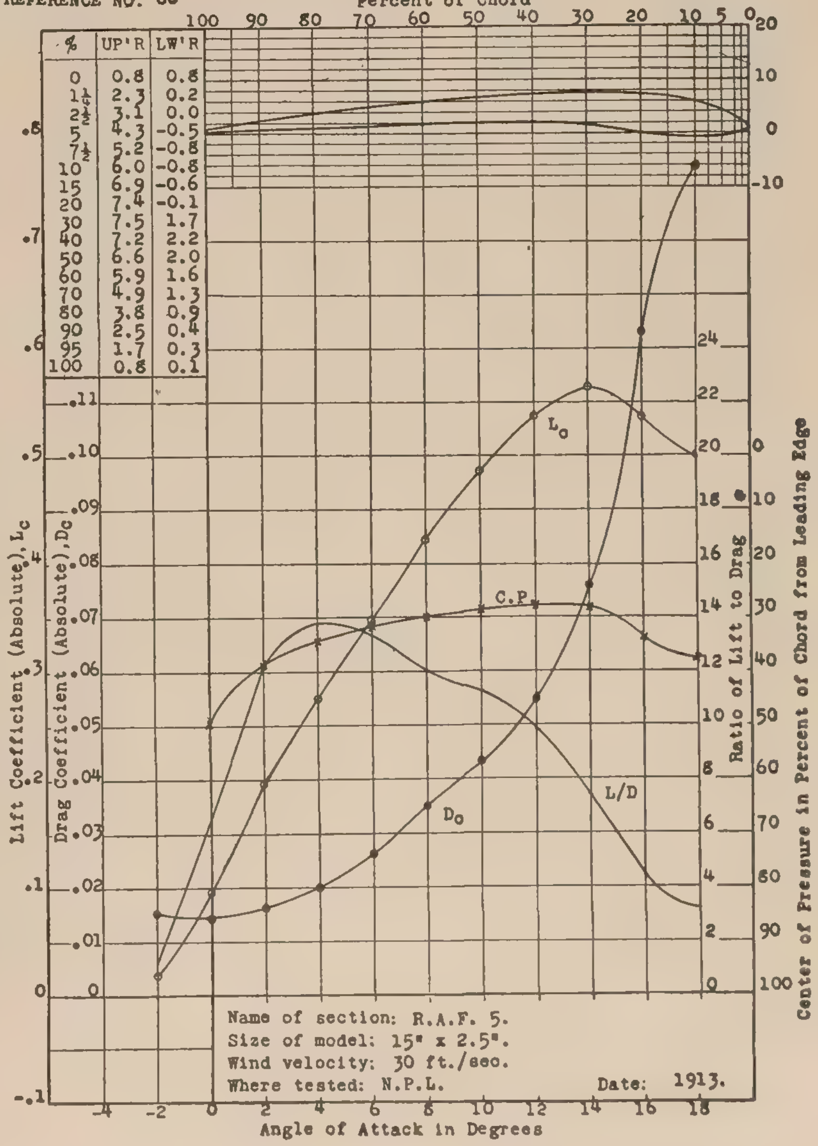
%	UP'R	LW'R
0	0.8	0.8
1½	2.0	0.0
2½	3.0	0.0
5	4.4	0.9
7½	5.5	1.3
10	6.4	1.6
15	7.7	2.1
20	8.4	2.4
30	8.8	2.9
40	8.5	3.2
50	7.8	3.1
60	6.9	2.6
70	5.6	2.1
80	4.1	1.4
90	2.4	0.8
95	1.6	0.4
100	0.7	0.0

REFERENCE NO. 59



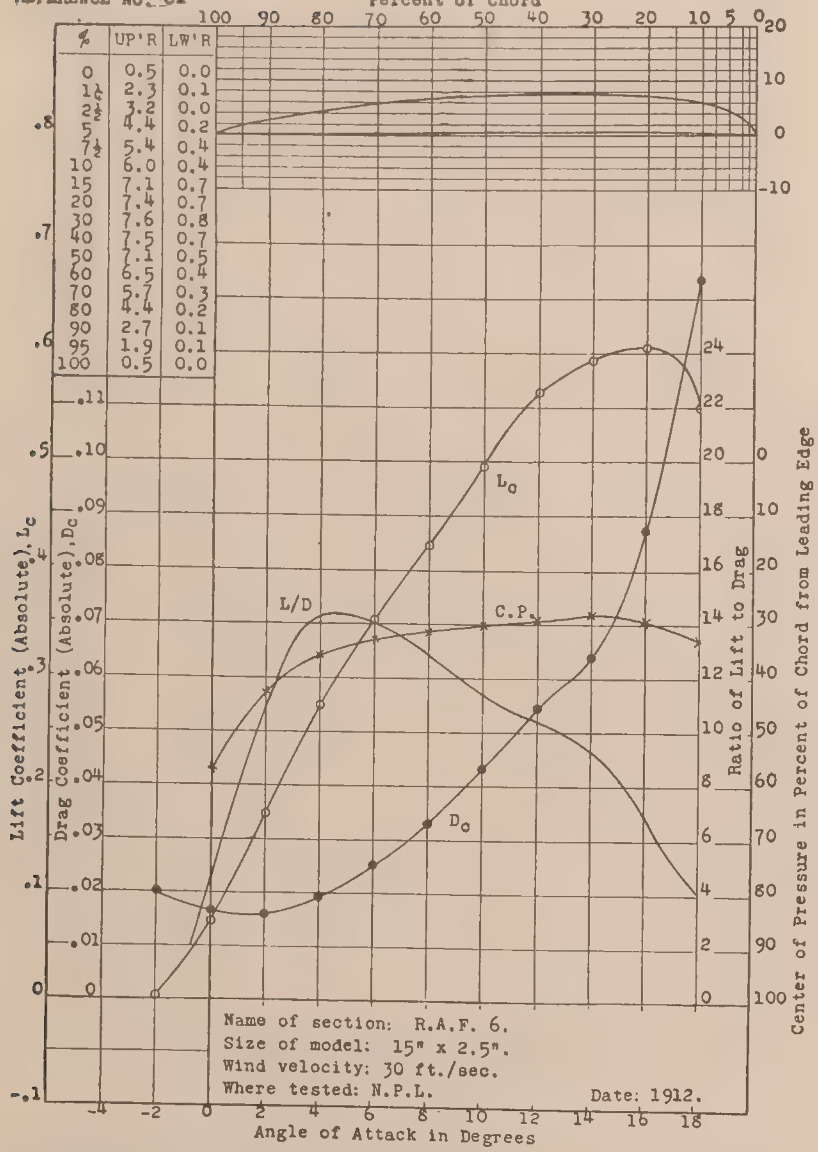
%	UP'R	LW'R
0	0.8	0.8
1½	2.2	0.1
2½	3.1	0.0
5	4.3	0.9
7½	5.3	1.3
10	6.0	1.6
15	6.9	2.0
20	7.4	2.1
30	7.5	2.2
40	7.2	2.1
50	6.6	1.9
60	5.9	1.6
70	4.9	1.3
80	3.8	0.9
90	2.5	0.4
95	1.7	0.2
100	0.8	0.1

REFERENCE NO. 60



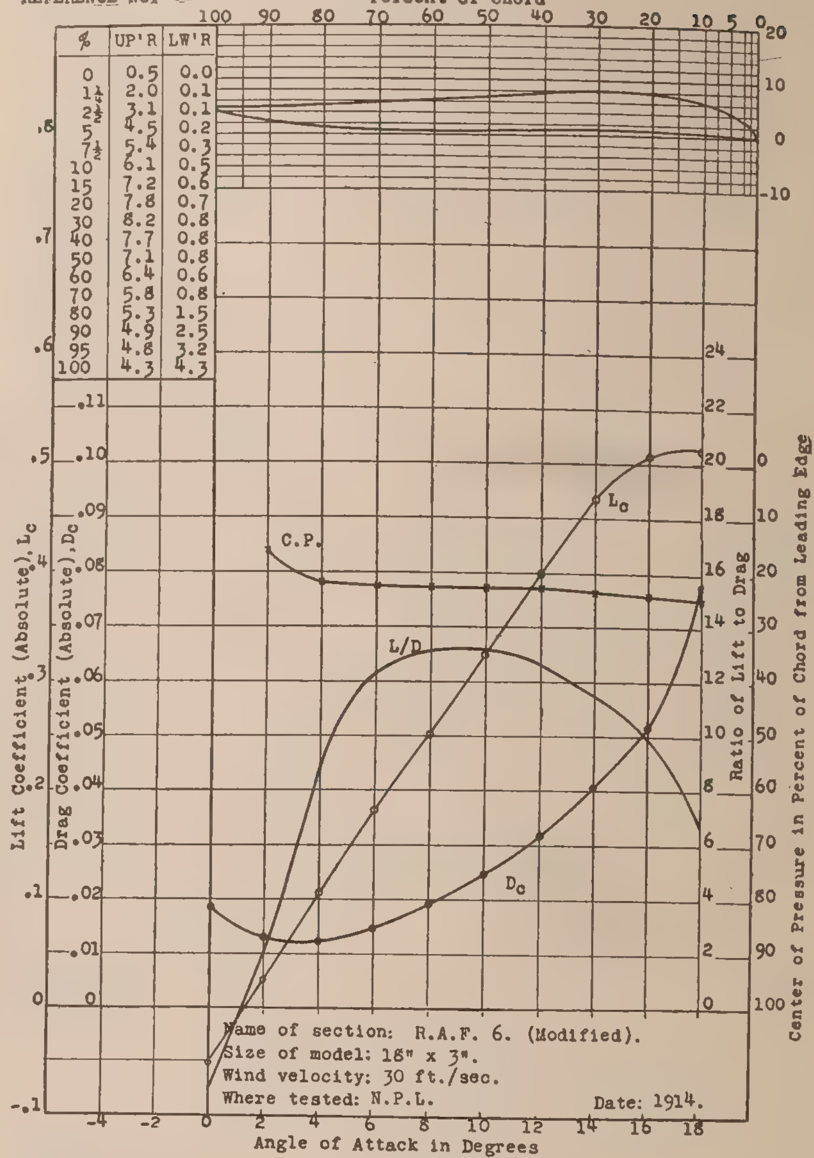
%	UP'R	LW'R
0	0.8	0.8
1½	2.3	0.2
2½	3.1	0.0
5	4.3	-0.5
7½	5.2	-0.8
10	6.0	-0.8
15	6.9	-0.6
20	7.4	-0.1
30	7.5	1.7
40	7.2	2.2
50	6.6	2.0
60	5.9	1.6
70	4.9	1.3
80	3.8	0.9
90	2.5	0.4
95	1.7	0.3
100	0.8	0.1

REFERENCE NO. 61



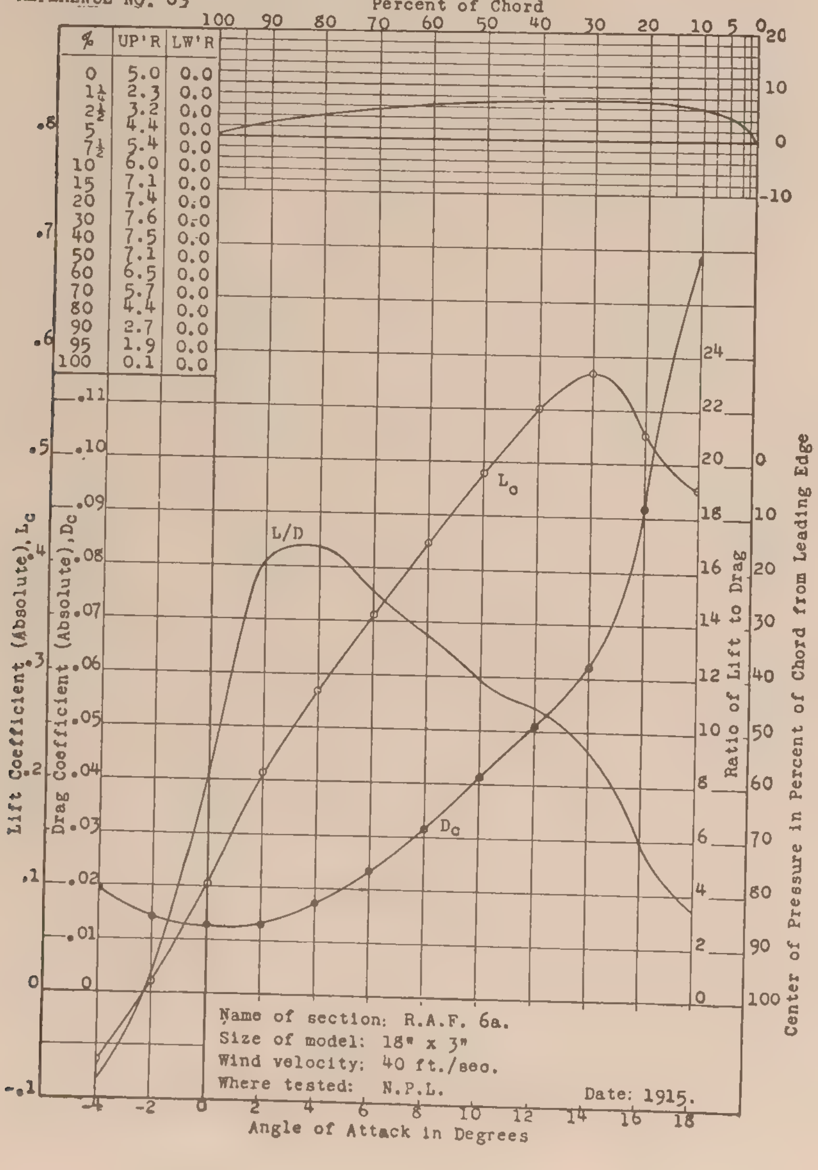
%	UP'R	LW'R
0	0.5	0.0
1¼	2.3	0.1
2½	3.2	0.0
5	4.4	0.2
7½	5.4	0.4
10	6.0	0.4
15	7.1	0.7
20	7.4	0.7
30	7.6	0.8
40	7.5	0.7
50	7.1	0.5
60	6.5	0.4
70	5.7	0.3
80	4.4	0.2
90	2.7	0.1
95	1.9	0.1
100	0.5	0.0

REFERENCE NO. 62



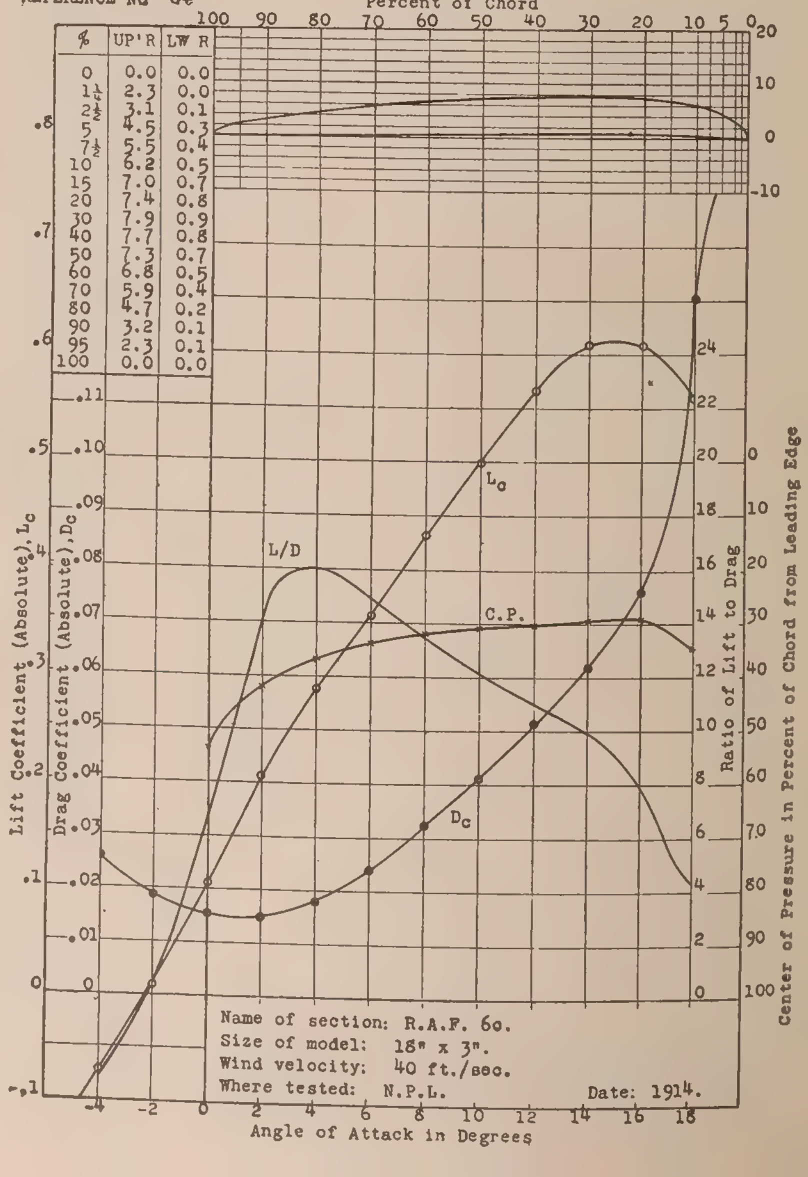
%	UP'R	LW'R
0	0.5	0.0
1¼	2.0	0.1
2½	3.1	0.1
5	4.5	0.2
7½	5.4	0.3
10	6.1	0.5
15	7.2	0.6
20	7.8	0.7
30	8.2	0.8
40	7.7	0.8
50	7.1	0.8
60	6.4	0.6
70	5.8	0.8
80	5.3	1.5
90	4.9	2.5
95	4.8	3.2
100	4.3	4.3

REFERENCE NO. 63



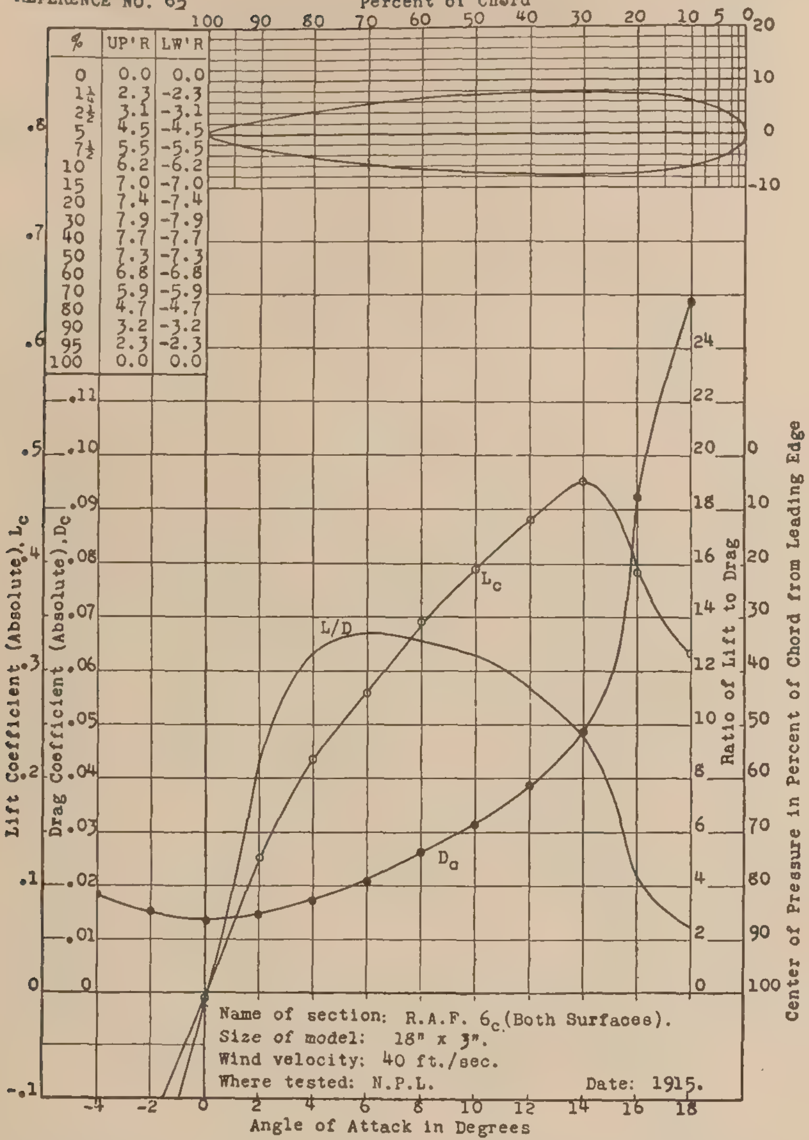
%	UP'R	LW'R
0	5.0	0.0
1¼	2.3	0.0
2½	3.2	0.0
5	4.4	0.0
7½	5.4	0.0
10	6.0	0.0
15	7.1	0.0
20	7.4	0.0
30	7.6	0.0
40	7.5	0.0
50	7.1	0.0
60	6.5	0.0
70	5.7	0.0
80	4.4	0.0
90	2.7	0.0
95	1.9	0.0
100	0.1	0.0

REFERENCE NO 64



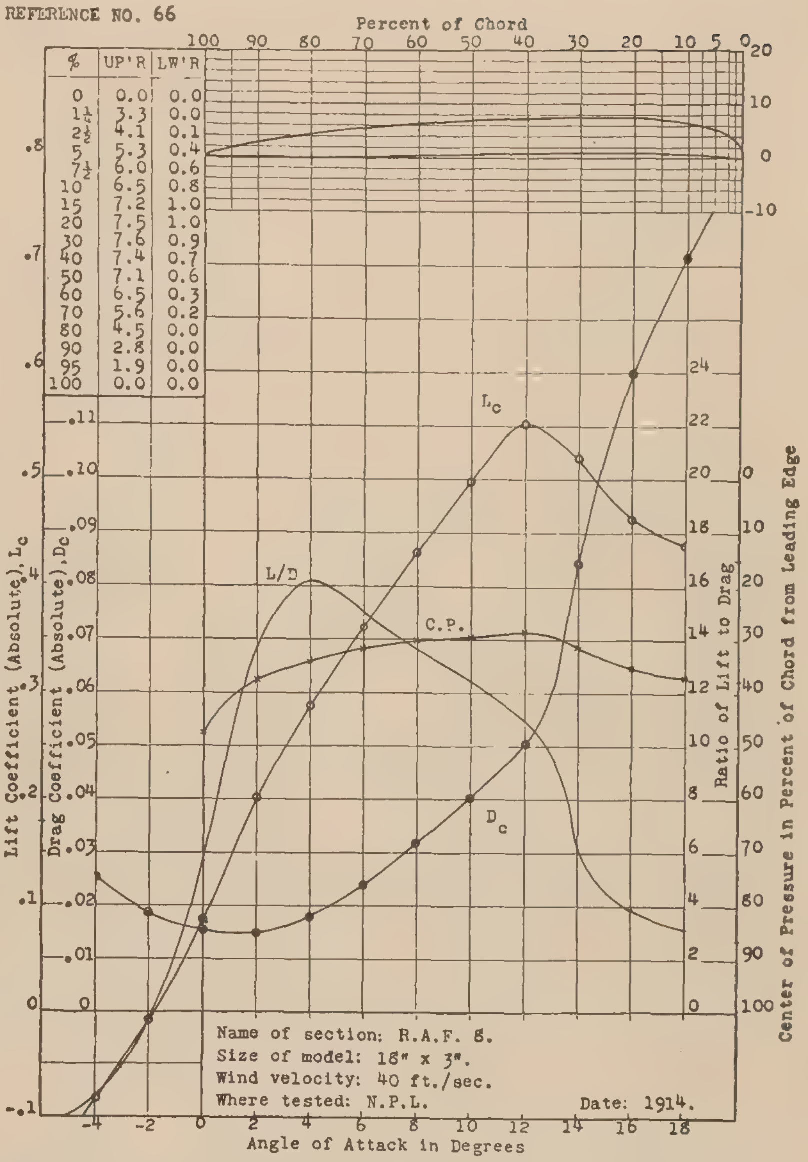
%	UP'R	LW R
0	0.0	0.0
1¼	2.3	0.0
2½	3.1	0.1
5	4.5	0.3
7½	5.5	0.4
10	6.2	0.5
15	7.0	0.7
20	7.4	0.8
30	7.9	0.9
40	7.7	0.8
50	7.3	0.7
60	6.8	0.5
70	5.9	0.4
80	4.7	0.2
90	3.2	0.1
95	2.3	0.1
100	0.0	0.0

REFERENCE NO. 65



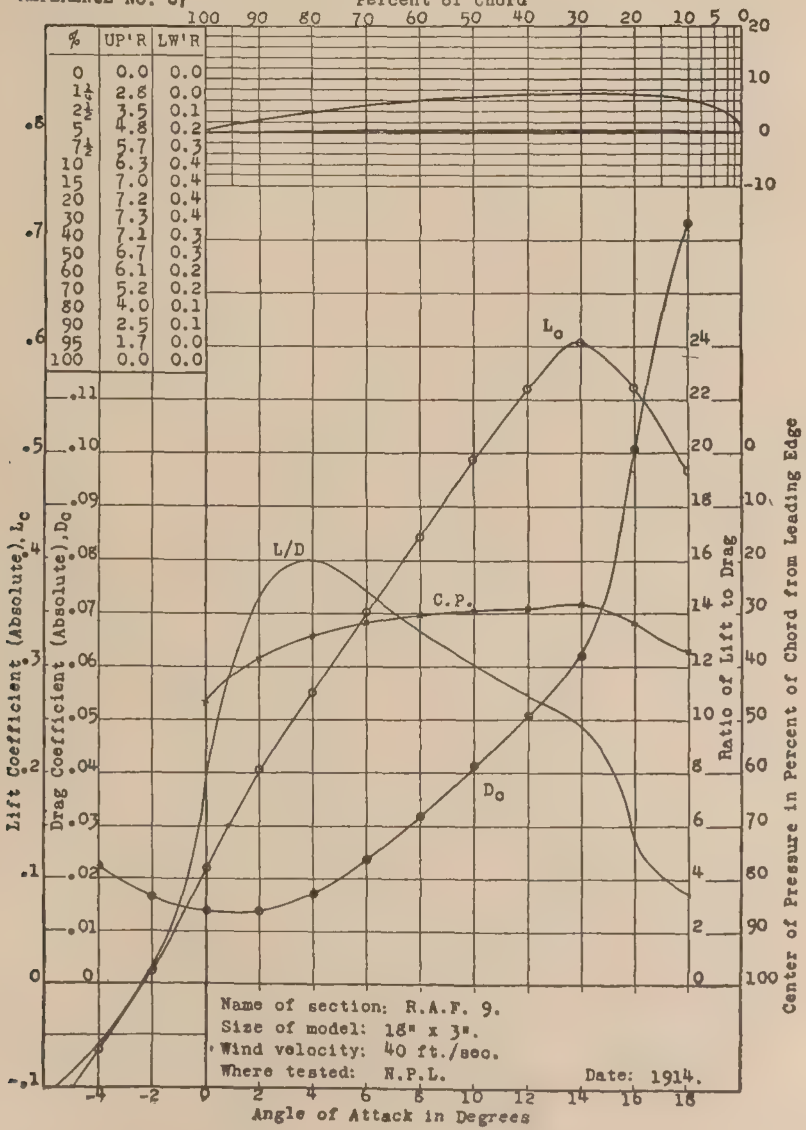
%	UP'R	LW'R
0	0.0	0.0
1¼	2.3	-2.3
2½	3.1	-3.1
5	4.5	-4.5
7½	5.5	-5.5
10	6.2	-6.2
15	7.0	-7.0
20	7.4	-7.4
30	7.9	-7.9
40	7.7	-7.7
50	7.3	-7.3
60	6.8	-6.8
70	5.9	-5.9
80	4.7	-4.7
90	3.2	-3.2
95	2.3	-2.3
100	0.0	0.0

REFERENCE NO. 66



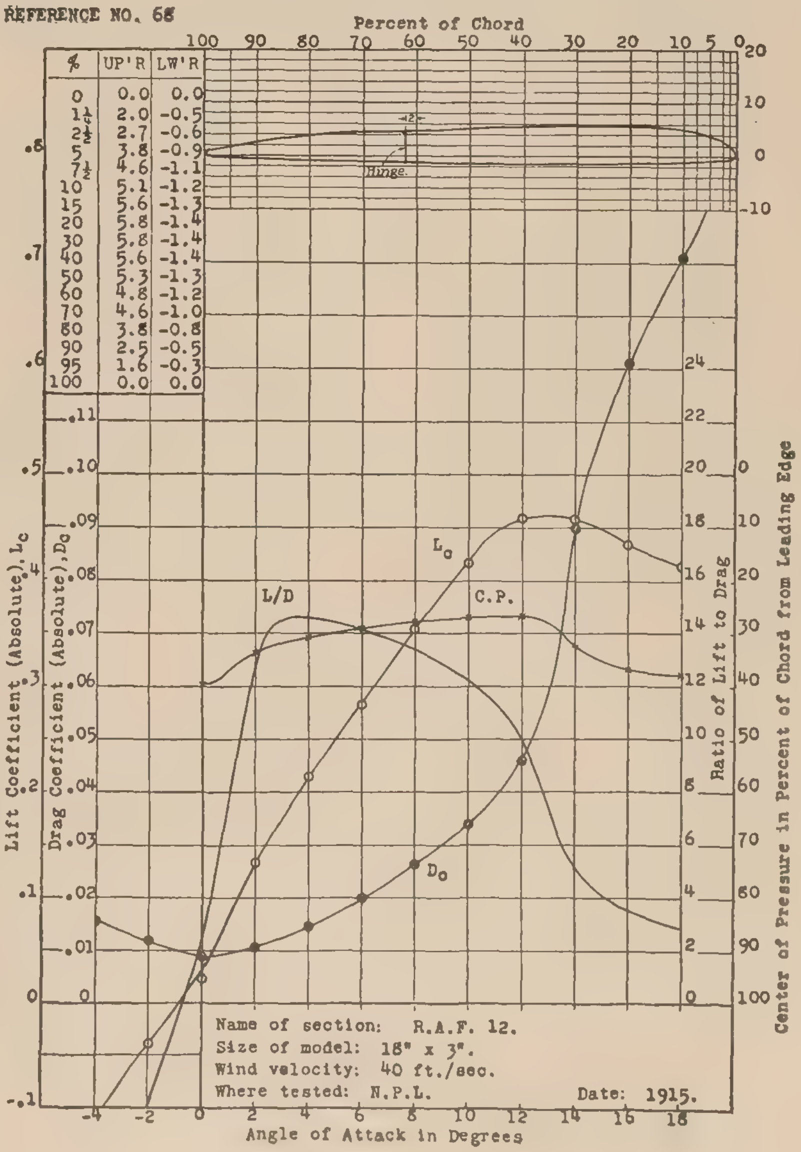
%	UP'R	LW'R
0	0.0	0.0
1¼	3.3	0.0
2½	4.1	0.1
5	5.3	0.4
7½	6.0	0.6
10	6.5	0.8
15	7.2	1.0
20	7.5	1.0
30	7.6	0.9
40	7.4	0.7
50	7.1	0.6
60	6.5	0.3
70	5.6	0.2
80	4.5	0.0
90	2.8	0.0
95	1.9	0.0
100	0.0	0.0

REFERENCE NO. 67



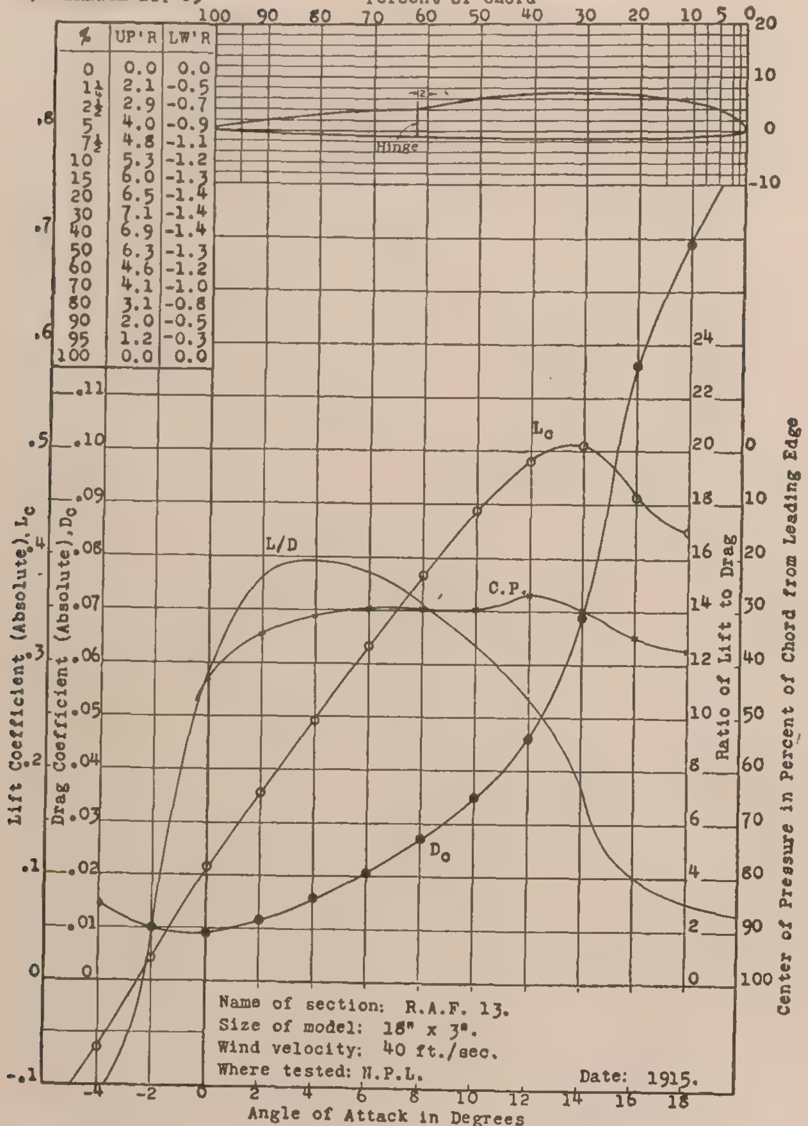
%	UP'R	LW'R
0	0.0	0.0
1¼	2.8	0.0
2½	3.5	0.1
5	4.8	0.2
7½	5.7	0.3
10	6.3	0.4
15	7.0	0.4
20	7.2	0.4
30	7.3	0.4
40	7.1	0.3
50	6.7	0.3
60	6.1	0.2
70	5.2	0.2
80	4.0	0.1
90	2.5	0.1
95	1.7	0.0
100	0.0	0.0

REFERENCE NO. 68



%	UP'R	LW'R
0	0.0	0.0
1¼	2.0	-0.5
2½	2.7	-0.6
5	3.8	-0.9
7½	4.6	-1.1
10	5.1	-1.2
15	5.6	-1.3
20	5.8	-1.4
30	5.8	-1.4
40	5.6	-1.4
50	5.3	-1.3
60	4.8	-1.2
70	4.6	-1.0
80	3.8	-0.8
90	2.5	-0.5
95	1.6	-0.3
100	0.0	0.0

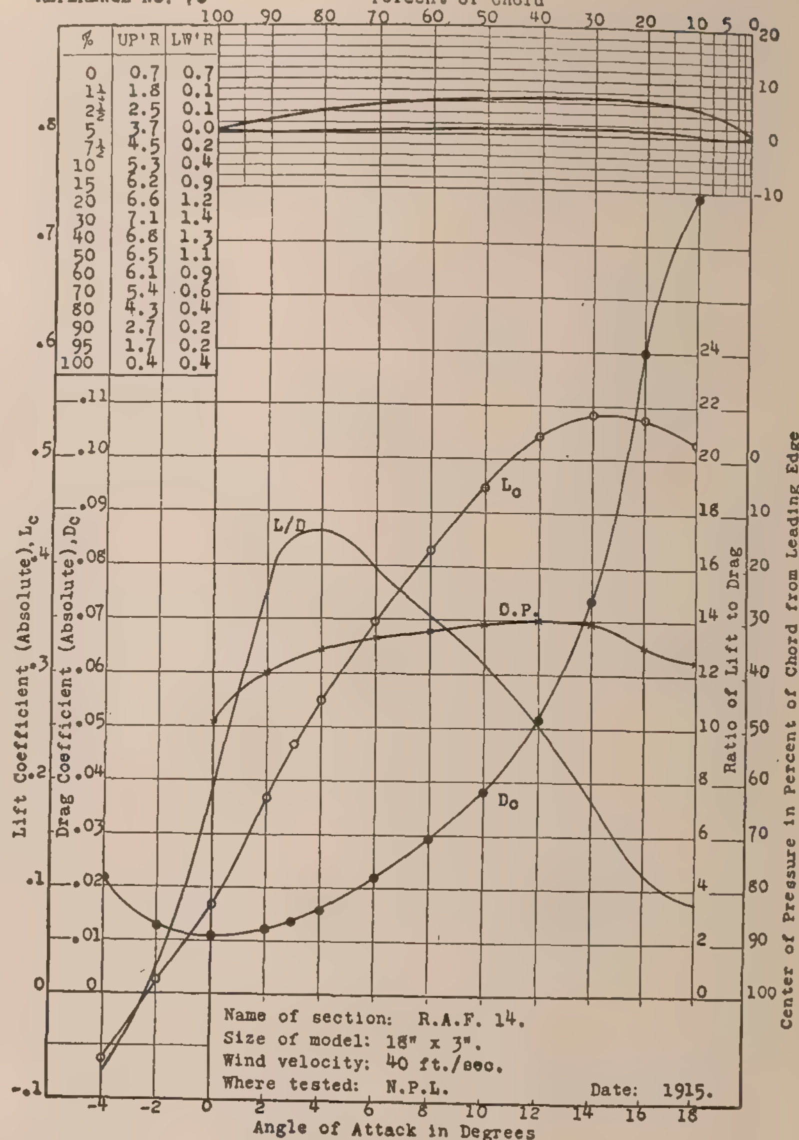
REFERENCE NO. 69



%	UP'R	LW'R
0	0.0	0.0
1¼	2.1	-0.5
2½	2.9	-0.7
5	4.0	-0.9
7½	4.8	-1.1
10	5.3	-1.2
15	6.0	-1.3
20	6.5	-1.4
30	7.1	-1.4
40	6.9	-1.4
50	6.3	-1.3
60	4.6	-1.2
70	4.1	-1.0
80	3.1	-0.8
90	2.0	-0.5
95	1.2	-0.3
100	0.0	0.0

Name of section: R.A.F. 13.
Size of model: 18" x 3".
Wind velocity: 40 ft./sec.
Where tested: N.P.L.
Date: 1915.

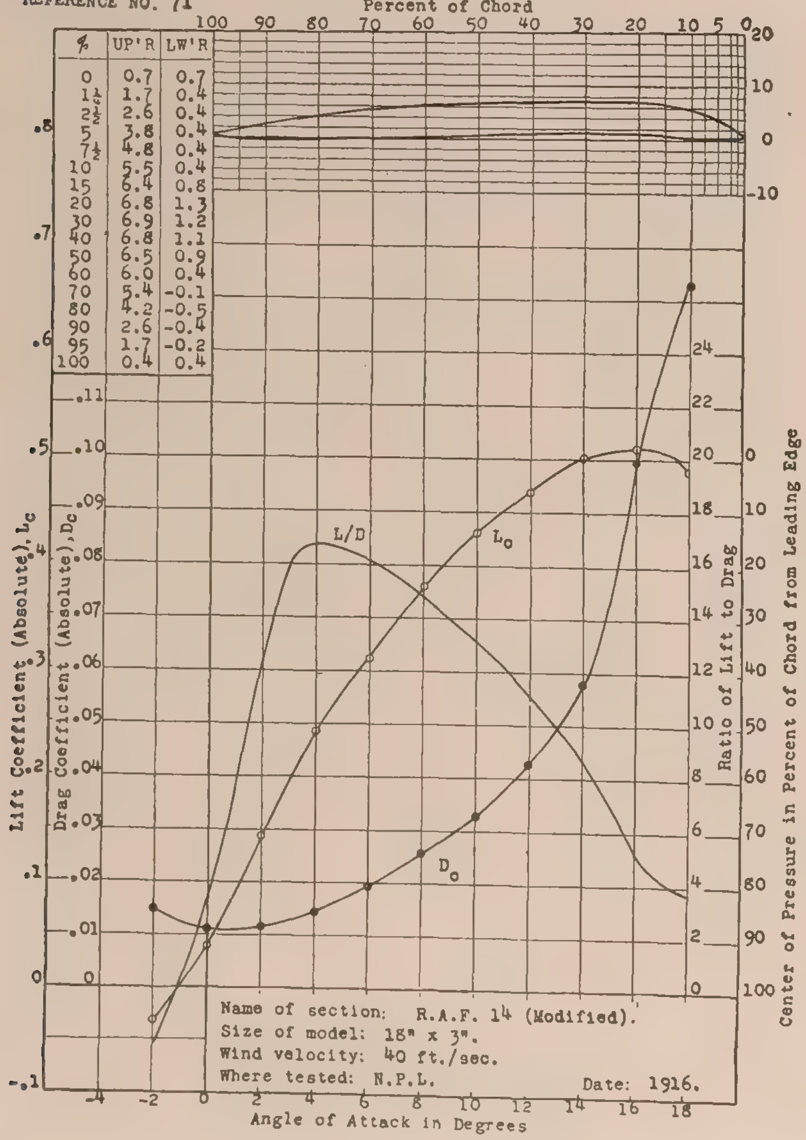
REFERENCE NO. 70



%	UP'R	LW'R
0	0.7	0.7
1¼	1.8	0.1
2½	2.5	0.1
5	3.7	0.0
7½	4.5	0.2
10	5.3	0.4
15	6.2	0.9
20	6.6	1.2
30	7.1	1.4
40	6.8	1.3
50	6.5	1.1
60	6.1	0.9
70	5.4	0.6
80	4.3	0.4
90	2.7	0.2
95	1.7	0.2
100	0.4	0.4

Name of section: R.A.F. 14.
Size of model: 18" x 3".
Wind velocity: 40 ft./sec.
Where tested: N.P.L.
Date: 1915.

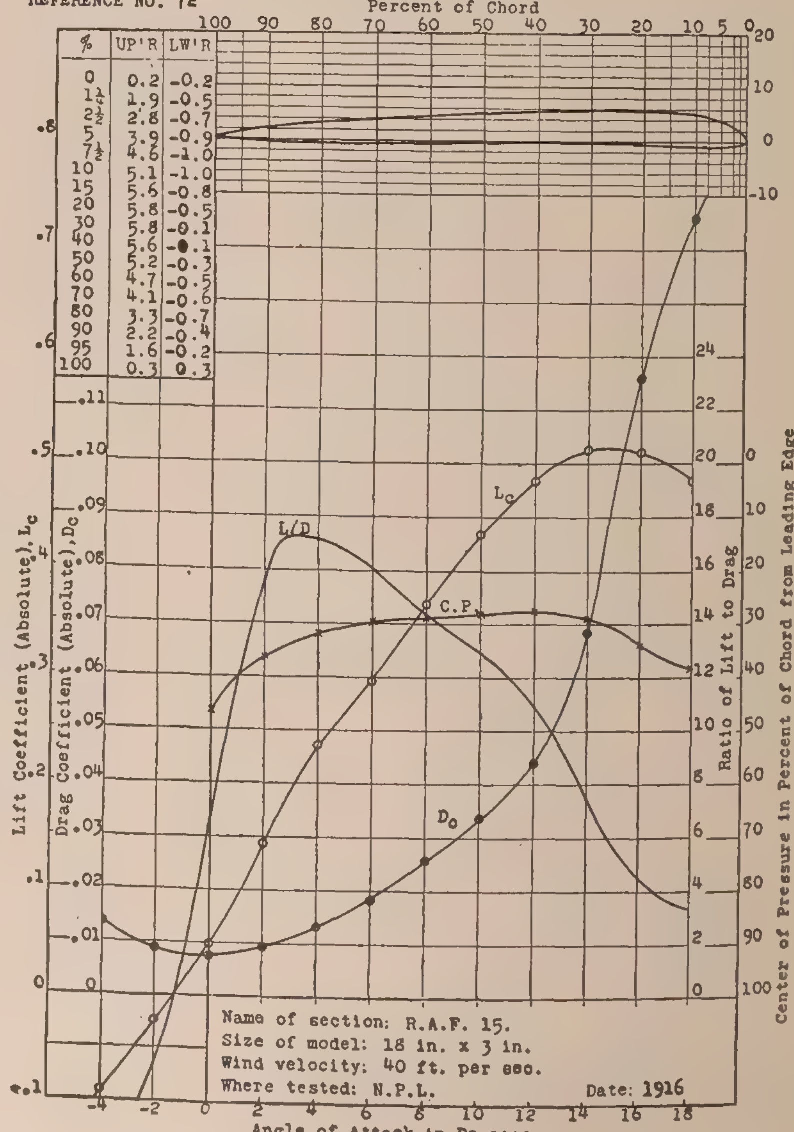
REFERENCE NO. 71



%	UP'R	LW'R
0	0.7	0.7
1¼	1.7	0.4
2½	2.6	0.4
5	3.8	0.4
7½	4.8	0.4
10	5.5	0.4
15	6.4	0.8
20	6.8	1.3
30	6.9	1.2
40	6.8	1.1
50	6.5	0.9
60	6.0	0.4
70	5.4	-0.1
80	4.2	-0.5
90	2.6	-0.4
95	1.7	-0.2
100	0.4	0.4

Name of section: R.A.F. 14 (Modified).
Size of model: 18" x 3".
Wind velocity: 40 ft./sec.
Where tested: N.P.L.
Date: 1916.

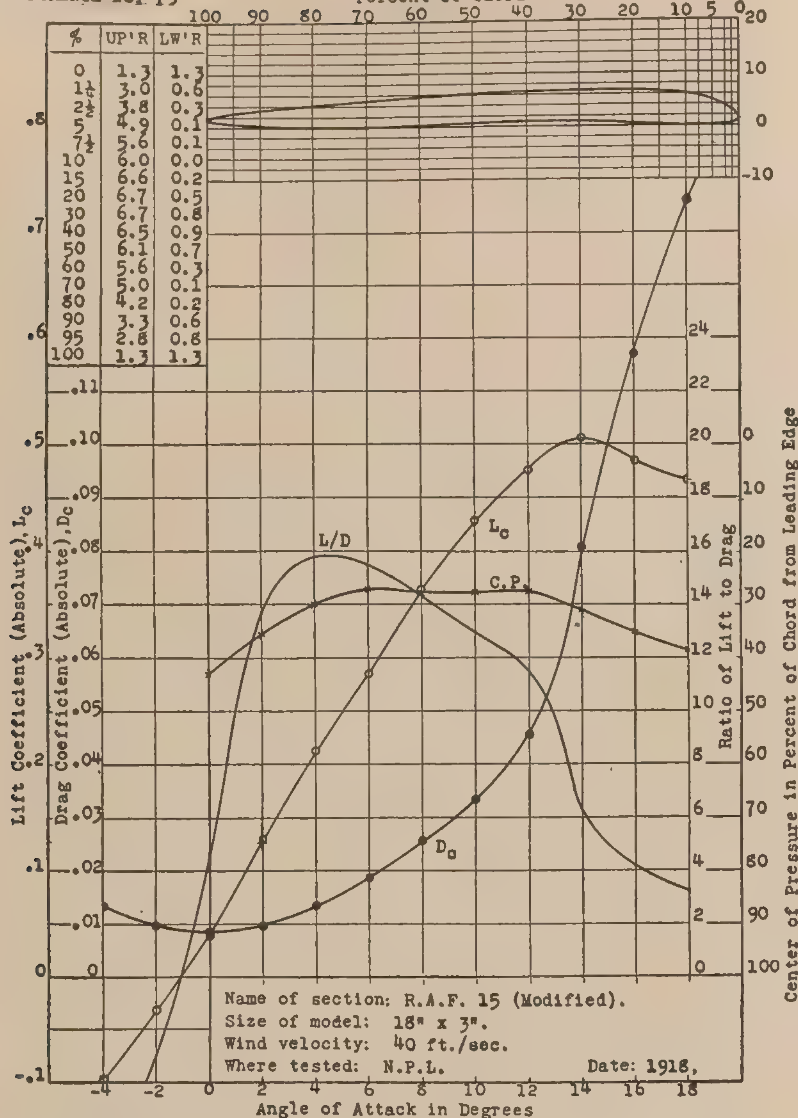
REFERENCE NO. 72



%	UP'R	LW'R
0	0.2	-0.2
1¼	1.9	-0.5
2½	2.8	-0.7
5	3.9	-0.9
7½	4.6	-1.0
10	5.1	-1.0
15	5.6	-0.8
20	5.8	-0.5
30	5.8	-0.1
40	5.6	-0.1
50	5.2	-0.3
60	4.7	-0.5
70	4.1	-0.6
80	3.3	-0.7
90	2.2	-0.4
95	1.6	-0.2
100	0.3	0.3

Name of section: R.A.F. 15.
Size of model: 18 in. x 3 in.
Wind velocity: 40 ft. per sec.
Where tested: N.P.L.
Date: 1916

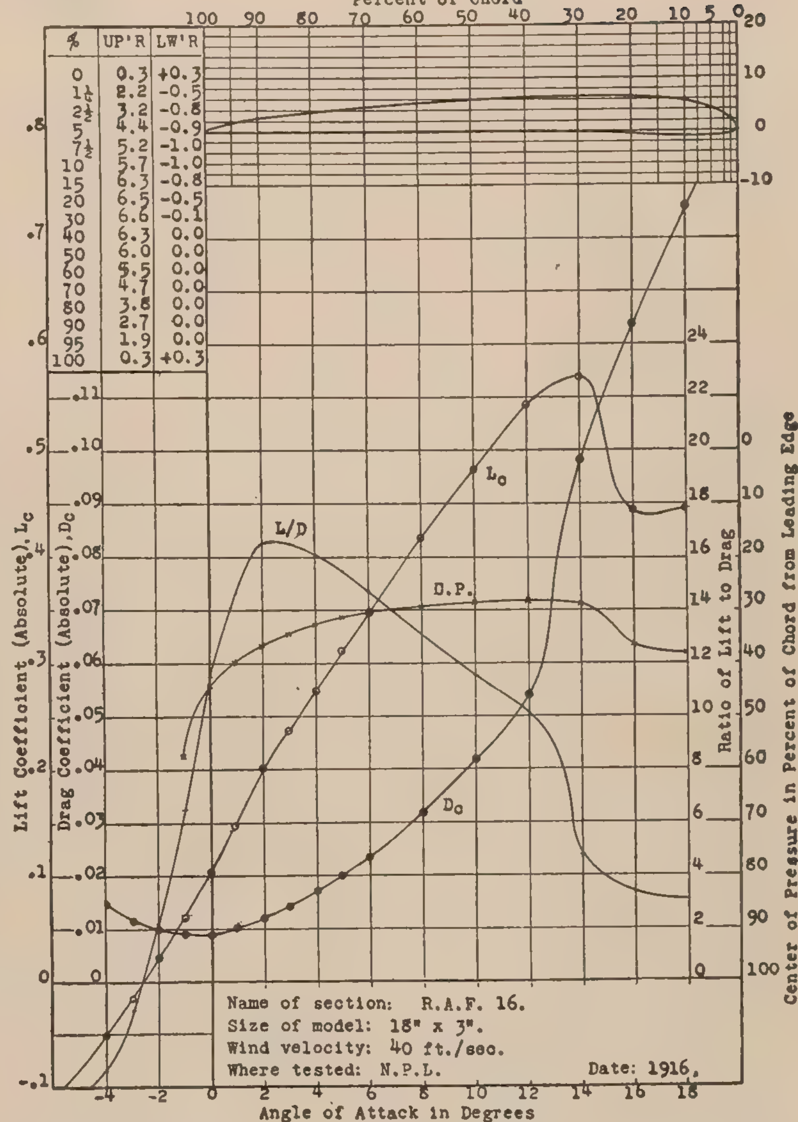
REFERENCE NO. 73



%	UP'R	LW'R
0	1.3	1.3
1¼	3.0	0.6
2½	3.8	0.3
5	4.9	0.1
7½	5.6	0.1
10	6.0	0.0
15	6.6	0.2
20	6.7	0.5
30	6.7	0.8
40	6.5	0.9
50	6.1	0.7
60	5.6	0.3
70	5.0	0.1
80	4.2	0.2
90	3.3	0.6
95	2.8	0.8
100	1.3	1.3

Name of section: R.A.F. 15 (Modified).
Size of model: 18" x 3".
Wind velocity: 40 ft./sec.
Where tested: N.P.L. Date: 1918.

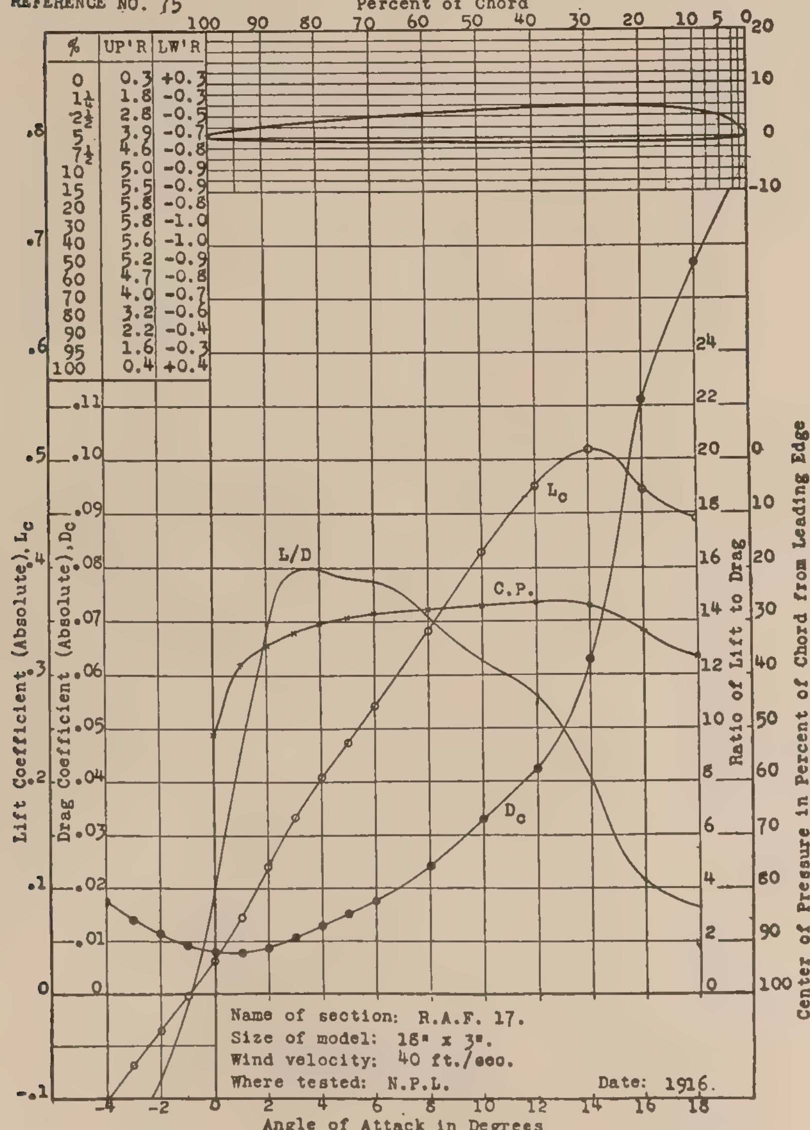
REFERENCE NO. 74



%	UP'R	LW'R
0	0.3	+0.3
1¼	2.2	-0.5
2½	3.2	-0.8
5	4.4	-0.9
7½	5.2	-1.0
10	5.7	-1.0
15	6.3	-0.8
20	6.5	-0.5
30	6.6	-0.1
40	6.3	0.0
50	6.0	0.0
60	5.5	0.0
70	4.7	0.0
80	3.8	0.0
90	2.7	0.0
95	1.9	0.0
100	0.3	+0.3

Name of section: R.A.F. 16.
Size of model: 18" x 3".
Wind velocity: 40 ft./sec.
Where tested: N.P.L. Date: 1916.

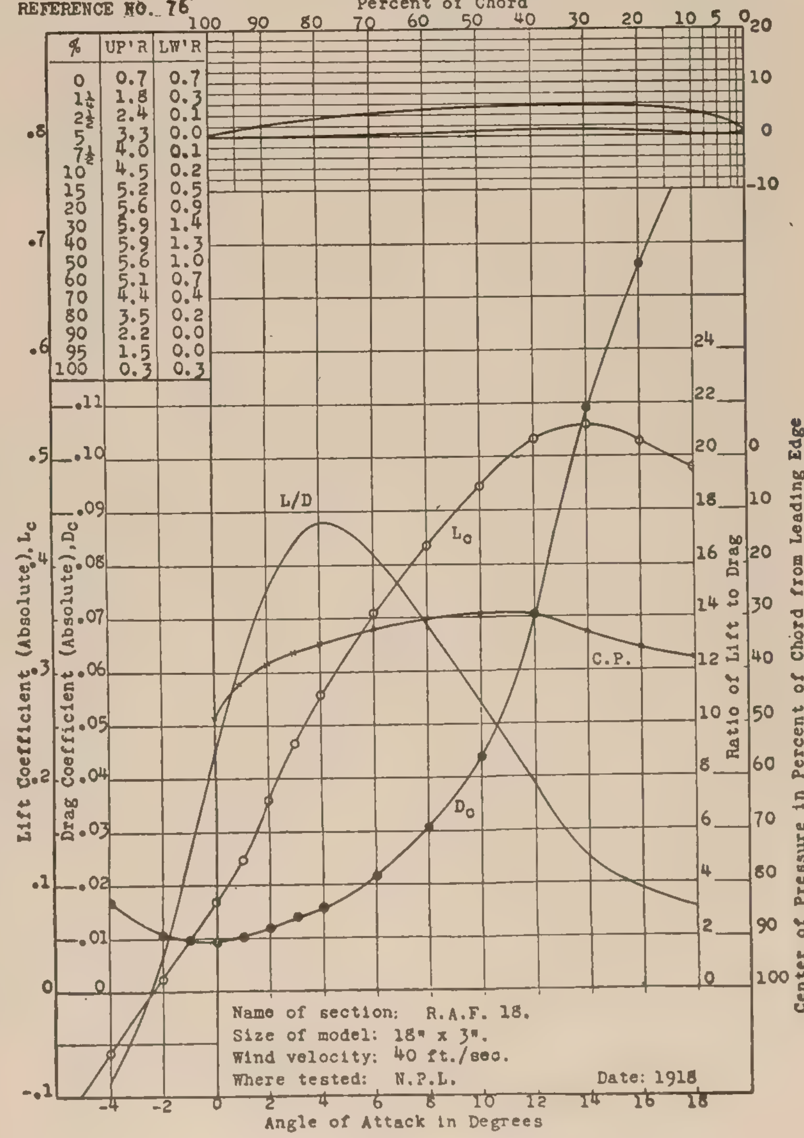
REFERENCE NO. 75



%	UP'R	LW'R
0	0.3	+0.3
1¼	1.8	-0.3
2½	2.8	-0.5
5	3.9	-0.7
7½	4.6	-0.8
10	5.0	-0.9
15	5.5	-0.9
20	5.8	-0.8
30	5.8	-1.0
40	5.6	-1.0
50	5.2	-0.9
60	4.7	-0.8
70	4.0	-0.7
80	3.2	-0.6
90	2.2	-0.4
95	1.6	-0.3
100	0.4	+0.4

Name of section: R.A.F. 17.
Size of model: 18" x 3".
Wind velocity: 40 ft./sec.
Where tested: N.P.L. Date: 1916.

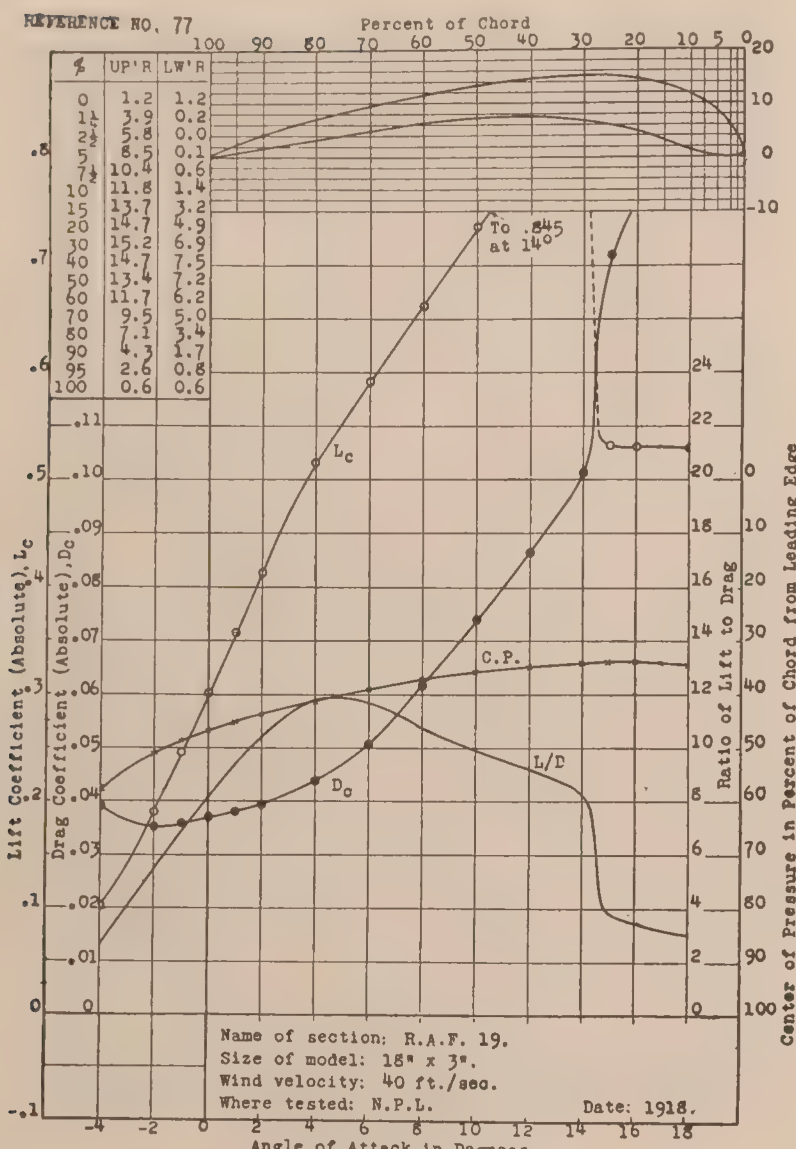
REFERENCE NO. 76



%	UP'R	LW'R
0	0.7	0.7
1¼	1.8	0.3
2½	2.4	0.1
5	3.3	0.0
7½	4.0	0.1
10	4.5	0.2
15	5.2	0.5
20	5.6	0.9
30	5.9	1.4
40	5.9	1.3
50	5.6	1.0
60	5.1	0.7
70	4.4	0.4
80	3.5	0.2
90	2.2	0.0
95	1.5	0.0
100	0.3	0.3

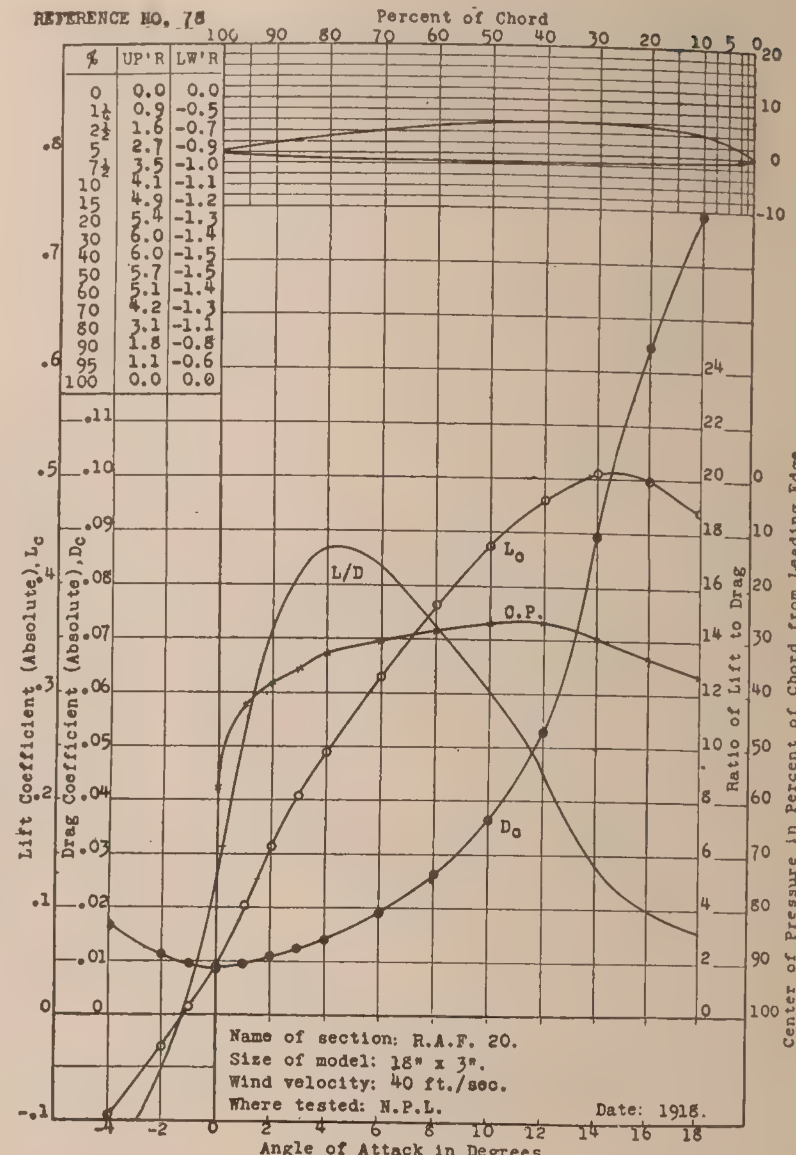
Name of section: R.A.F. 18.
Size of model: 18" x 3".
Wind velocity: 40 ft./sec.
Where tested: N.P.L. Date: 1918

REFERENCE NO. 77



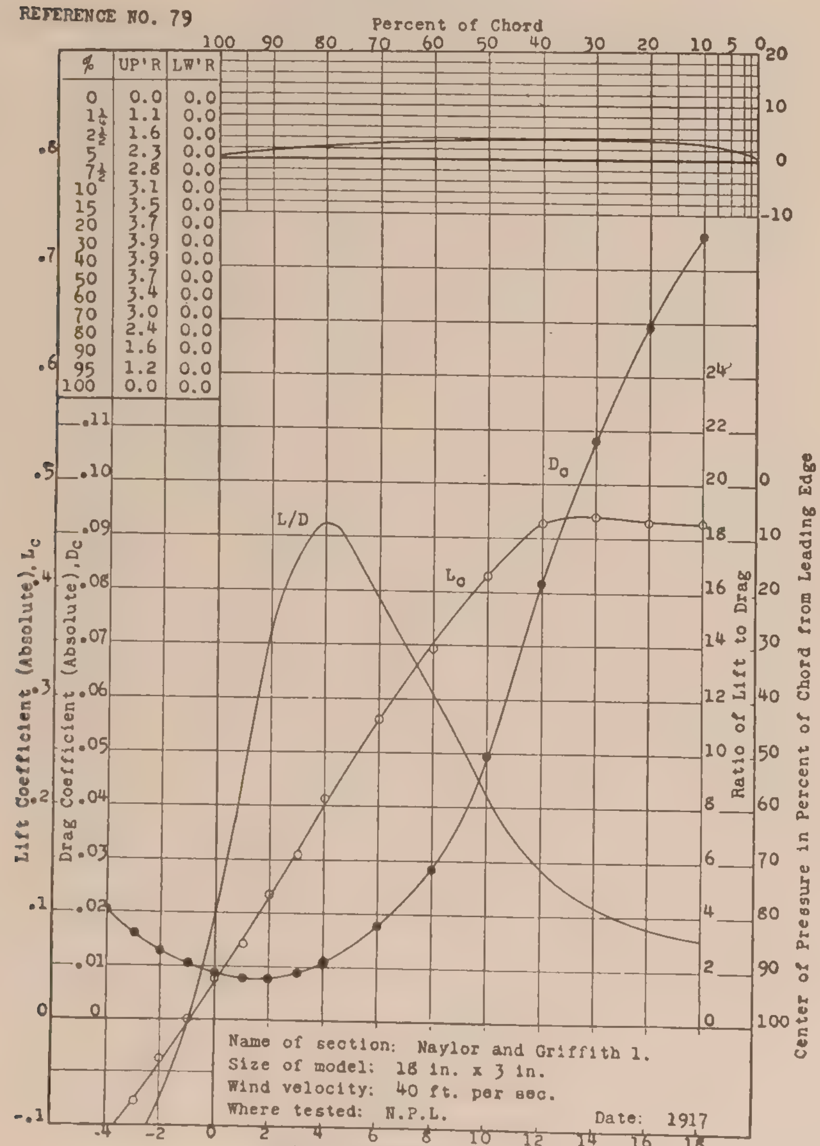
%	UP'R	LW'R
0	1.2	1.2
1¼	3.9	0.2
2½	5.8	0.0
5	8.5	0.1
7½	10.4	0.6
10	11.8	1.4
15	13.7	3.2
20	14.7	4.9
30	15.2	6.9
40	14.7	7.5
50	13.4	7.2
60	11.7	6.2
70	9.5	5.0
80	7.1	3.4
90	4.3	1.7
95	2.6	0.8
100	0.6	0.6

REFERENCE NO. 78



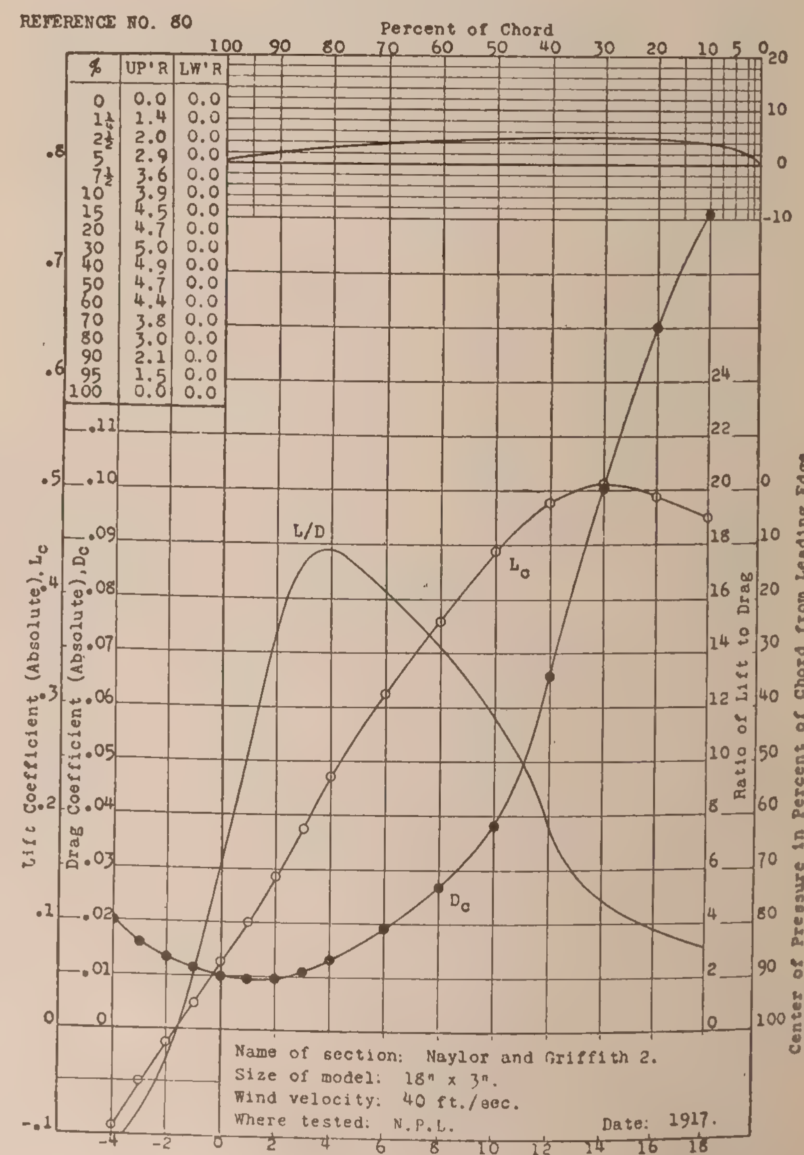
%	UP'R	LW'R
0	0.0	0.0
1¼	0.9	-0.5
2½	1.6	-0.7
5	2.7	-0.9
7½	3.5	-1.0
10	4.1	-1.1
15	4.9	-1.2
20	5.4	-1.3
30	6.0	-1.4
40	6.0	-1.5
50	5.7	-1.5
60	5.1	-1.4
70	4.2	-1.3
80	3.1	-1.1
90	1.8	-0.8
95	1.1	-0.6
100	0.0	0.0

REFERENCE NO. 79



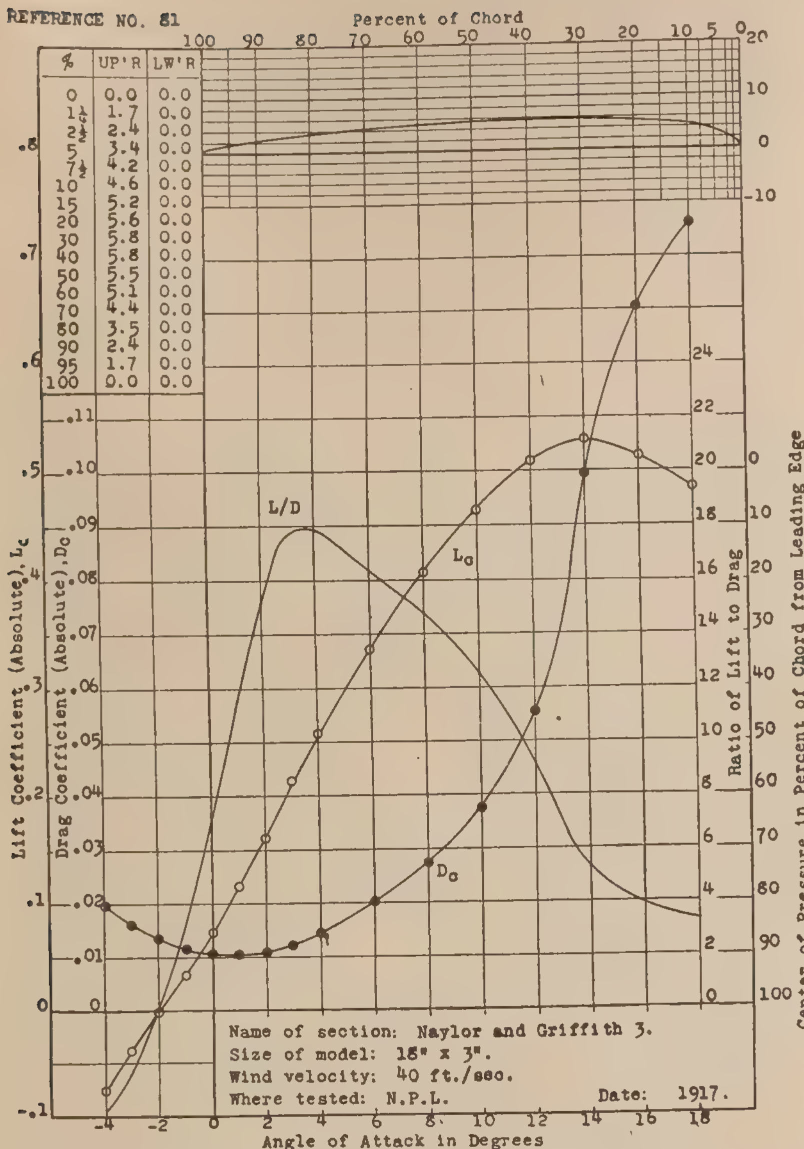
%	UP'R	LW'R
0	0.0	0.0
1¼	1.1	0.0
2½	1.6	0.0
5	2.3	0.0
7½	2.8	0.0
10	3.1	0.0
15	3.5	0.0
20	3.7	0.0
30	3.9	0.0
40	3.9	0.0
50	3.7	0.0
60	3.4	0.0
70	3.0	0.0
80	2.4	0.0
90	1.6	0.0
95	1.2	0.0
100	0.0	0.0

REFERENCE NO. 80



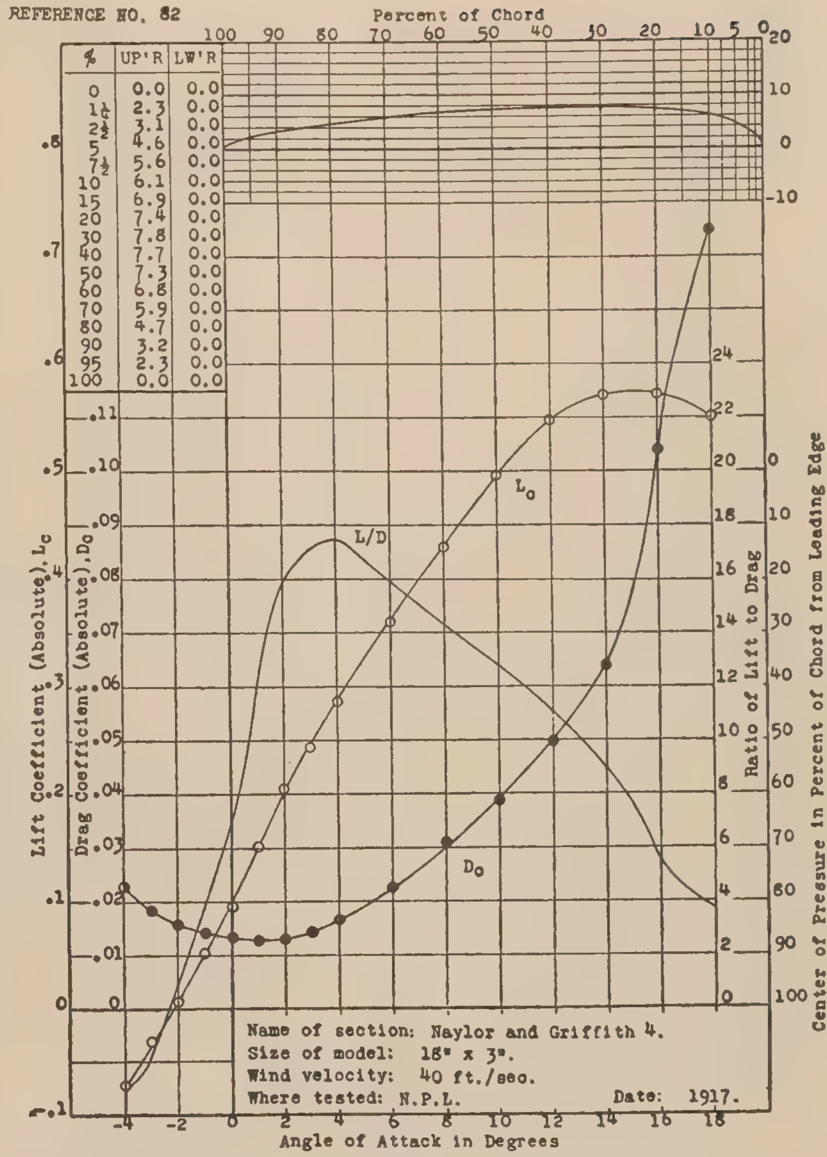
%	UP'R	LW'R
0	0.0	0.0
1¼	1.4	0.0
2½	2.0	0.0
5	2.9	0.0
7½	3.6	0.0
10	3.9	0.0
15	4.5	0.0
20	4.7	0.0
30	5.0	0.0
40	4.9	0.0
50	4.7	0.0
60	4.4	0.0
70	3.8	0.0
80	3.0	0.0
90	2.1	0.0
95	1.5	0.0
100	0.0	0.0

REFERENCE NO. 81



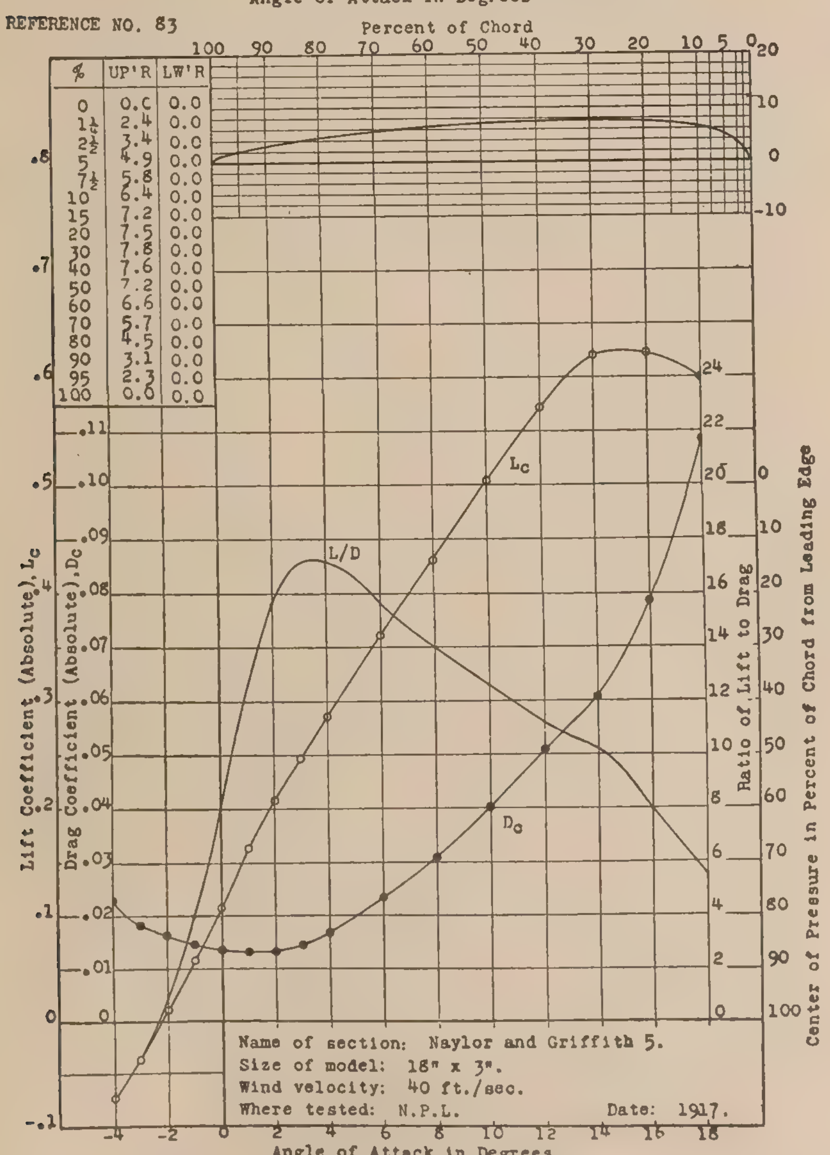
%	UP'R	LW'R
0	0.0	0.0
1½	1.7	0.0
2½	2.4	0.0
5	3.4	0.0
7½	4.2	0.0
10	4.6	0.0
15	5.2	0.0
20	5.6	0.0
30	5.8	0.0
40	5.8	0.0
50	5.5	0.0
60	5.1	0.0
70	4.4	0.0
80	3.5	0.0
90	2.4	0.0
95	1.7	0.0
100	0.0	0.0

REFERENCE NO. 82



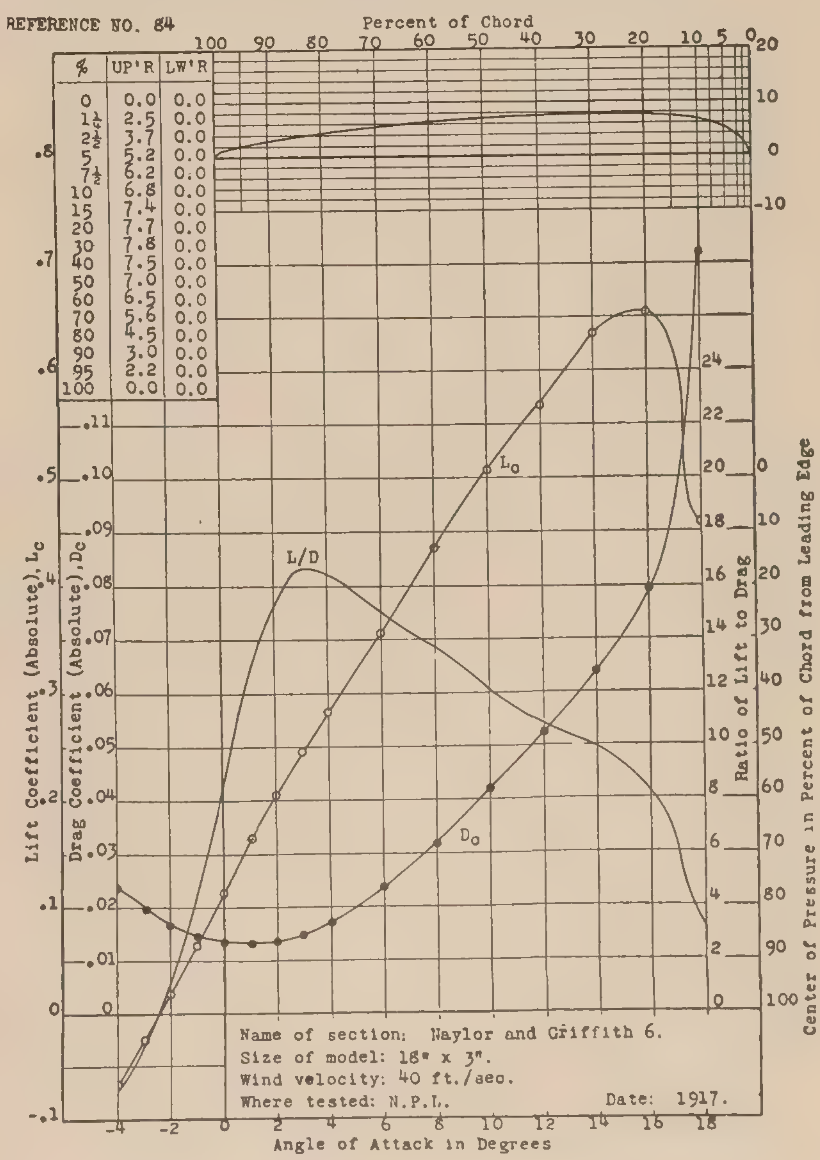
%	UP'R	LW'R
0	0.0	0.0
1½	2.3	0.0
2½	3.1	0.0
5	4.6	0.0
7½	5.6	0.0
10	6.1	0.0
15	6.9	0.0
20	7.4	0.0
30	7.8	0.0
40	7.7	0.0
50	7.3	0.0
60	6.8	0.0
70	5.9	0.0
80	4.7	0.0
90	3.2	0.0
95	2.3	0.0
100	0.0	0.0

REFERENCE NO. 83



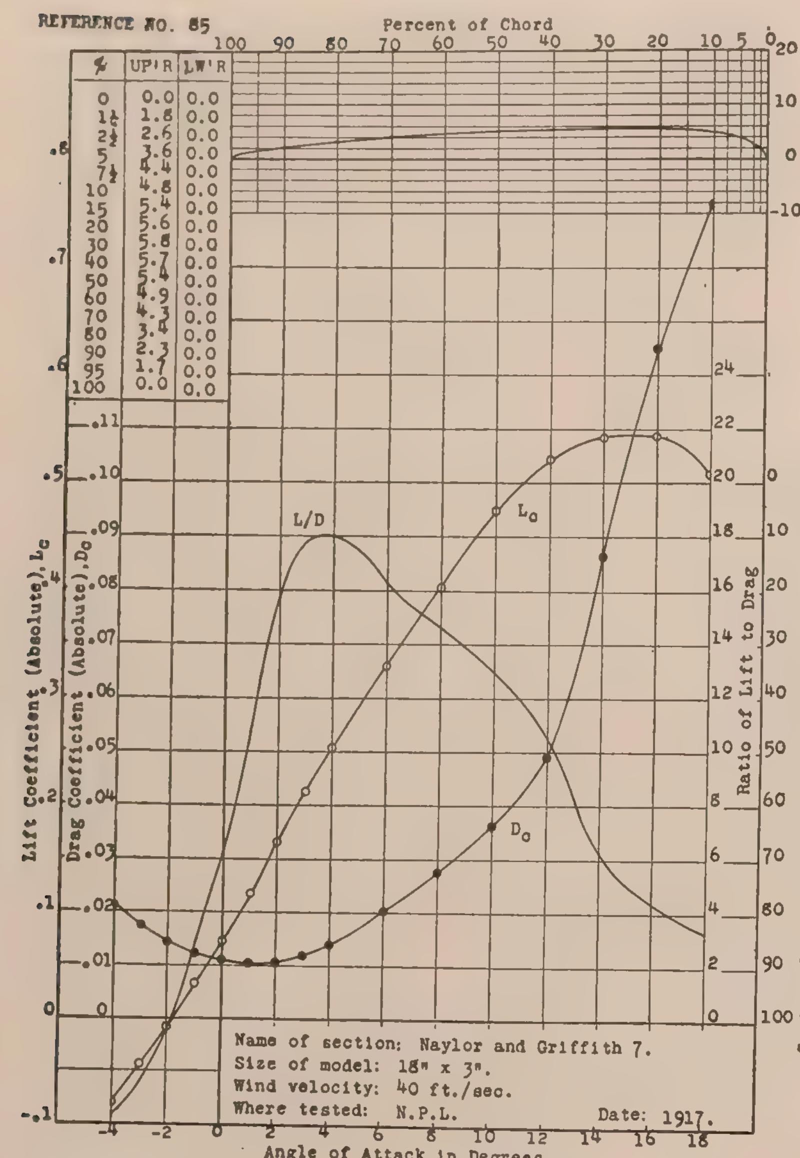
%	UP'R	LW'R
0	0.0	0.0
1½	2.4	0.0
2½	3.4	0.0
5	4.9	0.0
7½	5.8	0.0
10	6.4	0.0
15	7.2	0.0
20	7.5	0.0
30	7.8	0.0
40	7.6	0.0
50	7.2	0.0
60	6.6	0.0
70	5.7	0.0
80	4.5	0.0
90	3.1	0.0
95	2.3	0.0
100	0.0	0.0

REFERENCE NO. 84



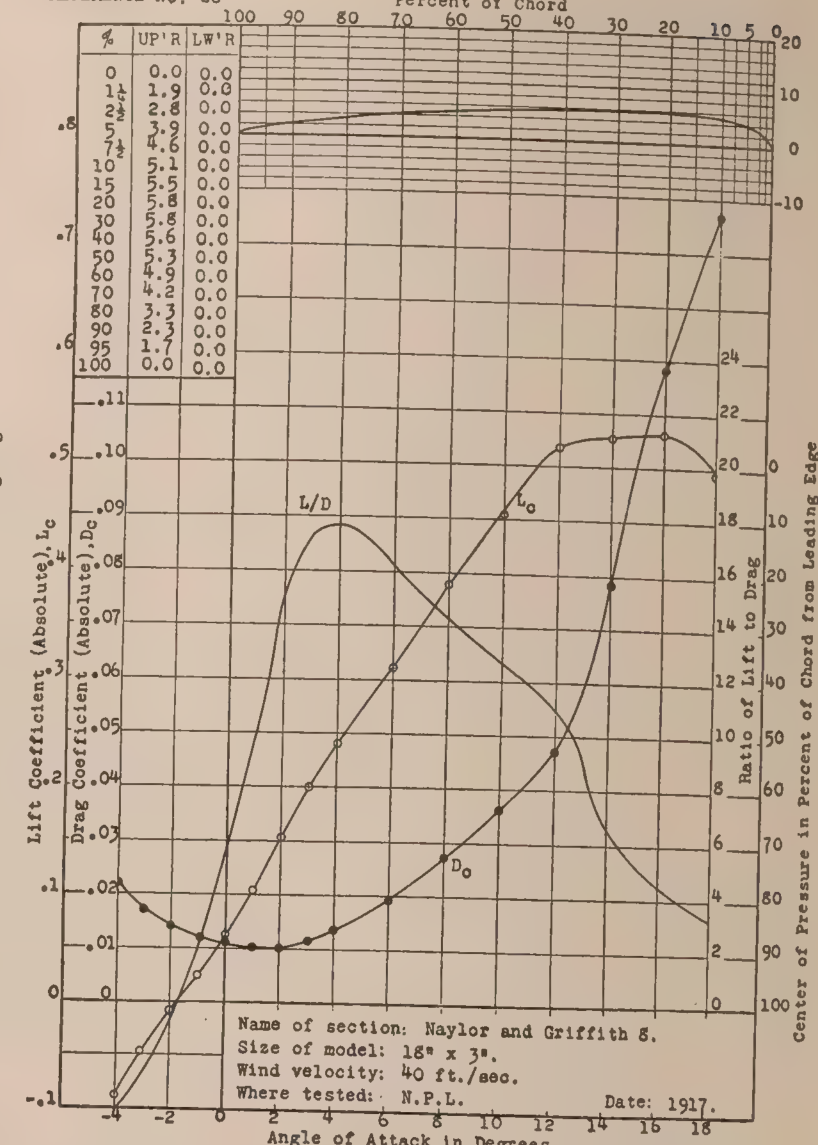
%	UP'R	LW'R
0	0.0	0.0
1½	2.5	0.0
2½	3.7	0.0
5	5.2	0.0
7½	6.2	0.0
10	6.8	0.0
15	7.4	0.0
20	7.7	0.0
30	7.8	0.0
40	7.5	0.0
50	7.0	0.0
60	6.5	0.0
70	5.6	0.0
80	4.5	0.0
90	3.0	0.0
95	2.2	0.0
100	0.0	0.0

REFERENCE NO. 85



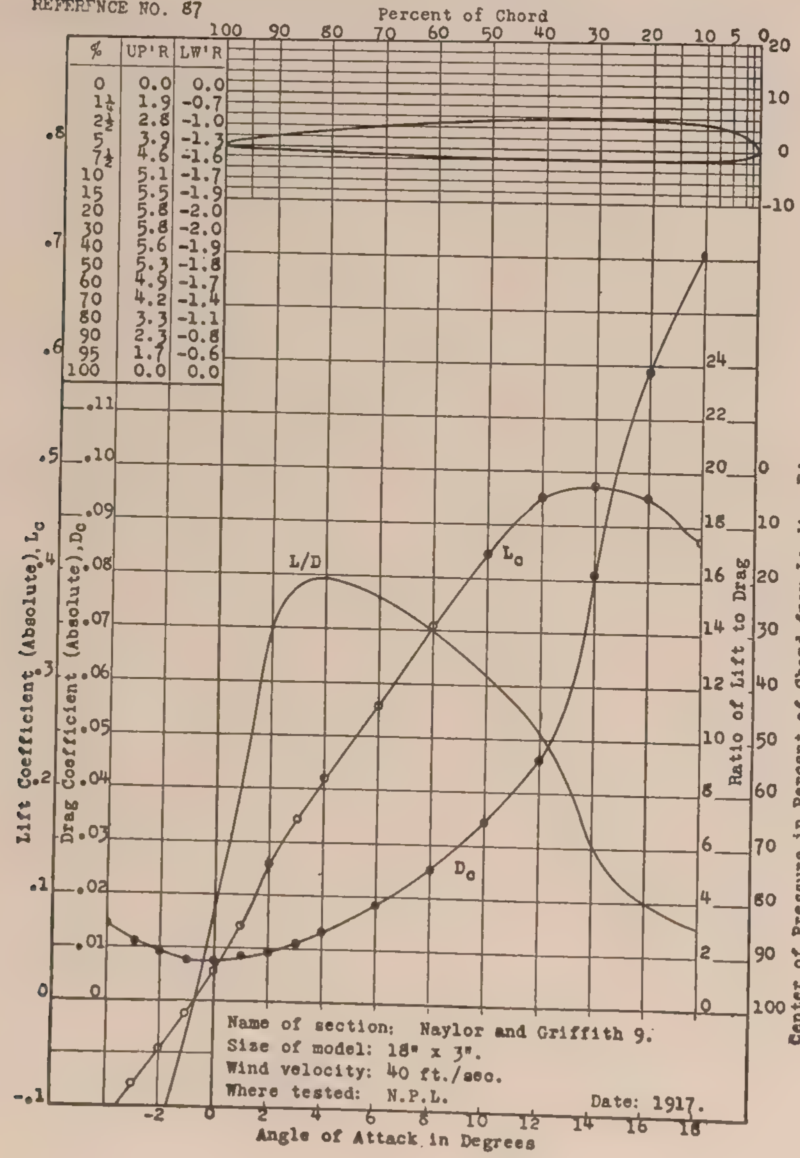
%	UP'R	LW'R
0	0.0	0.0
1¼	1.8	0.0
2½	2.6	0.0
5	3.6	0.0
7½	4.4	0.0
10	4.8	0.0
15	5.4	0.0
20	5.6	0.0
30	5.8	0.0
40	5.7	0.0
50	5.4	0.0
60	4.9	0.0
70	4.3	0.0
80	3.4	0.0
90	2.3	0.0
95	1.7	0.0
100	0.0	0.0

REFERENCE NO. 86



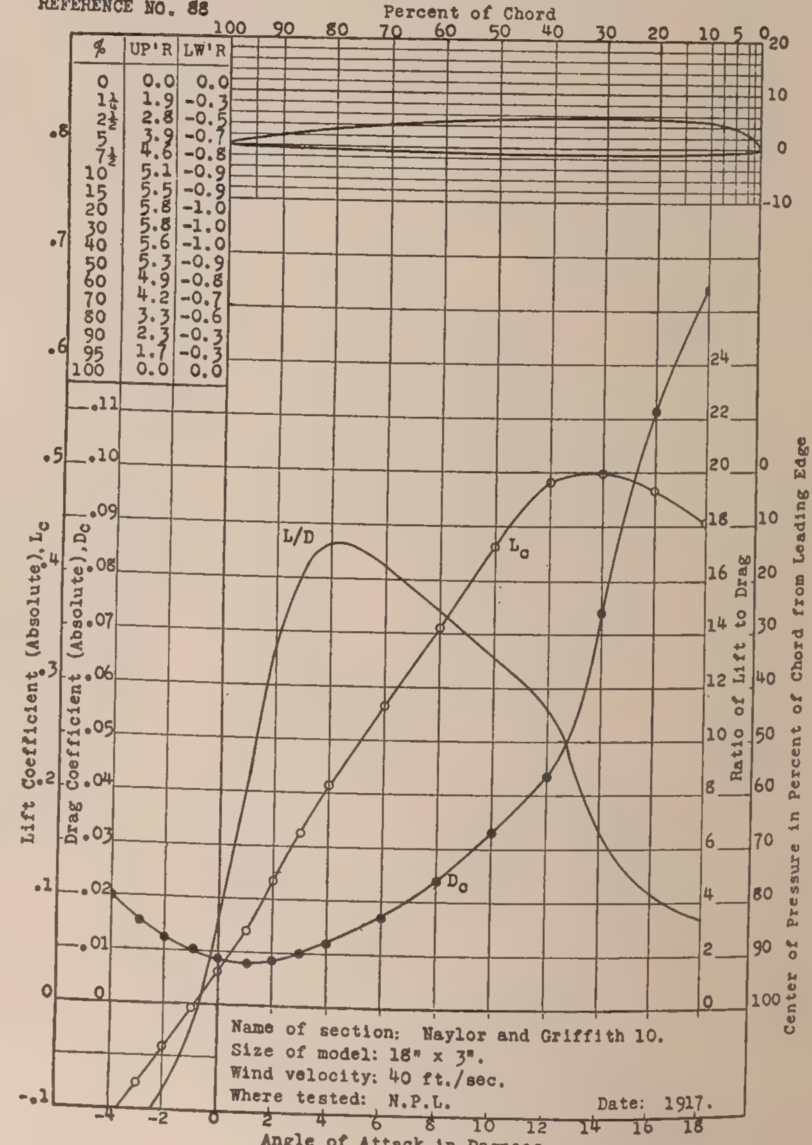
%	UP'R	LW'R
0	0.0	0.0
1¼	1.9	0.0
2½	2.8	0.0
5	3.9	0.0
7½	4.6	0.0
10	5.1	0.0
15	5.5	0.0
20	5.8	0.0
30	5.8	0.0
40	5.6	0.0
50	5.3	0.0
60	4.9	0.0
70	4.2	0.0
80	3.3	0.0
90	2.3	0.0
95	1.7	0.0
100	0.0	0.0

REFERENCE NO. 87



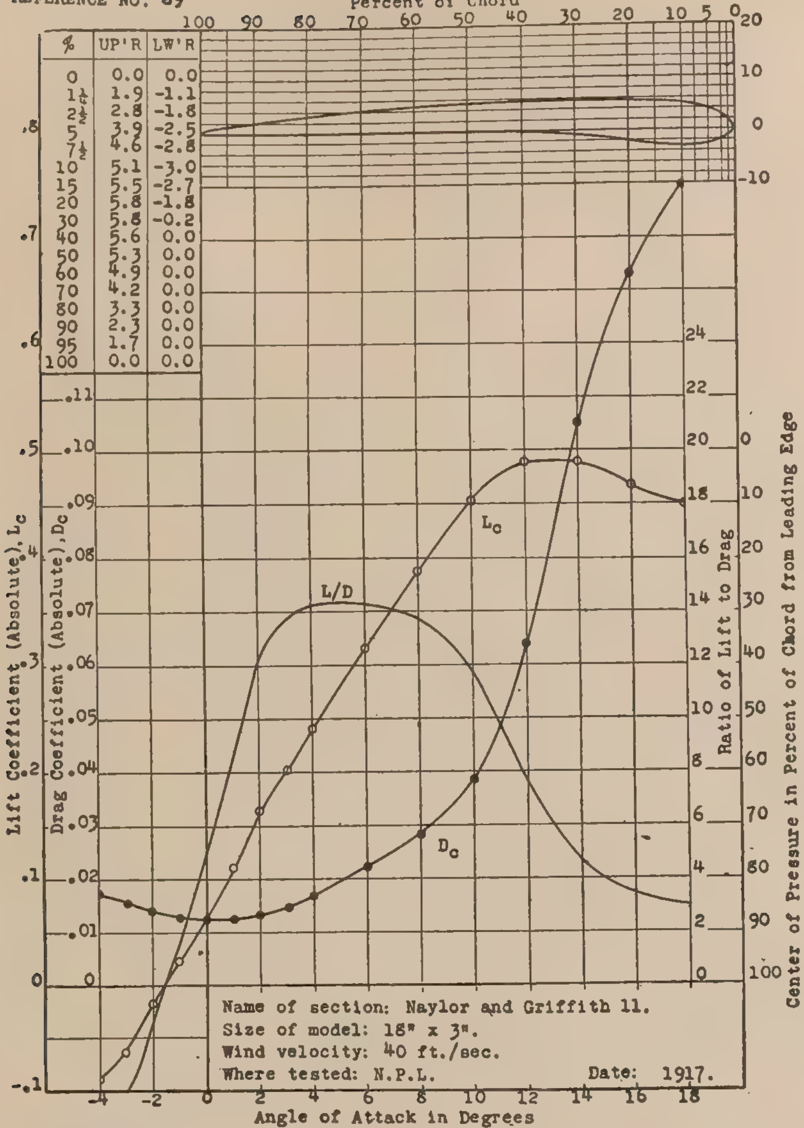
%	UP'R	LW'R
0	0.0	0.0
1¼	1.9	-0.7
2½	2.8	-1.0
5	3.9	-1.3
7½	4.6	-1.6
10	5.1	-1.7
15	5.5	-1.9
20	5.8	-2.0
30	5.8	-2.0
40	5.6	-1.9
50	5.3	-1.8
60	4.9	-1.7
70	4.2	-1.4
80	3.3	-1.1
90	2.3	-0.8
95	1.7	-0.6
100	0.0	0.0

REFERENCE NO. 88



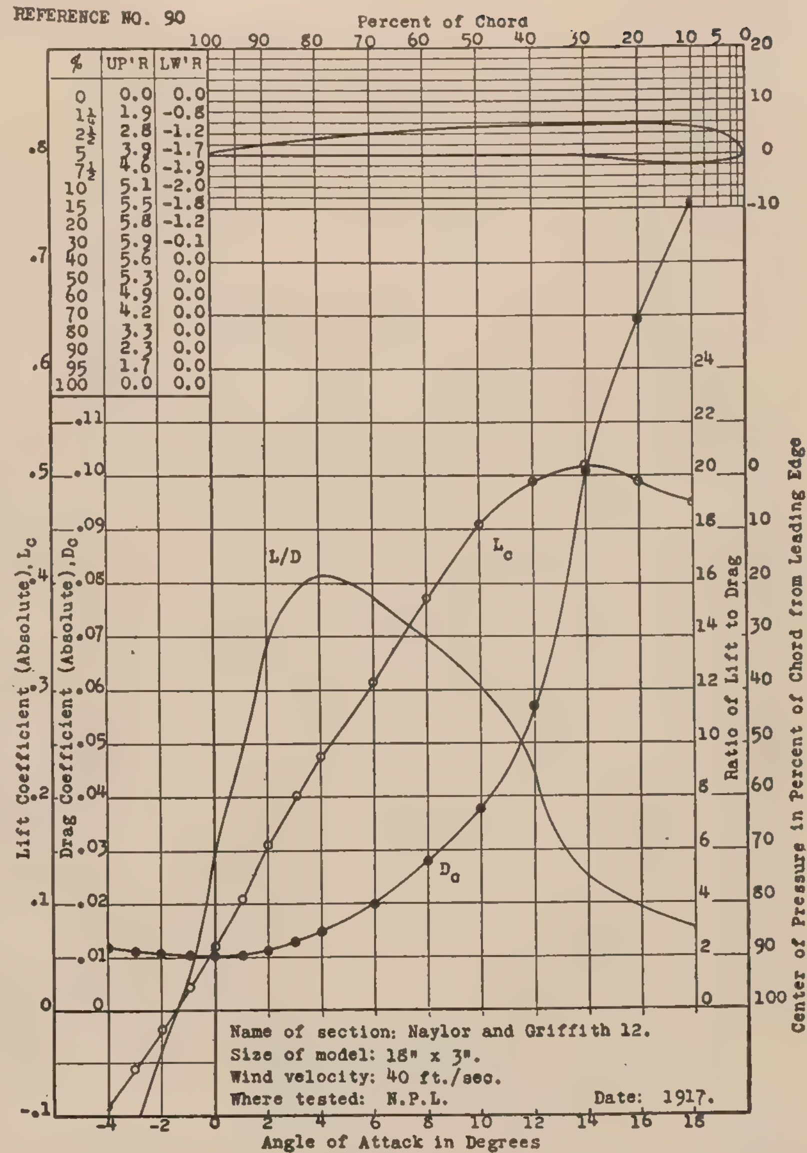
%	UP'R	LW'R
0	0.0	0.0
1¼	1.9	-0.3
2½	2.8	-0.5
5	3.9	-0.7
7½	4.6	-0.8
10	5.1	-0.9
15	5.5	-0.9
20	5.8	-1.0
30	5.8	-1.0
40	5.6	-1.0
50	5.3	-0.9
60	4.9	-0.8
70	4.2	-0.7
80	3.3	-0.6
90	2.3	-0.3
95	1.7	-0.3
100	0.0	0.0

REFERENCE NO. 89



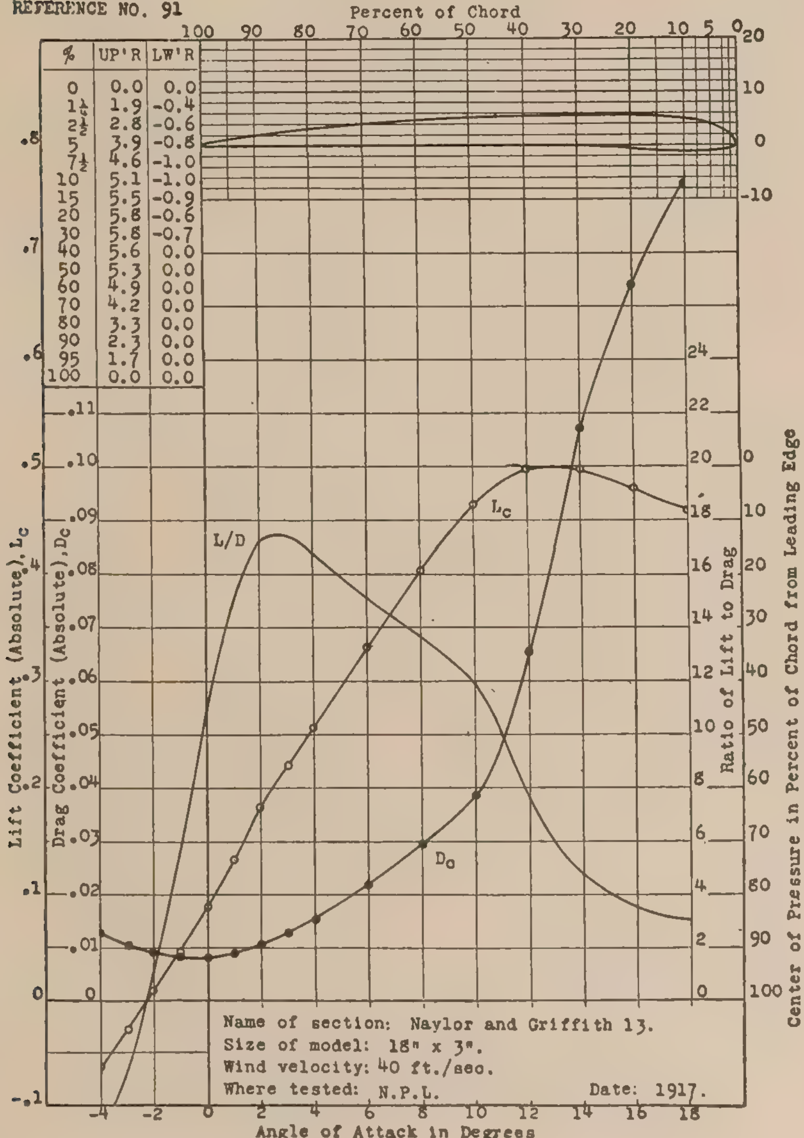
%	UP'R	LW'R
0	0.0	0.0
1¼	1.9	-1.1
2½	2.8	-1.8
5	3.9	-2.5
7½	4.6	-2.8
10	5.1	-3.0
15	5.5	-2.7
20	5.8	-1.8
30	5.8	-0.2
40	5.6	0.0
50	5.3	0.0
60	4.9	0.0
70	4.2	0.0
80	3.3	0.0
90	2.3	0.0
95	1.7	0.0
100	0.0	0.0

REFERENCE NO. 90



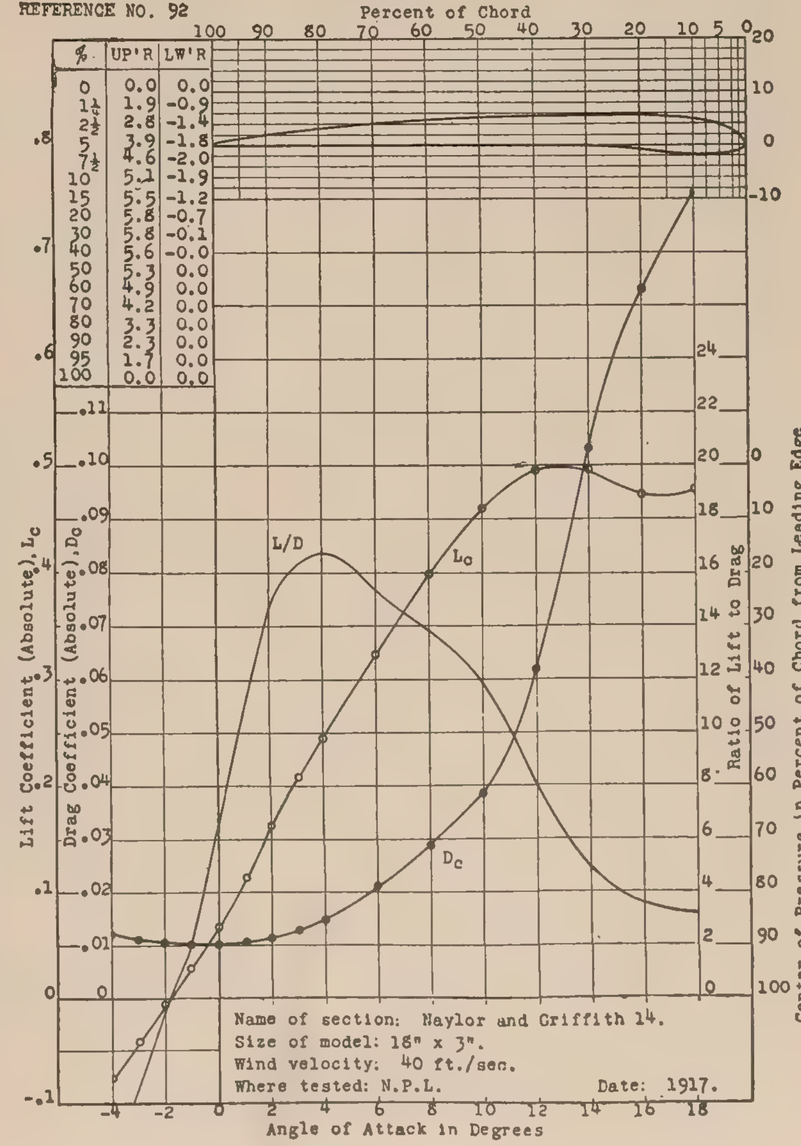
%	UP'R	LW'R
0	0.0	0.0
1¼	1.9	-0.8
2½	2.8	-1.2
5	3.9	-1.7
7½	4.6	-1.9
10	5.1	-2.0
15	5.5	-1.8
20	5.8	-1.2
30	5.9	-0.1
40	5.6	0.0
50	5.3	0.0
60	4.9	0.0
70	4.2	0.0
80	3.3	0.0
90	2.3	0.0
95	1.7	0.0
100	0.0	0.0

REFERENCE NO. 91



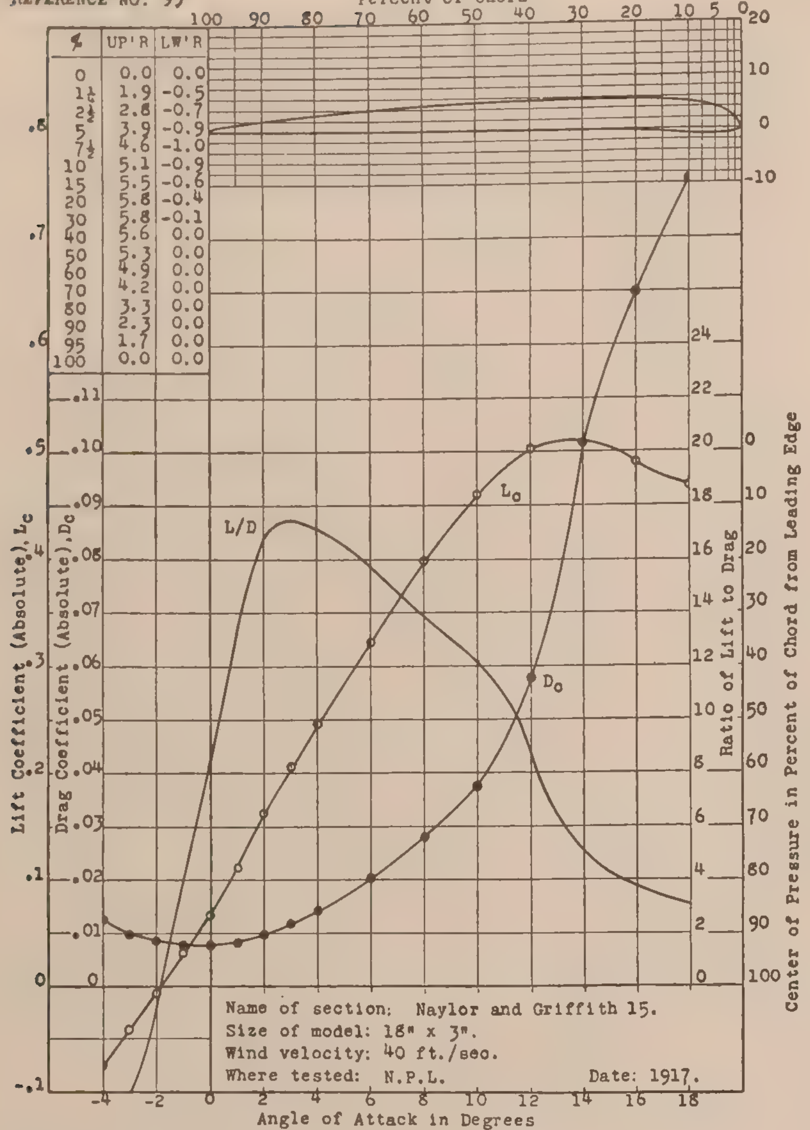
%	UP'R	LW'R
0	0.0	0.0
1¼	1.9	-0.4
2½	2.8	-0.6
5	3.9	-0.8
7½	4.6	-1.0
10	5.1	-1.0
15	5.5	-0.9
20	5.8	-0.6
30	5.8	-0.7
40	5.6	0.0
50	5.3	0.0
60	4.9	0.0
70	4.2	0.0
80	3.3	0.0
90	2.3	0.0
95	1.7	0.0
100	0.0	0.0

REFERENCE NO. 92

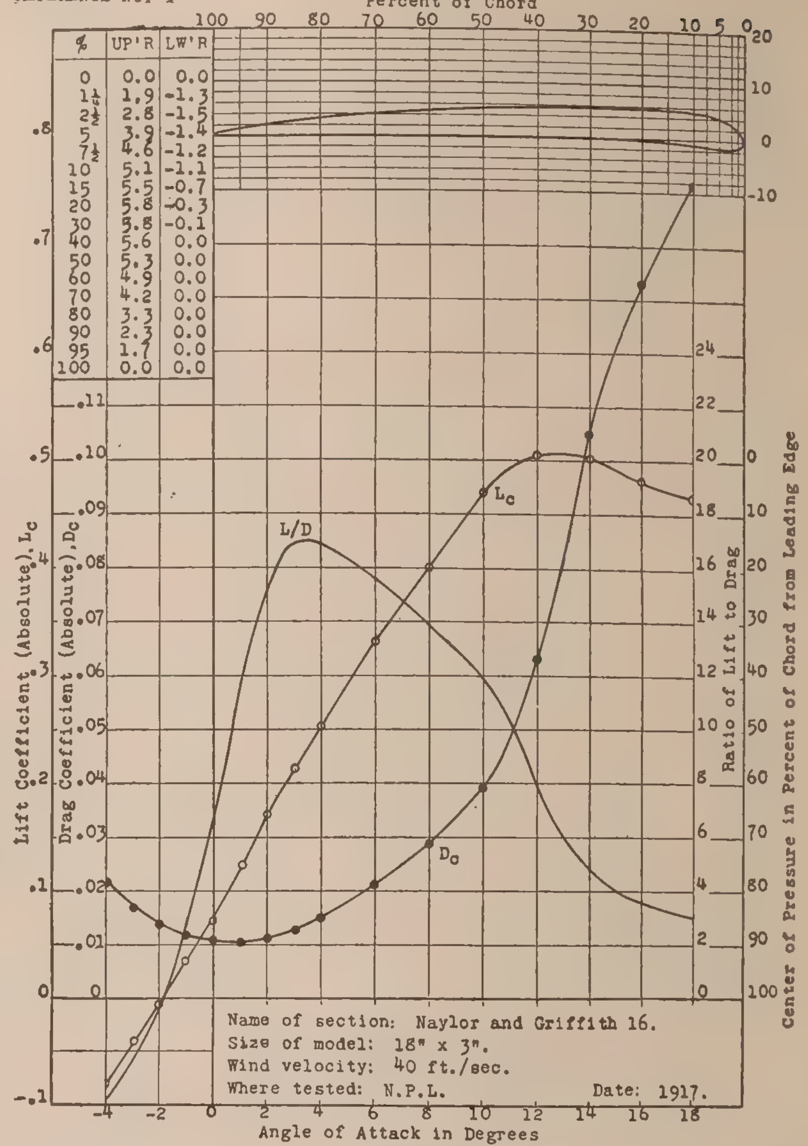


%	UP'R	LW'R
0	0.0	0.0
1¼	1.9	-0.9
2½	2.8	-1.4
5	3.9	-1.8
7½	4.6	-2.0
10	5.1	-1.9
15	5.5	-1.2
20	5.8	-0.7
30	5.8	-0.1
40	5.6	-0.0
50	5.3	0.0
60	4.9	0.0
70	4.2	0.0
80	3.3	0.0
90	2.3	0.0
95	1.7	0.0
100	0.0	0.0

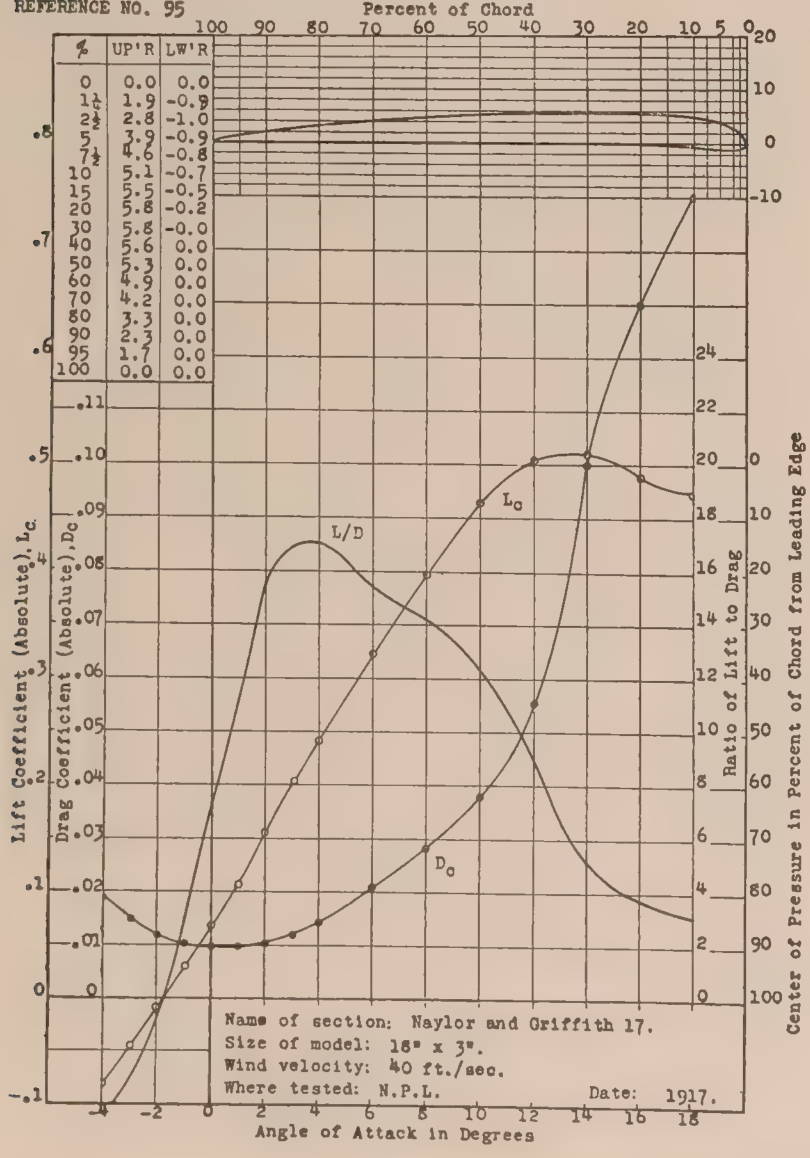
REFERENCE NO. 93



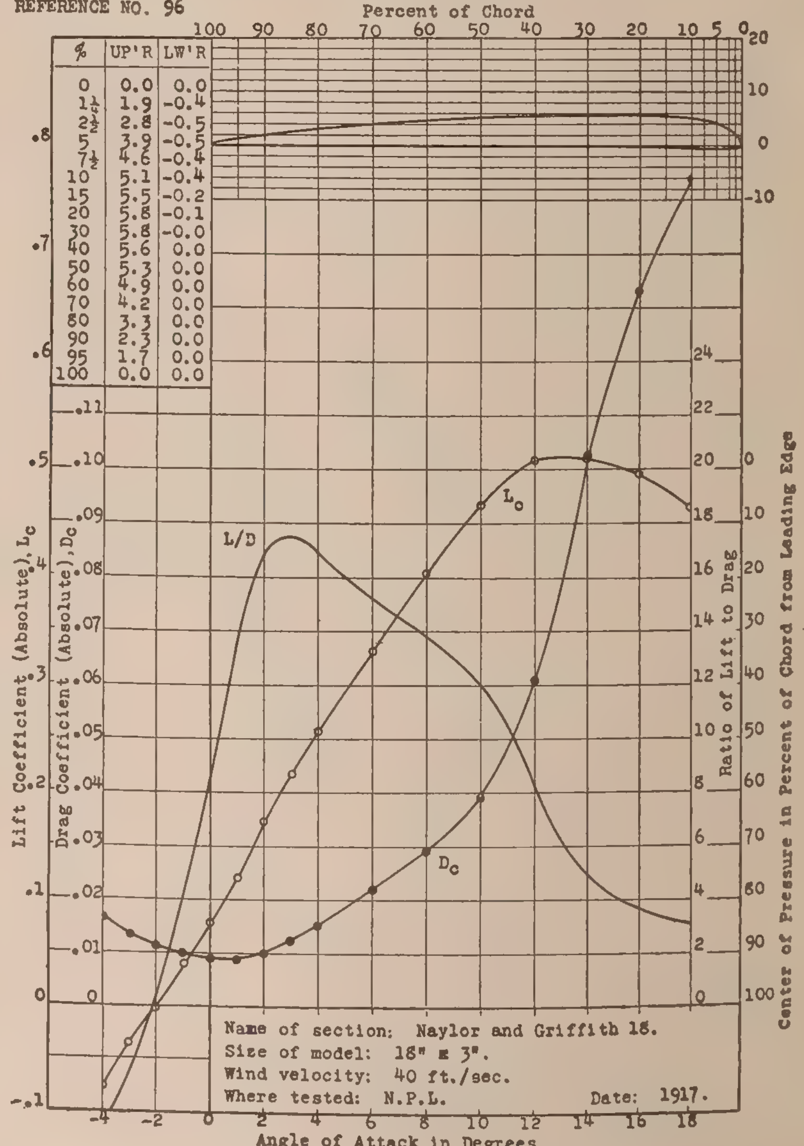
REFERENCE NO. 94



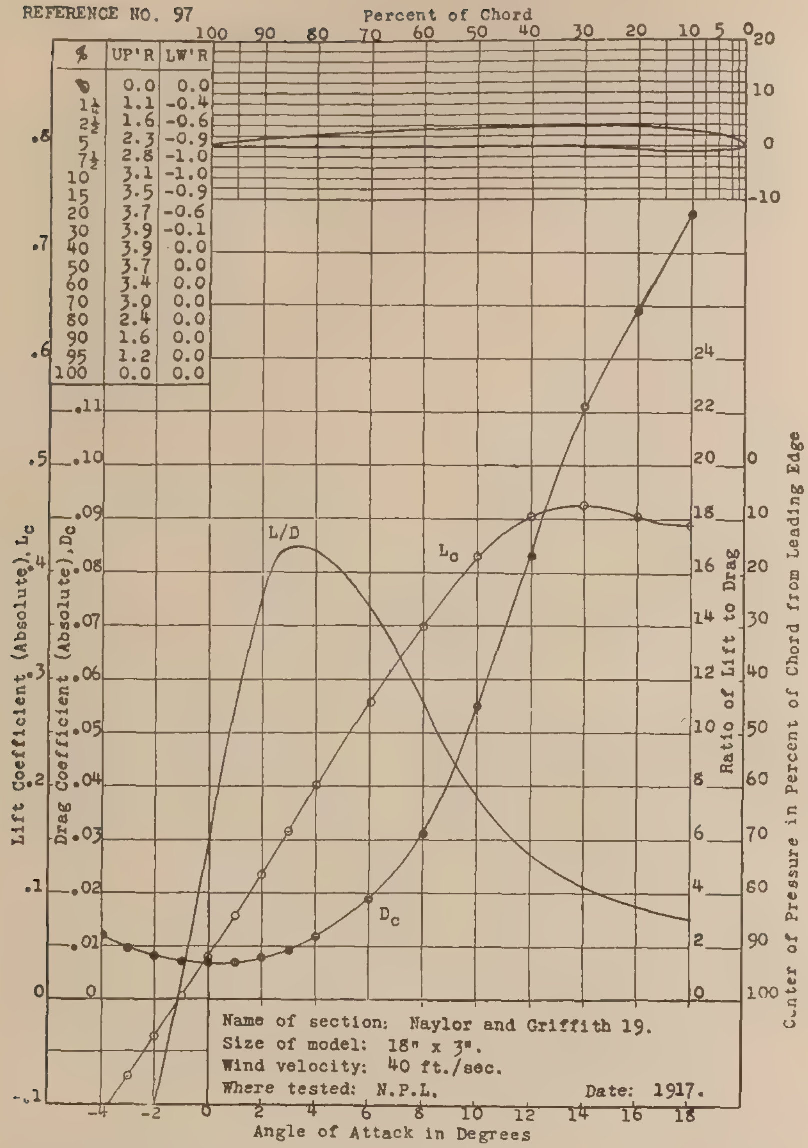
REFERENCE NO. 95



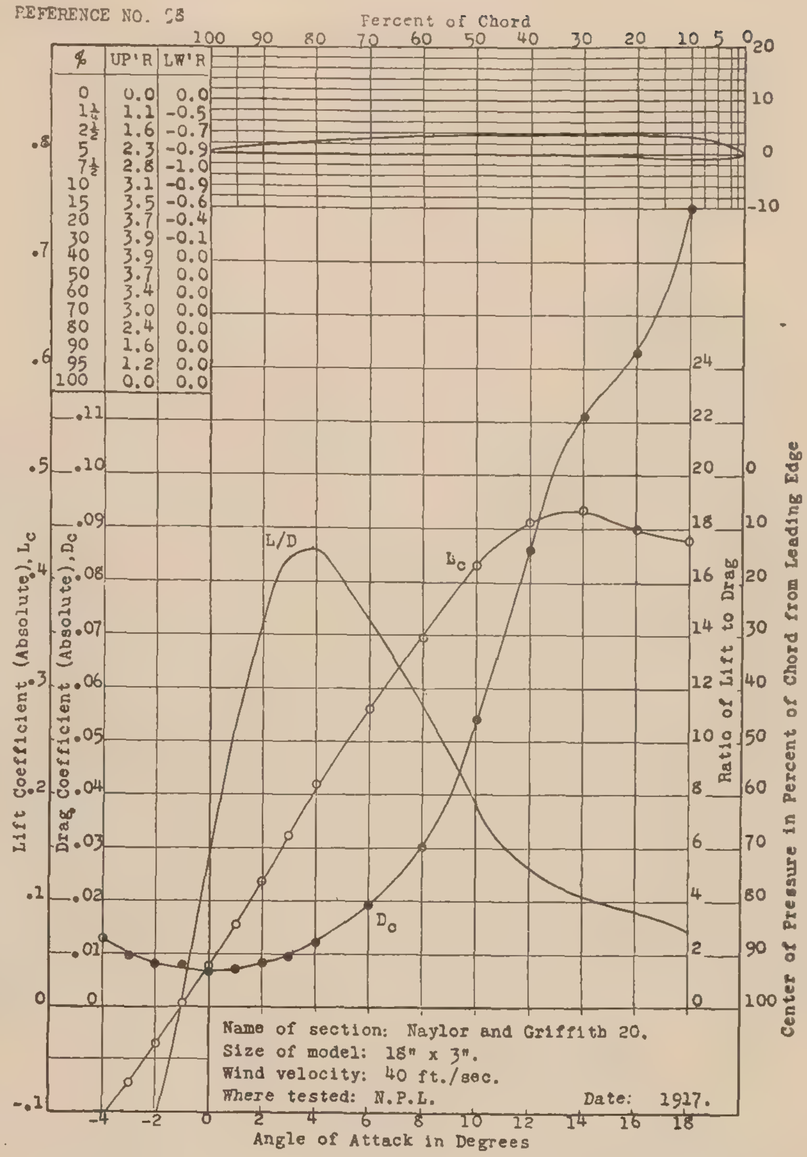
REFERENCE NO. 96



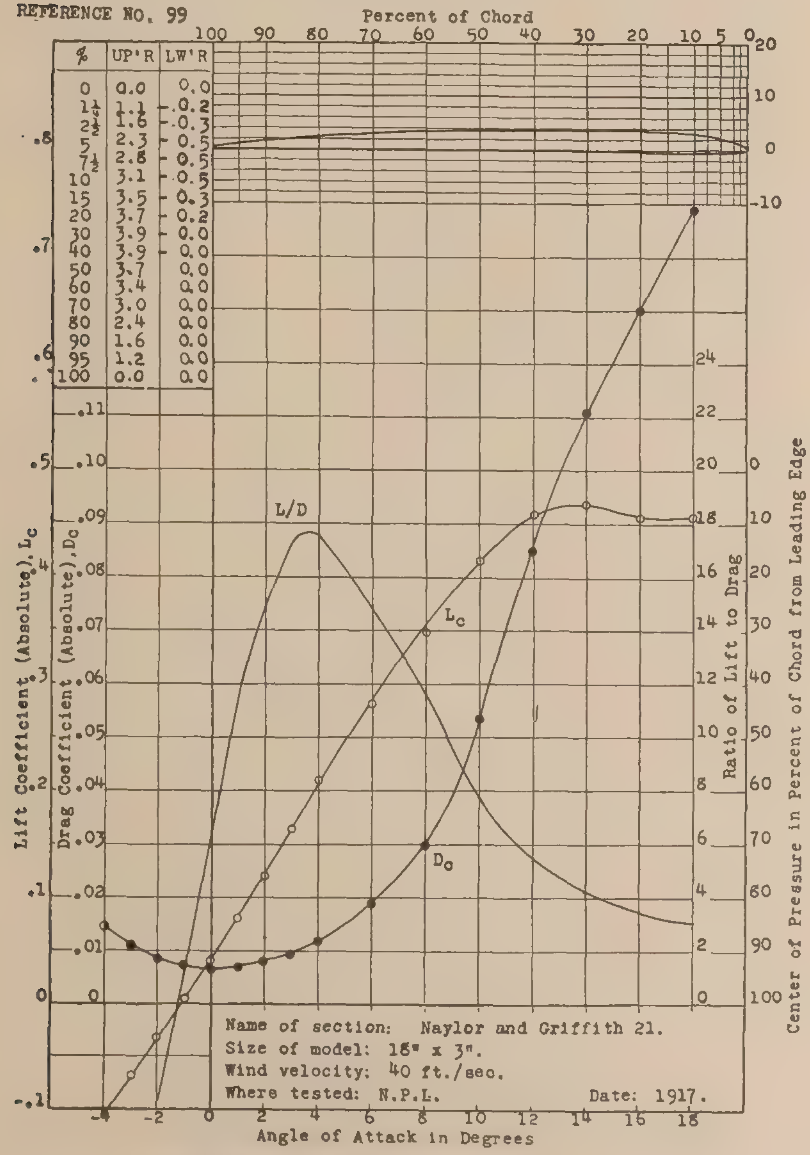
REFERENCE NO. 97



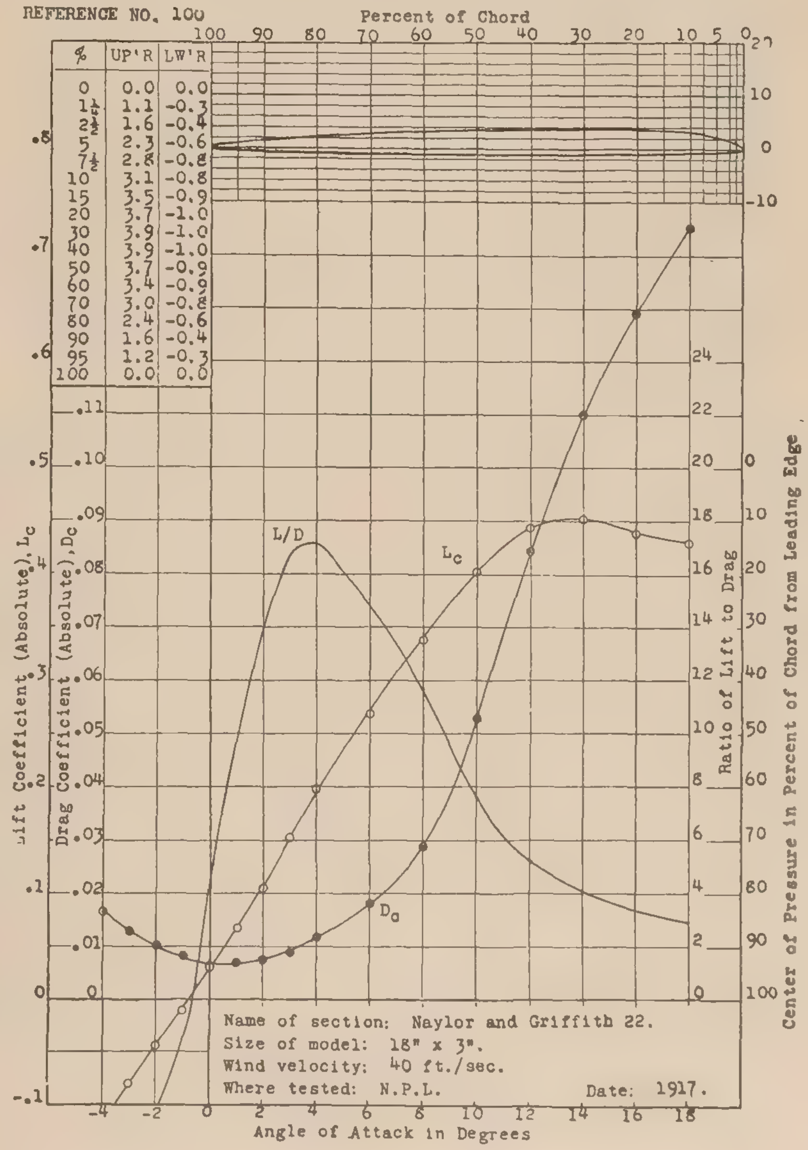
REFERENCE NO. 98



REFERENCE NO. 99

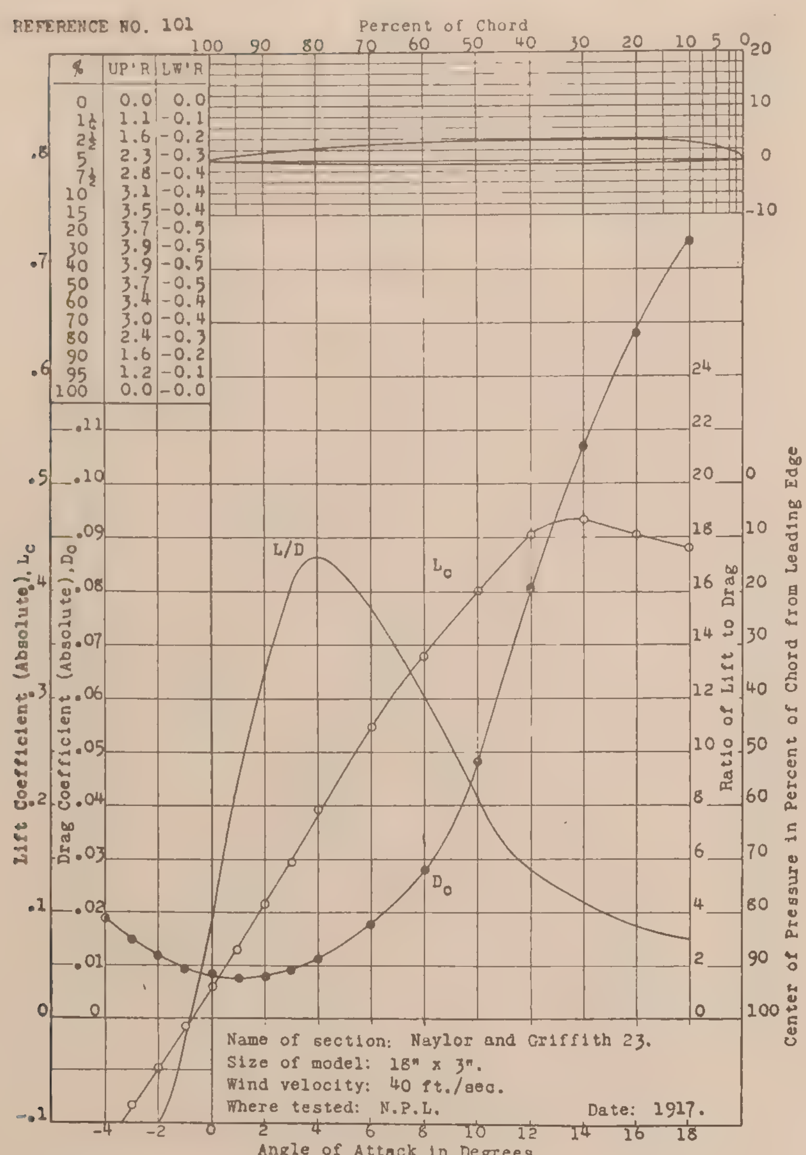


REFERENCE NO. 100



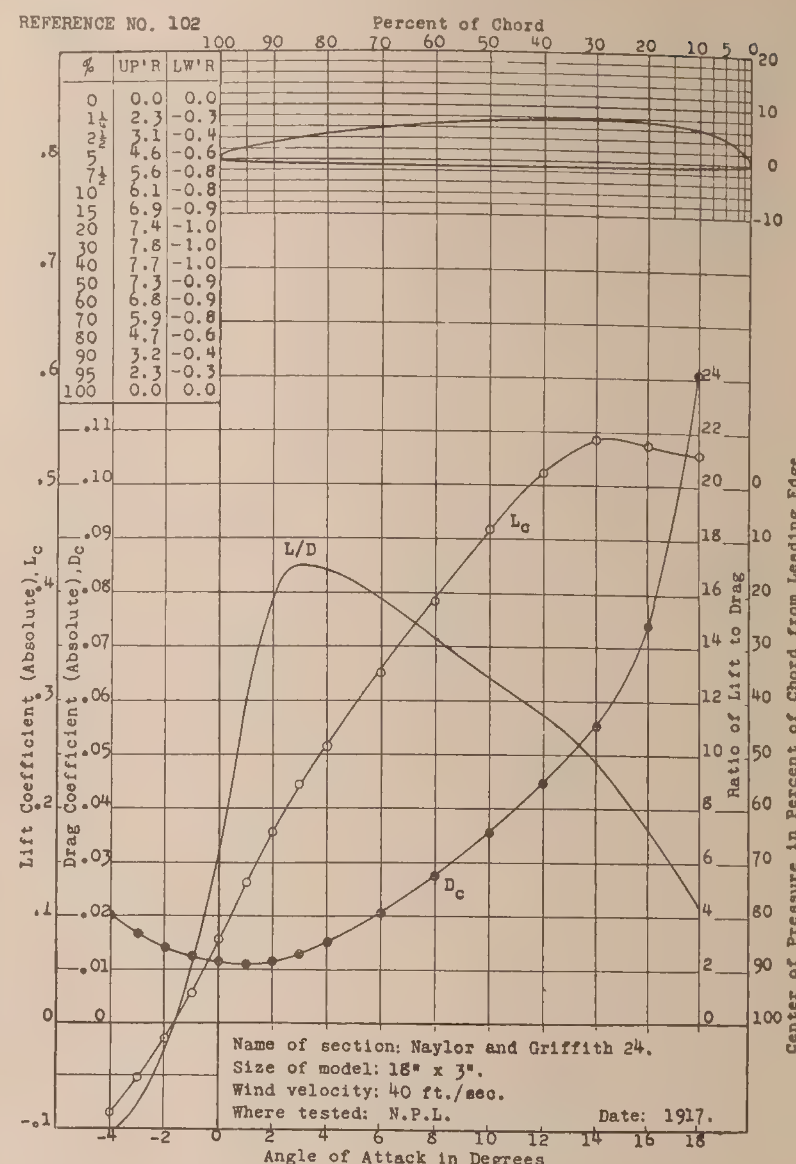
54889—21——20

REFERENCE NO. 101



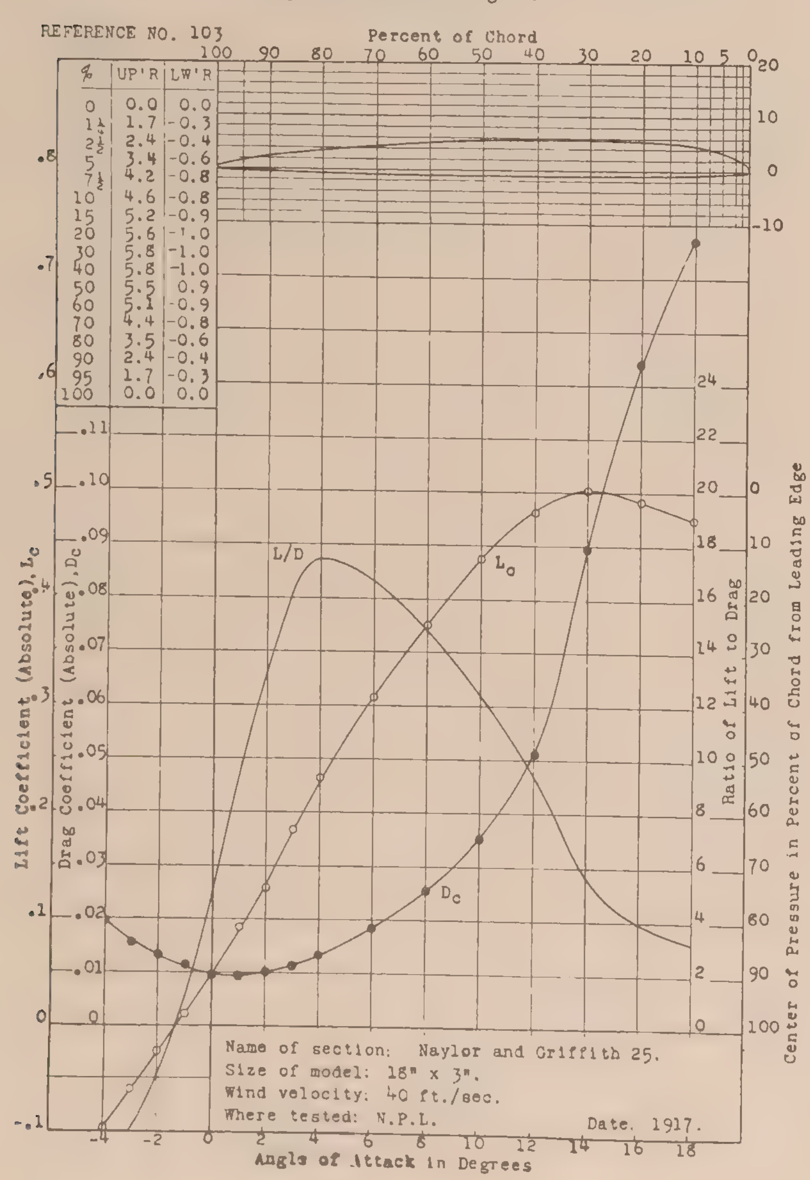
%	UP'R	LW'R
0	0.0	0.0
1¼	1.1	-0.1
2½	1.6	-0.2
5	2.3	-0.3
7½	2.8	-0.4
10	3.1	-0.4
15	3.5	-0.4
20	3.7	-0.5
30	3.9	-0.5
40	3.9	-0.5
50	3.7	-0.5
60	3.4	-0.4
70	3.0	-0.4
80	2.4	-0.3
90	1.6	-0.2
95	1.2	-0.1
100	0.0	-0.0

REFERENCE NO. 102



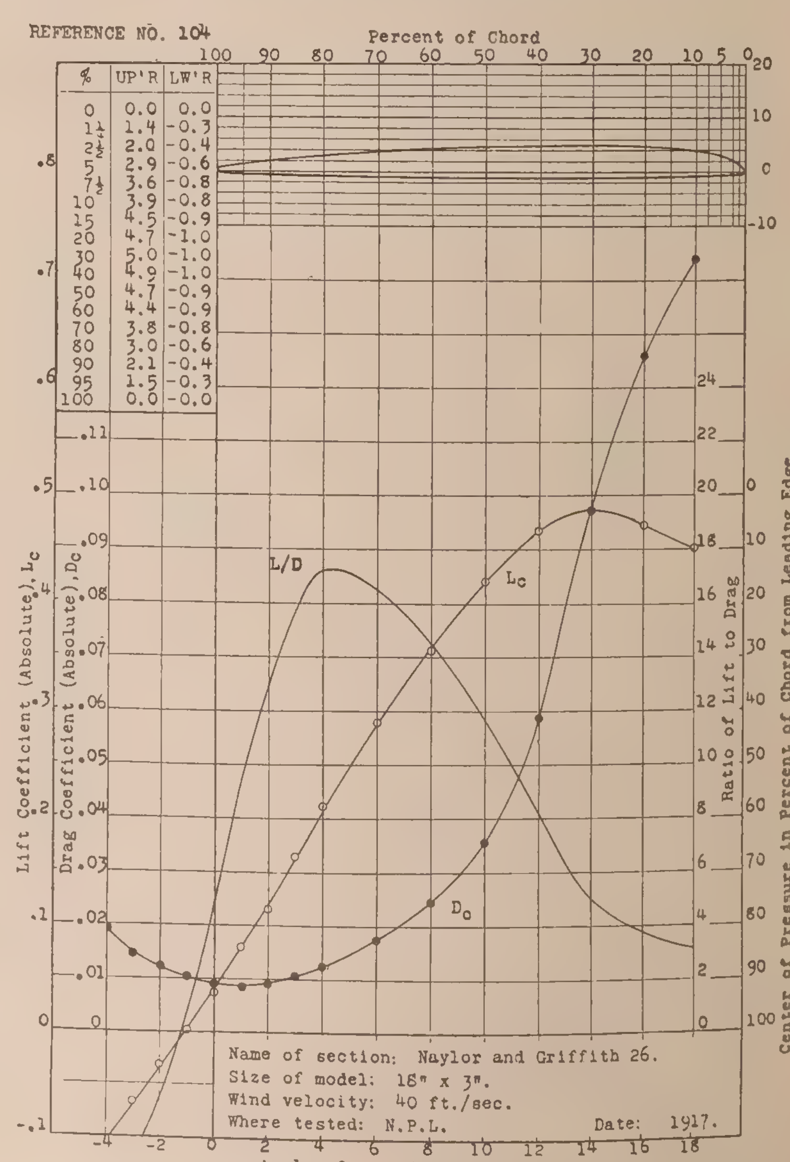
%	UP'R	LW'R
0	0.0	0.0
1¼	2.3	-0.3
2½	3.1	-0.4
5	4.6	-0.6
7½	5.6	-0.8
10	6.1	-0.8
15	6.9	-0.9
20	7.4	-1.0
30	7.8	-1.0
40	7.7	-1.0
50	7.3	-0.9
60	6.8	-0.9
70	5.9	-0.8
80	4.7	-0.6
90	3.2	-0.4
95	2.3	-0.3
100	0.0	0.0

REFERENCE NO. 103



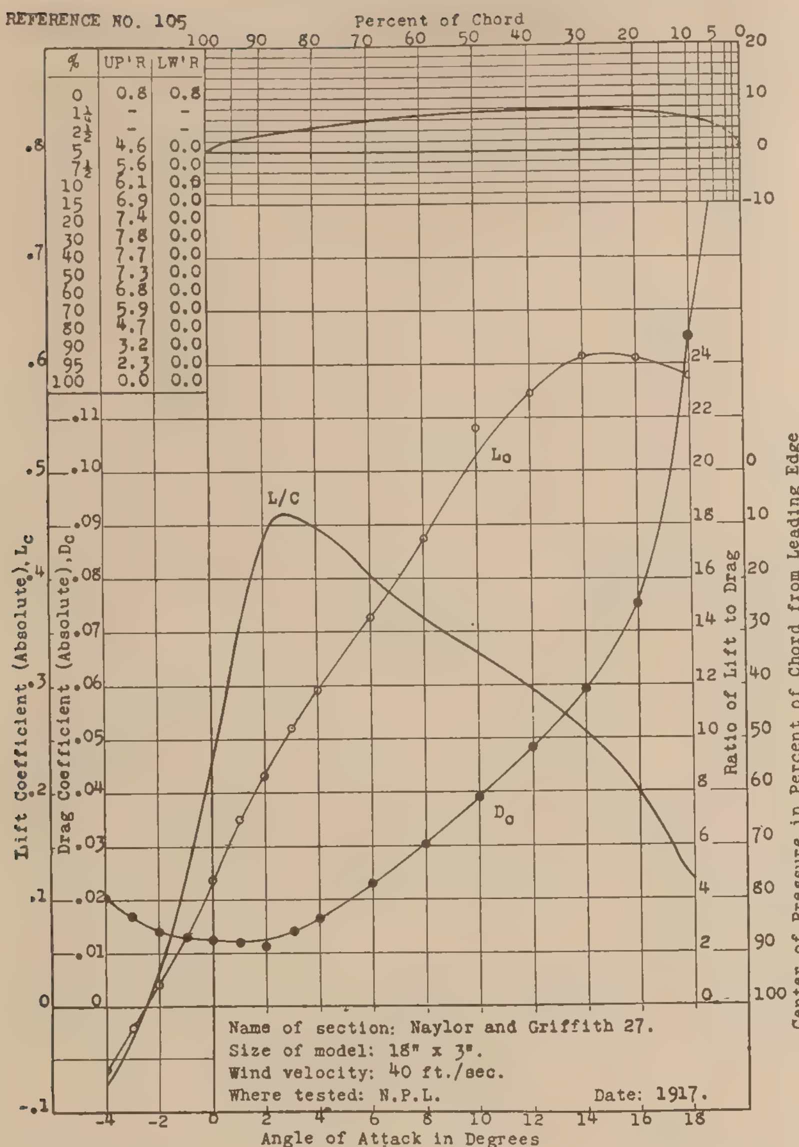
%	UP'R	LW'R
0	0.0	0.0
1¼	1.7	-0.3
2½	2.4	-0.4
5	3.4	-0.6
7½	4.2	-0.8
10	4.6	-0.8
15	5.2	-0.9
20	5.6	-1.0
30	5.8	-1.0
40	5.8	-1.0
50	5.5	0.9
60	5.1	-0.9
70	4.4	-0.8
80	3.5	-0.6
90	2.4	-0.4
95	1.7	-0.3
100	0.0	0.0

REFERENCE NO. 104



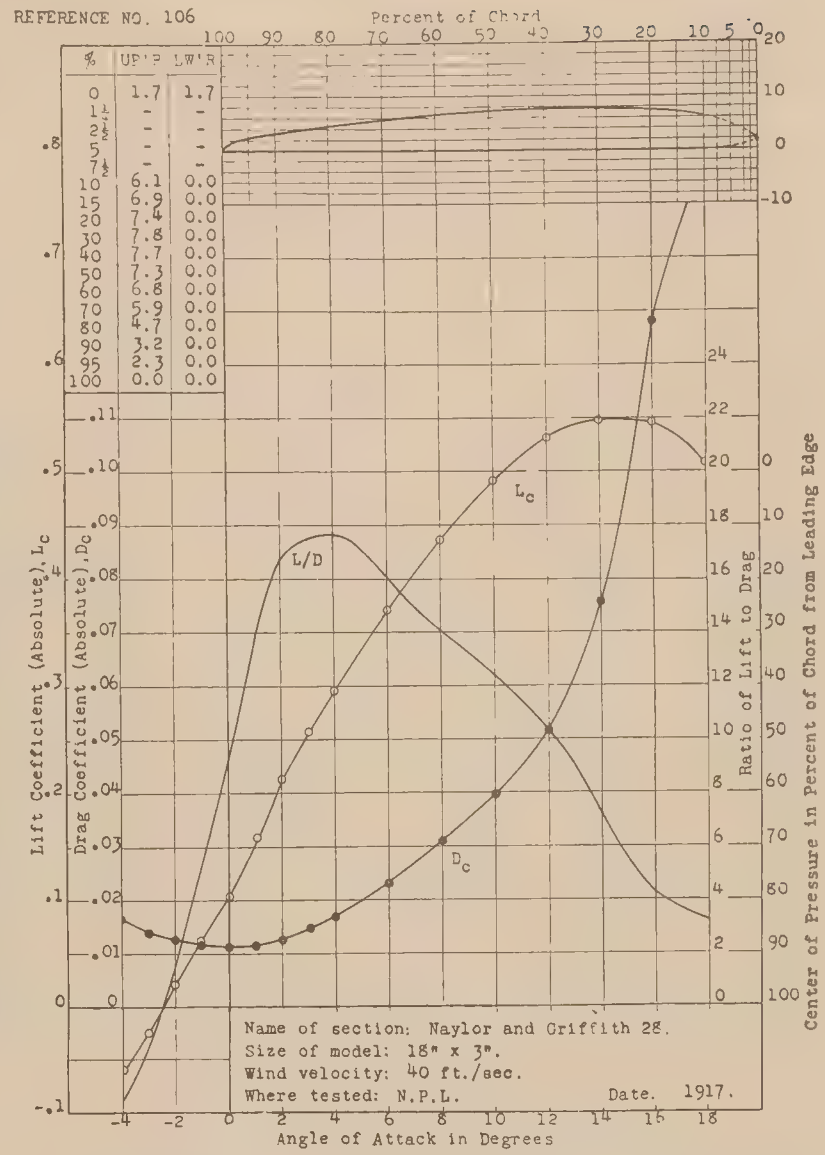
%	UP'R	LW'R
0	0.0	0.0
1¼	1.4	-0.3
2½	2.0	-0.4
5	2.9	-0.6
7½	3.6	-0.8
10	3.9	-0.8
15	4.5	-0.9
20	4.7	-1.0
30	5.0	-1.0
40	4.9	-1.0
50	4.7	-0.9
60	4.4	-0.9
70	3.8	-0.8
80	3.0	-0.6
90	2.1	-0.4
95	1.5	-0.3
100	0.0	-0.0

REFERENCE NO. 105



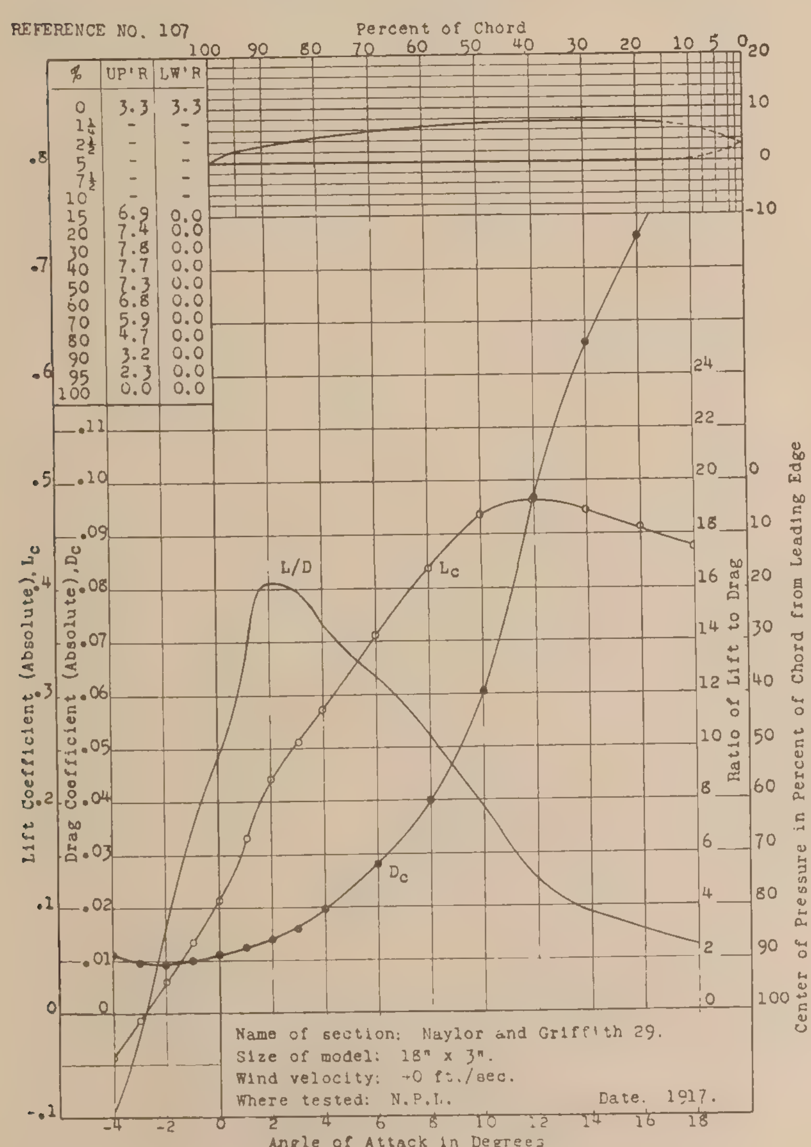
%	UP'R	LW'R
0	0.8	0.8
1¼	-	-
2½	-	-
5	4.6	0.0
7½	5.6	0.0
10	6.1	0.0
15	6.9	0.0
20	7.4	0.0
30	7.8	0.0
40	7.7	0.0
50	7.3	0.0
60	6.8	0.0
70	5.9	0.0
80	4.7	0.0
90	3.2	0.0
95	2.3	0.0
100	0.0	0.0

REFERENCE NO. 106



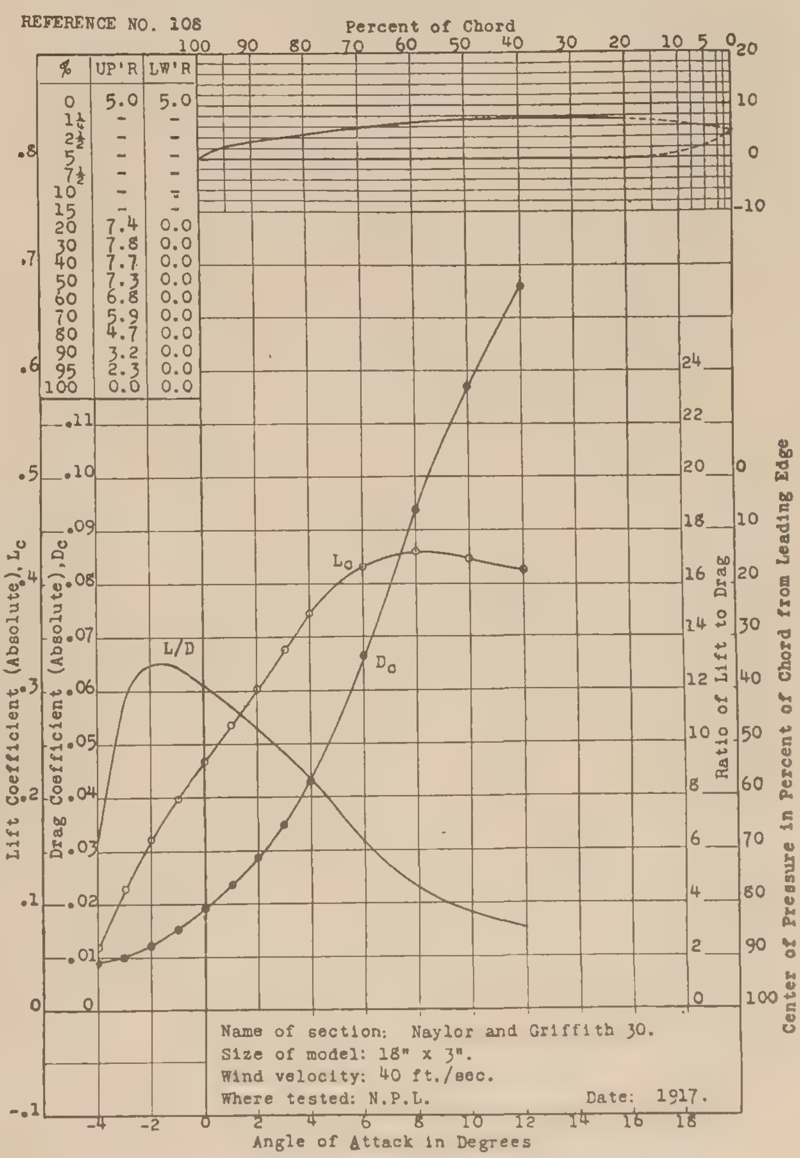
%	UP'R	LW'R
0	1.7	1.7
1¼	-	-
2½	-	-
5	-	-
7½	-	-
10	6.1	0.0
15	6.9	0.0
20	7.4	0.0
30	7.8	0.0
40	7.7	0.0
50	7.3	0.0
60	6.8	0.0
70	5.9	0.0
80	4.7	0.0
90	3.2	0.0
95	2.3	0.0
100	0.0	0.0

REFERENCE NO. 107



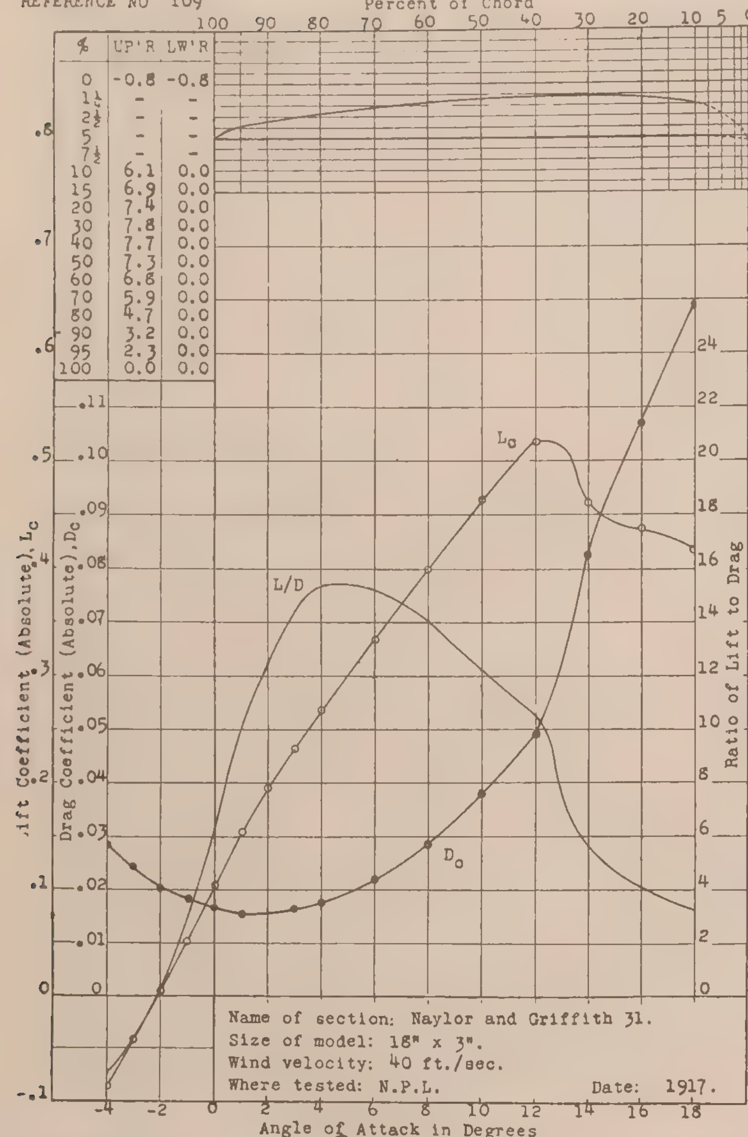
%	UP'R	LW'R
0	3.3	3.3
1¼	-	-
2½	-	-
5	-	-
7½	-	-
10	-	-
15	6.9	0.0
20	7.4	0.0
30	7.8	0.0
40	7.7	0.0
50	7.3	0.0
60	6.8	0.0
70	5.9	0.0
80	4.7	0.0
90	3.2	0.0
95	2.3	0.0
100	0.0	0.0

REFERENCE NO. 108



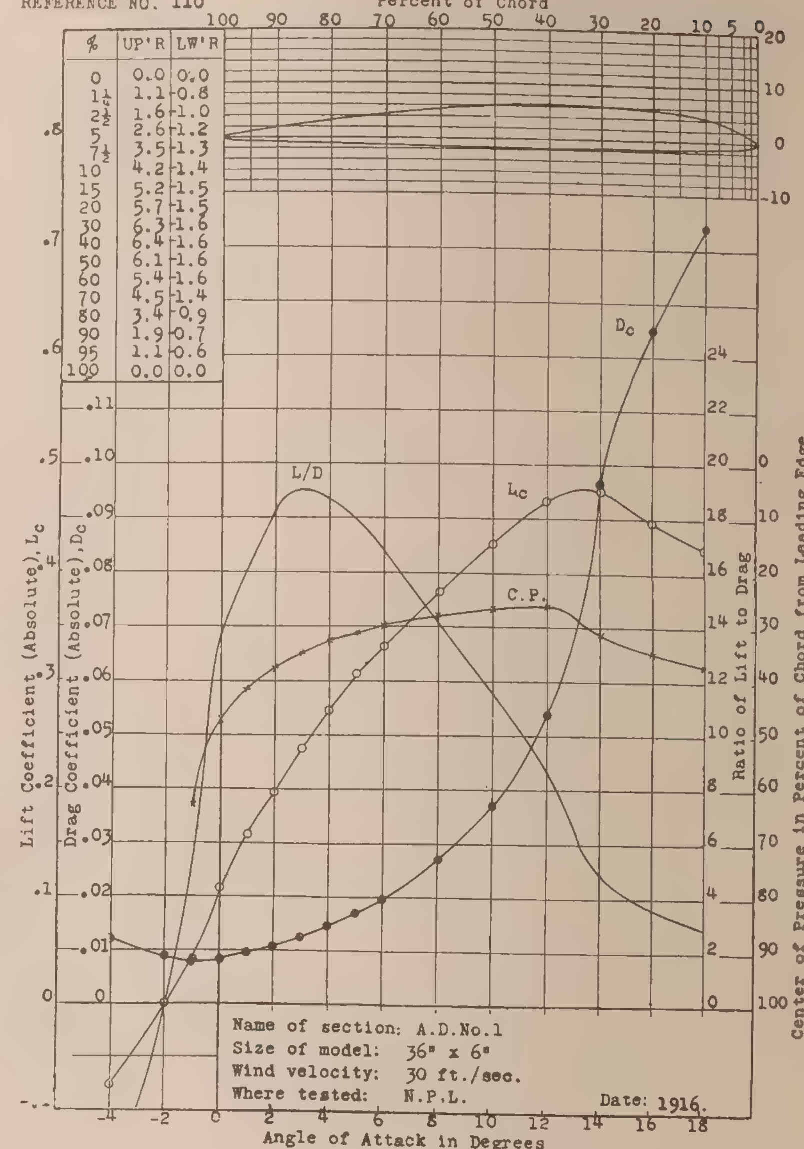
%	UP'R	LW'R
0	5.0	5.0
1¼	-	-
2½	-	-
5	-	-
7½	-	-
10	-	-
15	-	-
20	7.4	0.0
30	7.8	0.0
40	7.7	0.0
50	7.3	0.0
60	6.8	0.0
70	5.9	0.0
80	4.7	0.0
90	3.2	0.0
95	2.3	0.0
100	0.0	0.0

REFERENCE NO 109



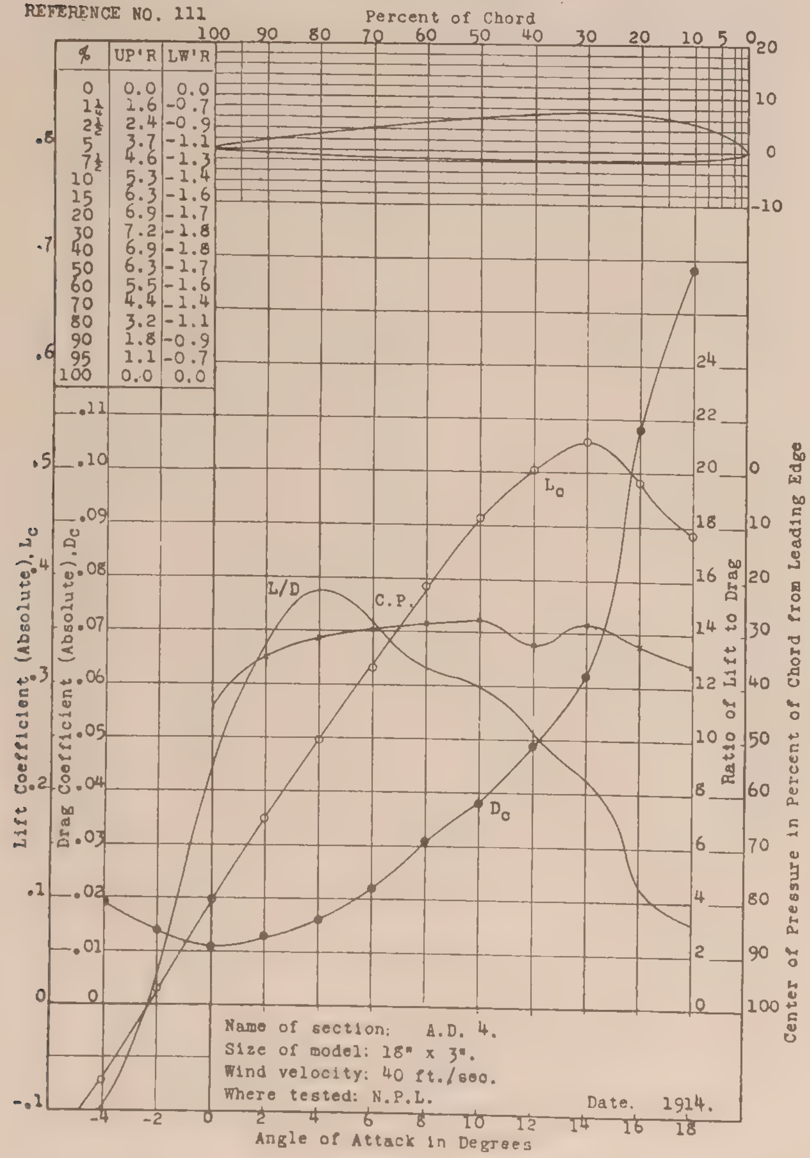
%	UP'R	LW'R
0	-0.8	-0.8
$1\frac{1}{4}$	-	-
$2\frac{1}{2}$	-	-
5	-	-
$7\frac{1}{2}$	-	-
10	6.1	0.0
15	6.9	0.0
20	7.4	0.0
30	7.8	0.0
40	7.7	0.0
50	7.3	0.0
60	6.8	0.0
70	5.9	0.0
80	4.7	0.0
90	3.2	0.0
95	2.3	0.0
100	0.0	0.0

REFERENCE NO. 110



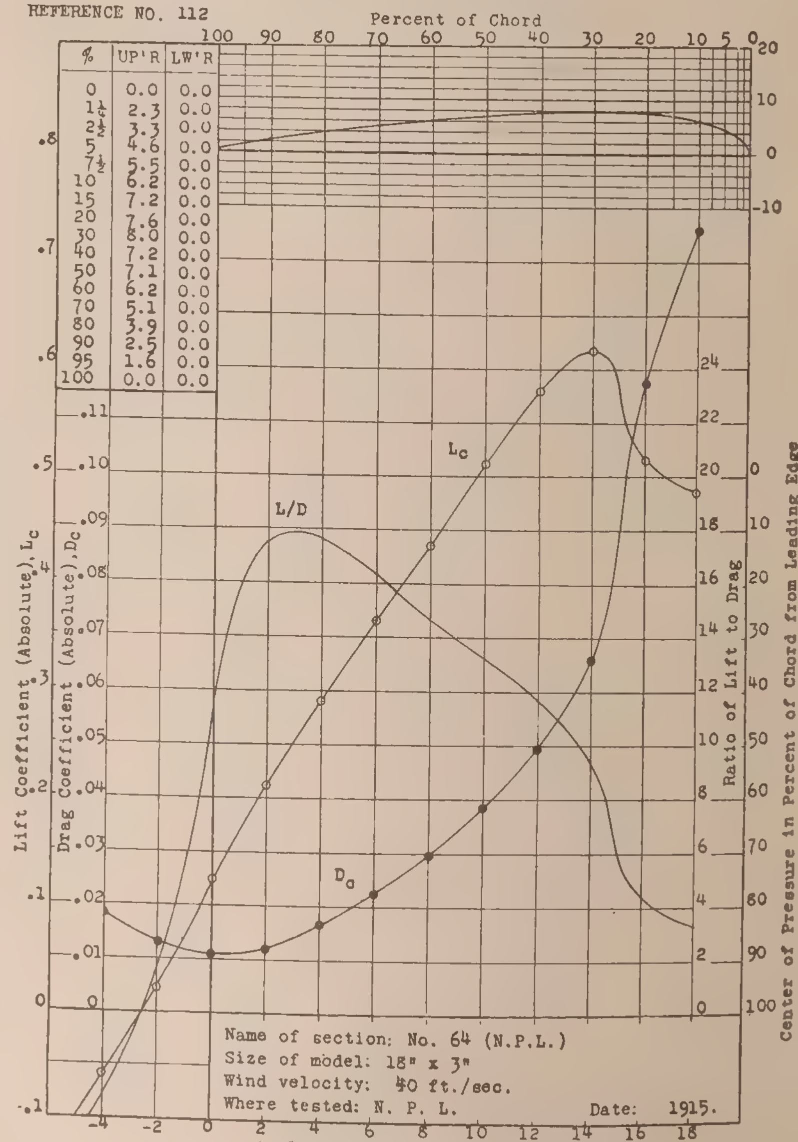
%	UP'R	LW'R
0	0.0	0.0
$1\frac{1}{4}$	1.1	-0.8
$2\frac{1}{2}$	1.6	-1.0
5	2.6	-1.2
$7\frac{1}{2}$	3.5	-1.3
10	4.2	-1.4
15	5.2	-1.5
20	5.7	-1.5
30	6.3	-1.6
40	6.4	-1.6
50	6.1	-1.6
60	5.4	-1.6
70	4.5	-1.4
80	3.4	-0.9
90	1.9	-0.7
95	1.1	-0.6
100	0.0	0.0

REFERENCE NO. 111



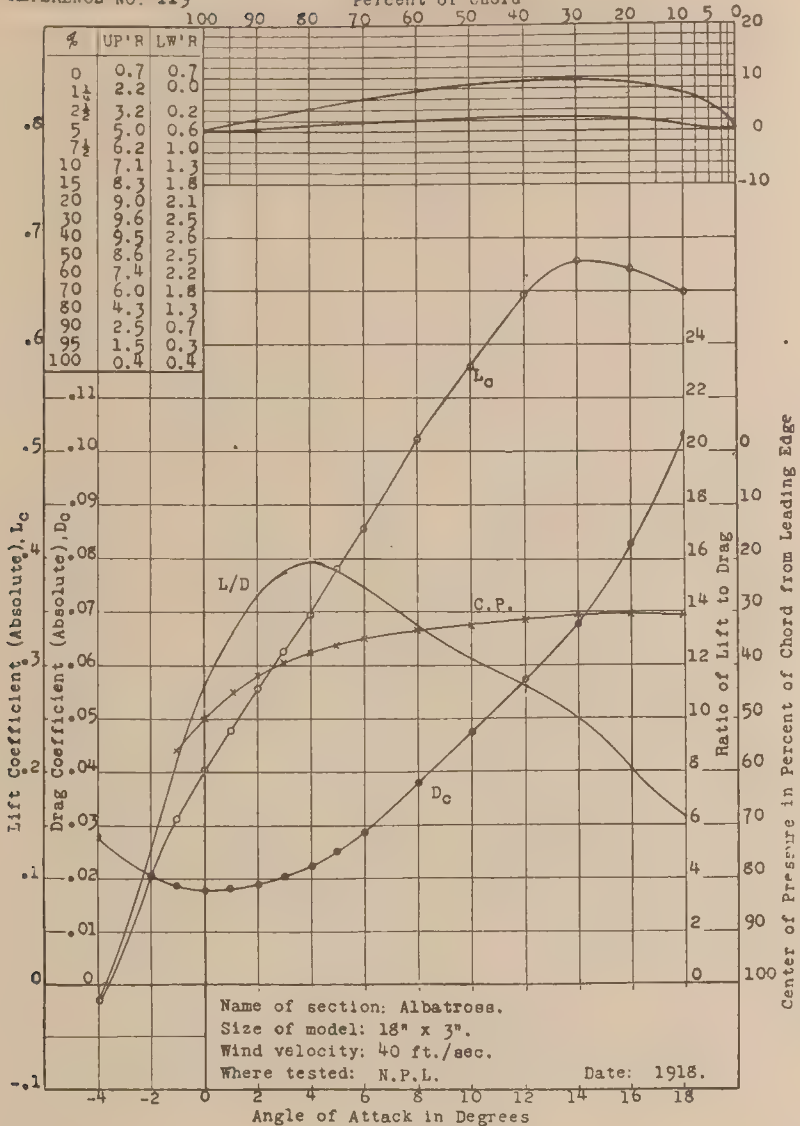
%	UP'R	LW'R
0	0.0	0.0
$1\frac{1}{4}$	1.6	-0.7
$2\frac{1}{2}$	2.4	-0.9
5	3.7	-1.1
$7\frac{1}{2}$	4.6	-1.3
10	5.3	-1.4
15	6.3	-1.6
20	6.9	-1.7
30	7.2	-1.8
40	6.9	-1.8
50	6.3	-1.7
60	5.5	-1.6
70	4.4	-1.4
80	3.2	-1.1
90	1.8	-0.9
95	1.1	-0.7
100	0.0	0.0

REFERENCE NO. 112



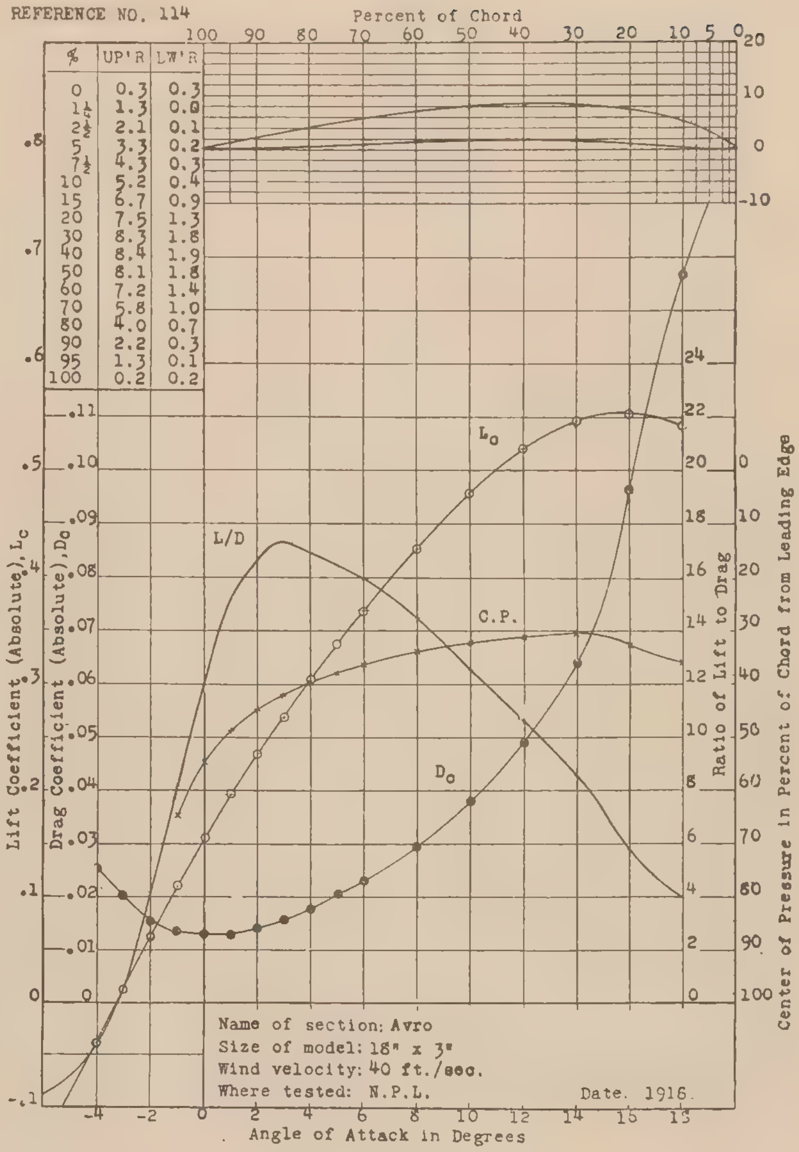
%	UP'R	LW'R
0	0.0	0.0
$1\frac{1}{4}$	2.3	0.0
$2\frac{1}{2}$	3.3	0.0
5	4.6	0.0
$7\frac{1}{2}$	5.5	0.0
10	6.2	0.0
15	7.2	0.0
20	7.6	0.0
30	8.0	0.0
40	7.2	0.0
50	7.1	0.0
60	6.2	0.0
70	5.1	0.0
80	3.9	0.0
90	2.5	0.0
95	1.6	0.0
100	0.0	0.0

REFERENCE NO. 113



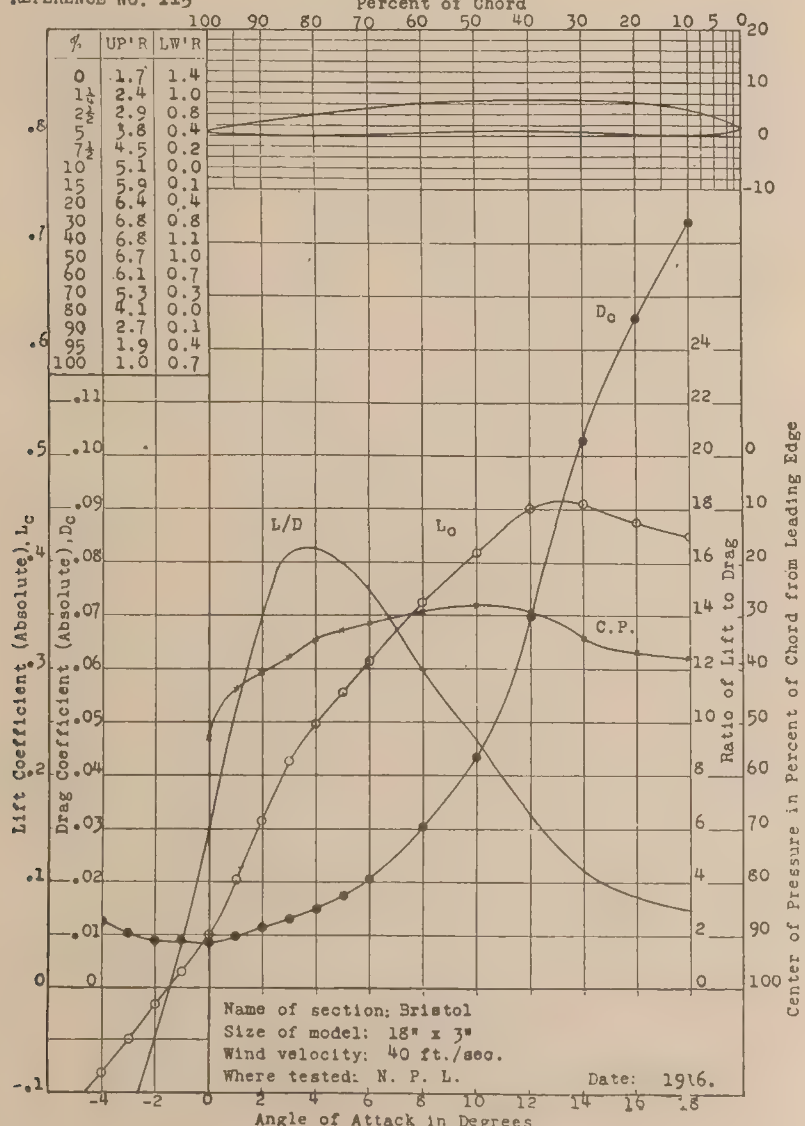
%	UP'R	LW'R
0	0.7	0.7
1¼	2.2	0.0
2½	3.2	0.2
5	5.0	0.6
7½	6.2	1.0
10	7.1	1.3
15	8.3	1.8
20	9.0	2.1
30	9.6	2.5
40	9.5	2.6
50	8.6	2.5
60	7.4	2.2
70	6.0	1.8
80	4.3	1.3
90	2.5	0.7
95	1.5	0.3
100	0.4	0.4

REFERENCE NO. 114



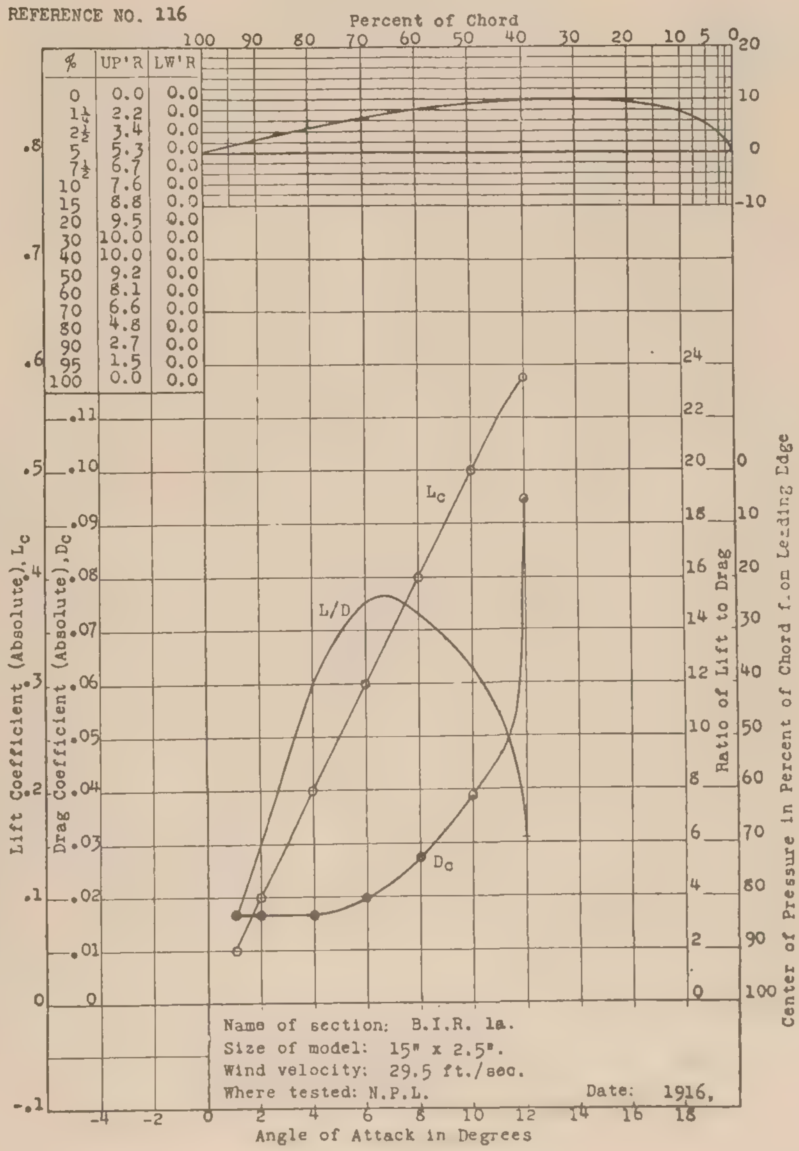
%	UP'R	LW'R
0	0.3	0.3
1¼	1.3	0.0
2½	2.1	0.1
5	3.3	0.2
7½	4.3	0.3
10	5.2	0.4
15	6.7	0.9
20	7.5	1.3
30	8.3	1.8
40	8.4	1.9
50	8.1	1.8
60	7.2	1.4
70	5.8	1.0
80	4.0	0.7
90	2.2	0.3
95	1.3	0.1
100	0.2	0.2

REFERENCE NO. 115



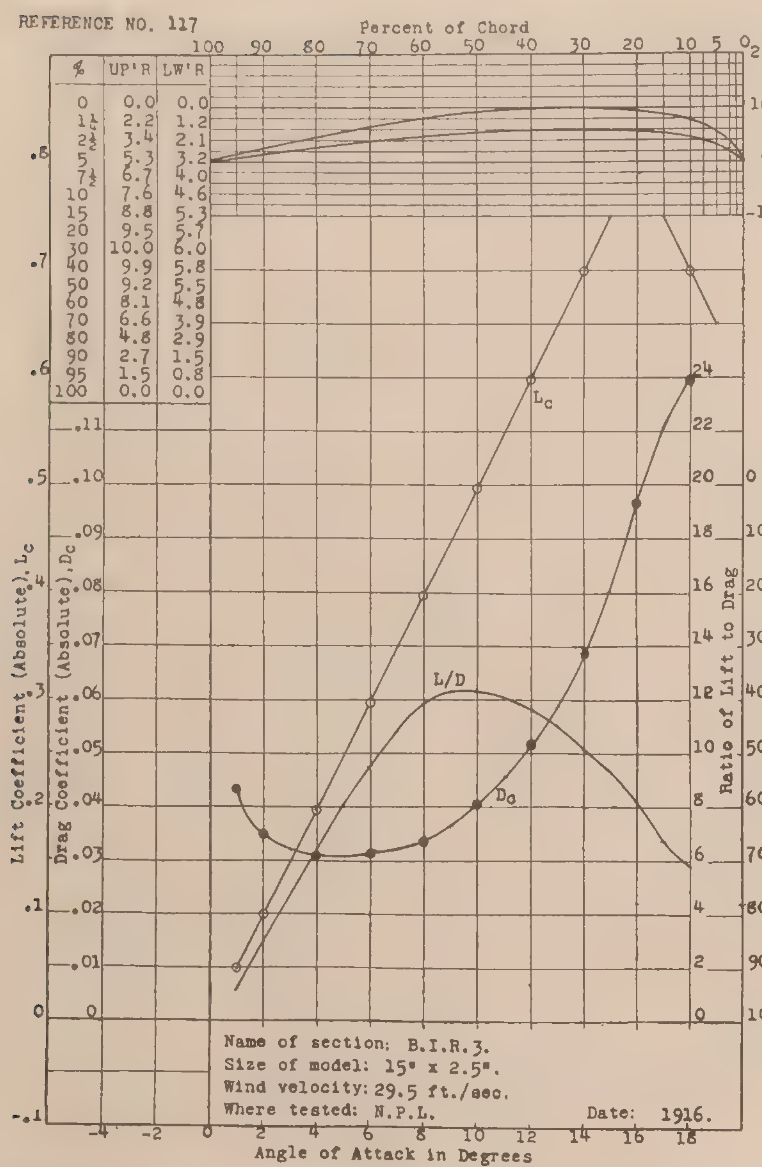
%	UP'R	LW'R
0	1.7	1.4
1¼	2.4	1.0
2½	2.9	0.8
5	3.8	0.4
7½	4.5	0.2
10	5.1	0.0
15	5.9	0.1
20	6.4	0.4
30	6.8	0.8
40	6.8	1.1
50	6.7	1.0
60	6.1	0.7
70	5.3	0.3
80	4.1	0.0
90	2.7	0.1
95	1.9	0.4
100	1.0	0.7

REFERENCE NO. 116



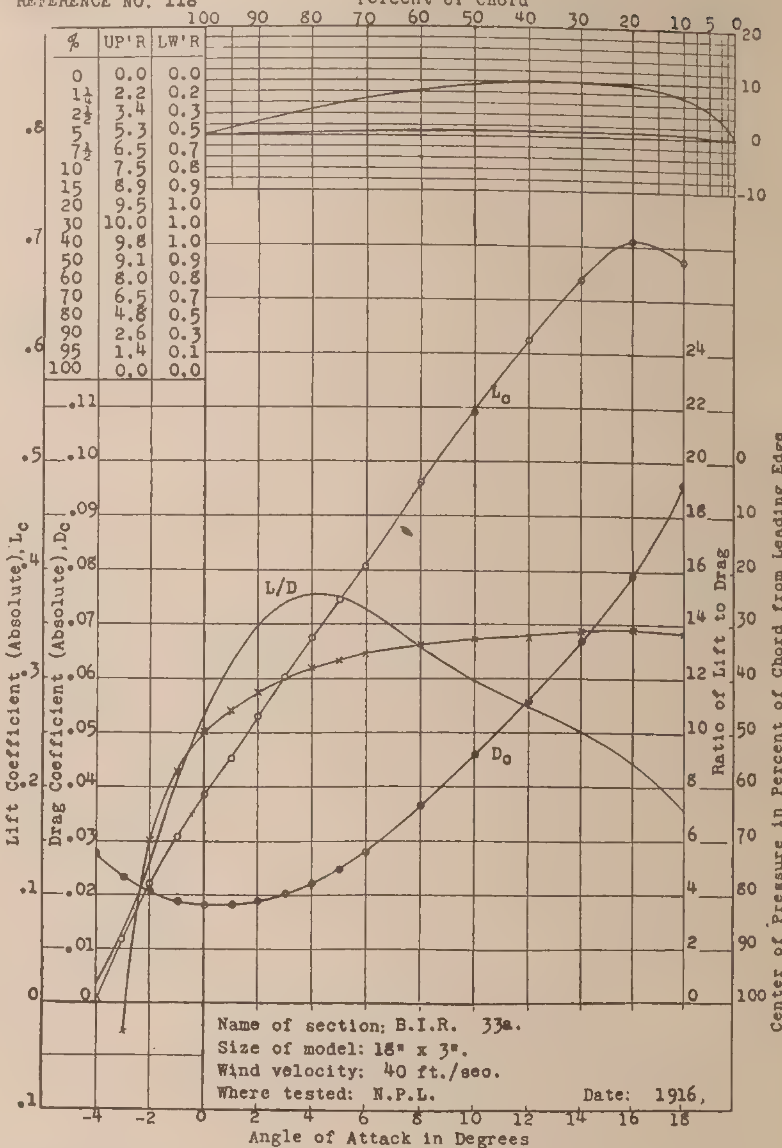
%	UP'R	LW'R
0	0.0	0.0
1¼	2.2	0.0
2½	3.4	0.0
5	5.3	0.0
7½	6.7	0.0
10	7.6	0.0
15	8.8	0.0
20	9.5	0.0
30	10.0	0.0
40	10.0	0.0
50	9.2	0.0
60	8.1	0.0
70	6.6	0.0
80	4.8	0.0
90	2.7	0.0
95	1.5	0.0
100	0.0	0.0

REFERENCE NO. 117



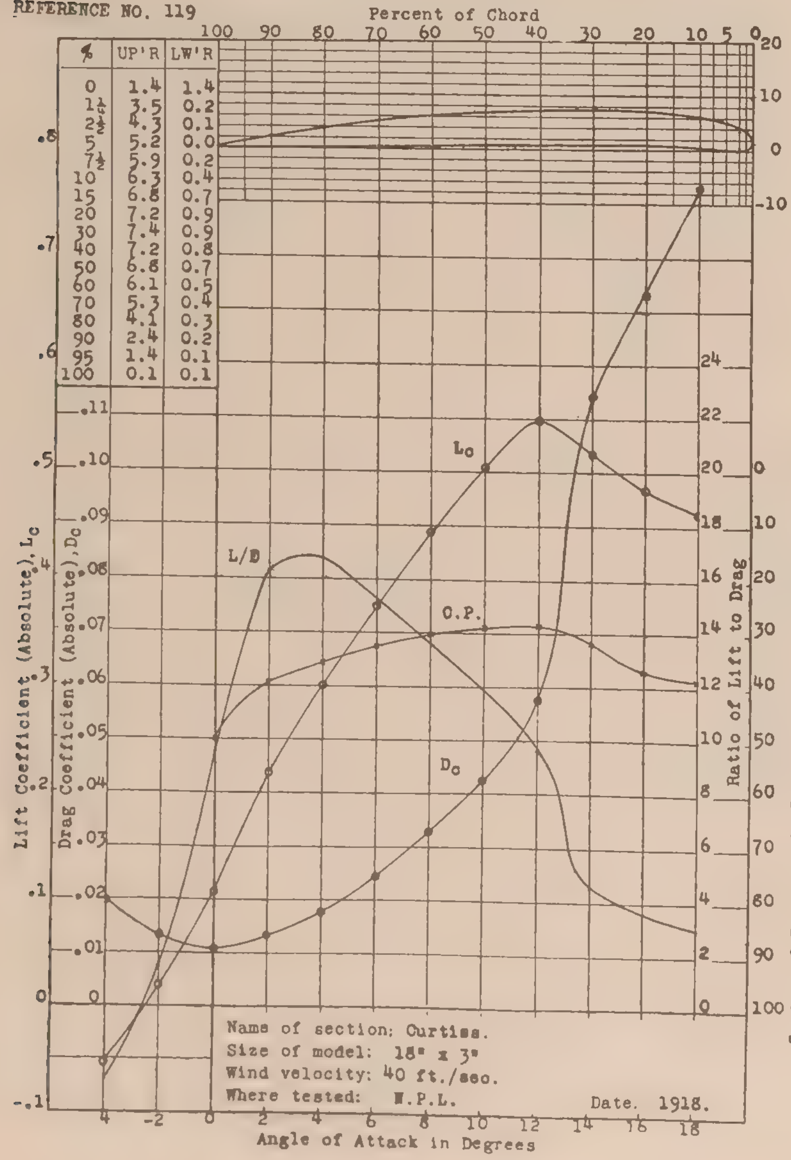
%	UP'R	LW'R
0	0.0	0.0
1¼	2.2	1.2
2½	3.4	2.1
5	5.3	3.2
7½	6.7	4.0
10	7.6	4.6
15	8.8	5.3
20	9.5	5.7
30	10.0	6.0
40	9.9	5.8
50	9.2	5.5
60	8.1	4.8
70	6.6	3.9
80	4.8	2.9
90	2.7	1.5
95	1.5	0.8
100	0.0	0.0

REFERENCE NO. 118



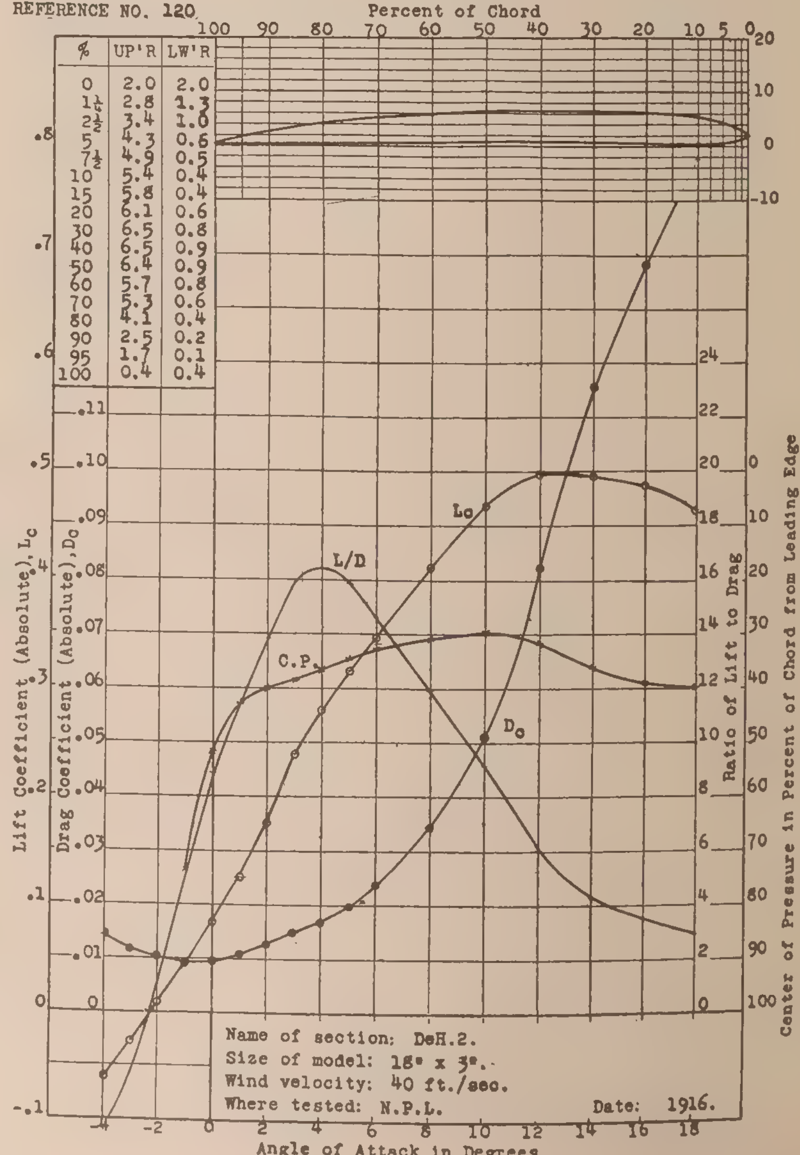
%	UP'R	LW'R
0	0.0	0.0
1¼	2.2	0.2
2½	3.4	0.3
5	5.3	0.5
7½	6.5	0.7
10	7.5	0.8
15	8.9	0.9
20	9.5	1.0
30	10.0	1.0
40	9.8	1.0
50	9.1	0.9
60	8.0	0.8
70	6.5	0.7
80	4.8	0.5
90	2.6	0.3
95	1.4	0.1
100	0.0	0.0

REFERENCE NO. 119

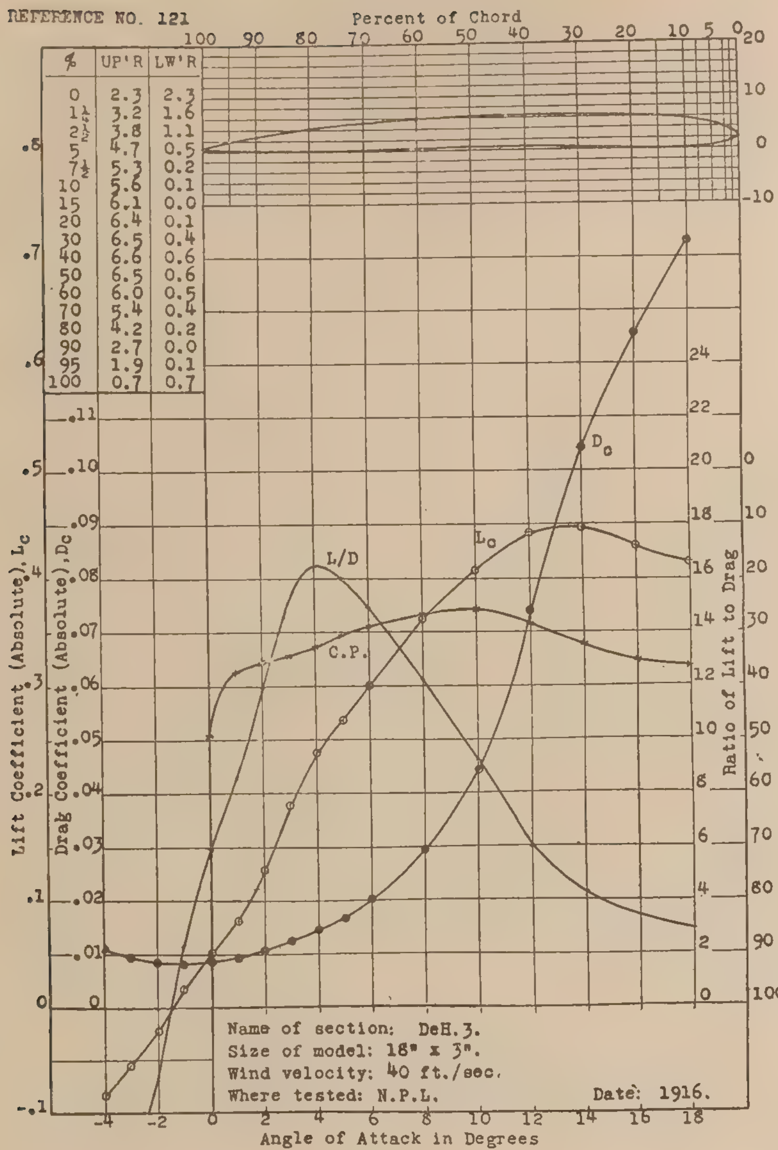


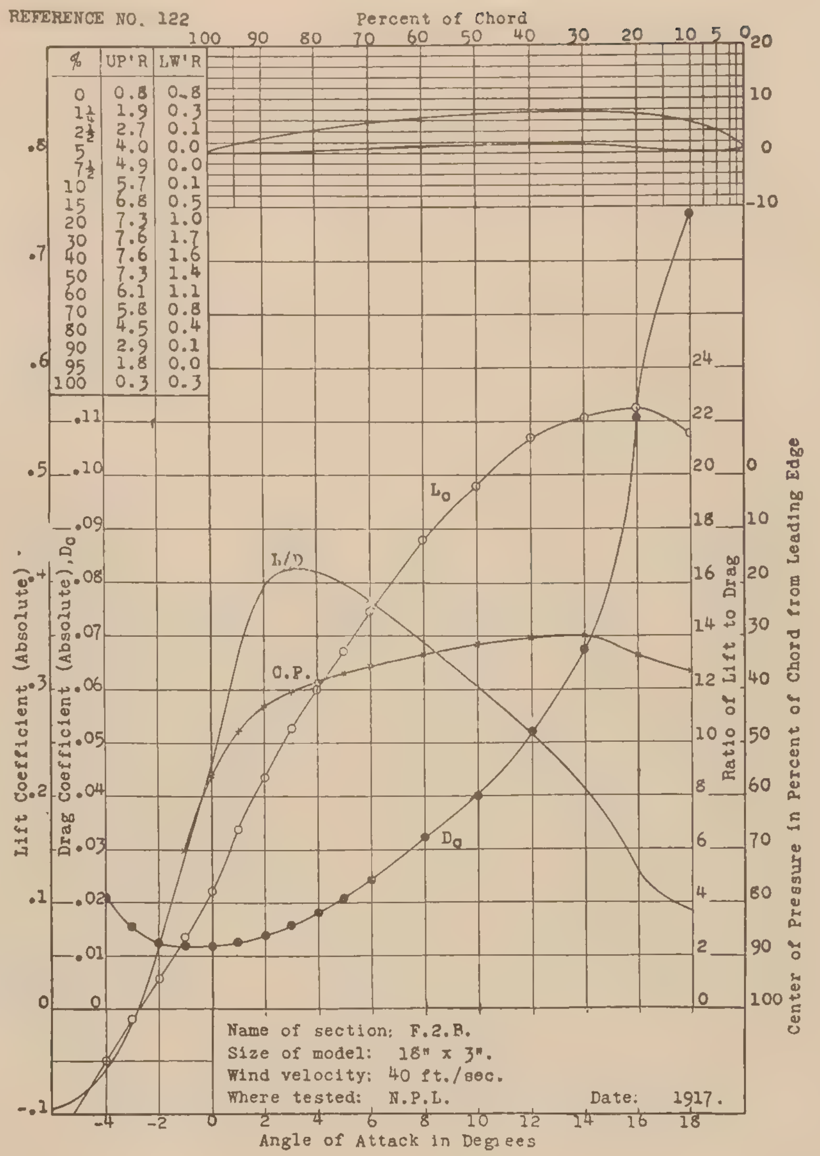
%	UP'R	LW'R
0	1.4	1.4
1¼	3.5	0.2
2½	4.3	0.1
5	5.2	0.0
7½	5.9	0.2
10	6.3	0.4
15	6.8	0.7
20	7.2	0.9
30	7.4	0.9
40	7.2	0.8
50	6.8	0.7
60	6.1	0.5
70	5.3	0.4
80	4.1	0.3
90	2.4	0.2
95	1.4	0.1
100	0.1	0.1

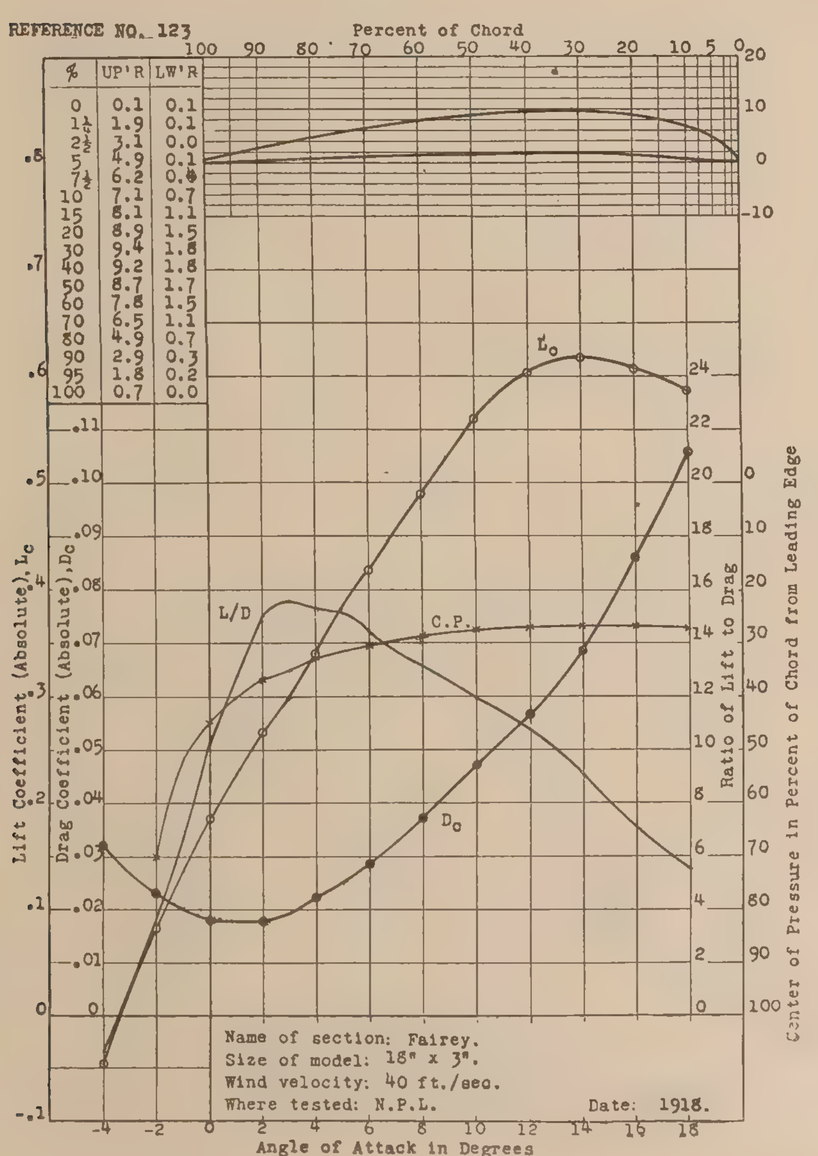
REFERENCE NO. 120

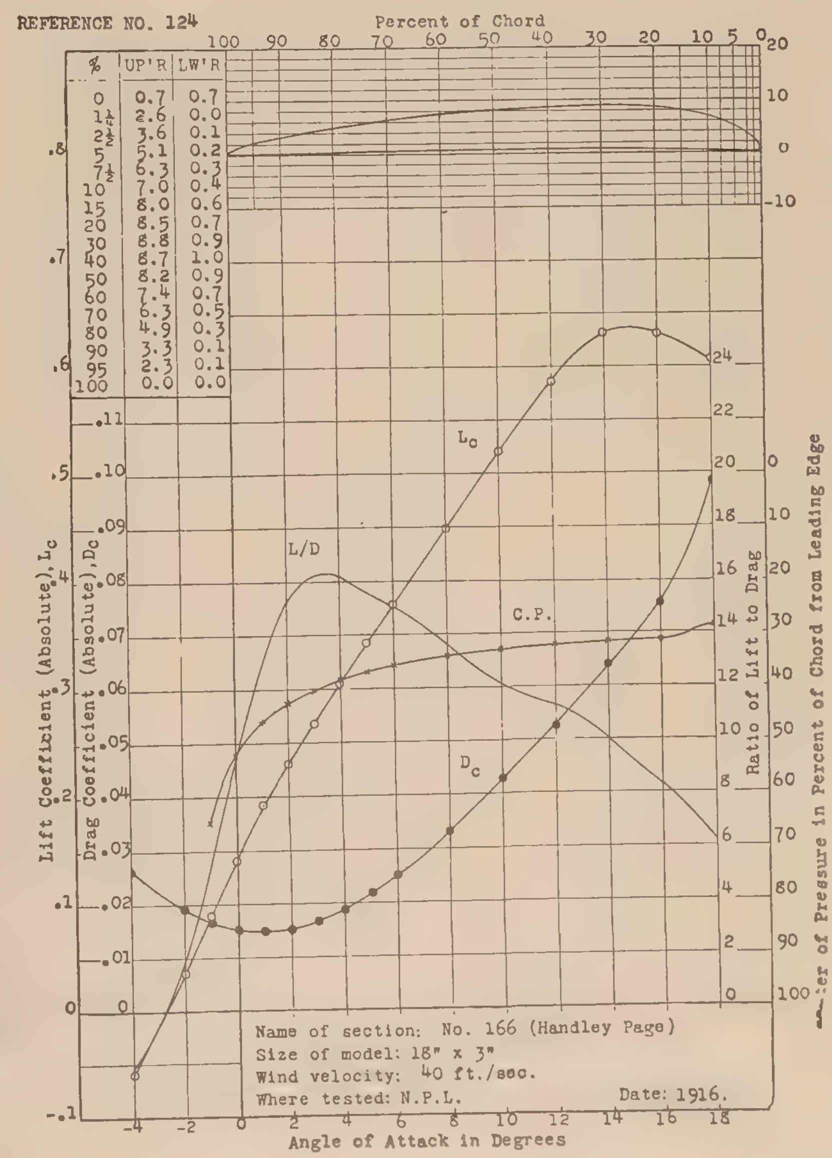


%	UP'R	LW'R
0	2.0	2.0
1¼	2.8	1.3
2½	3.4	1.0
5	4.3	0.6
7½	4.9	0.5
10	5.4	0.4
15	5.8	0.4
20	6.1	0.6
30	6.5	0.8
40	6.5	0.9
50	6.4	0.9
60	5.7	0.8
70	5.3	0.6
80	4.1	0.4
90	2.5	0.2
95	1.7	0.1
100	0.4	0.4

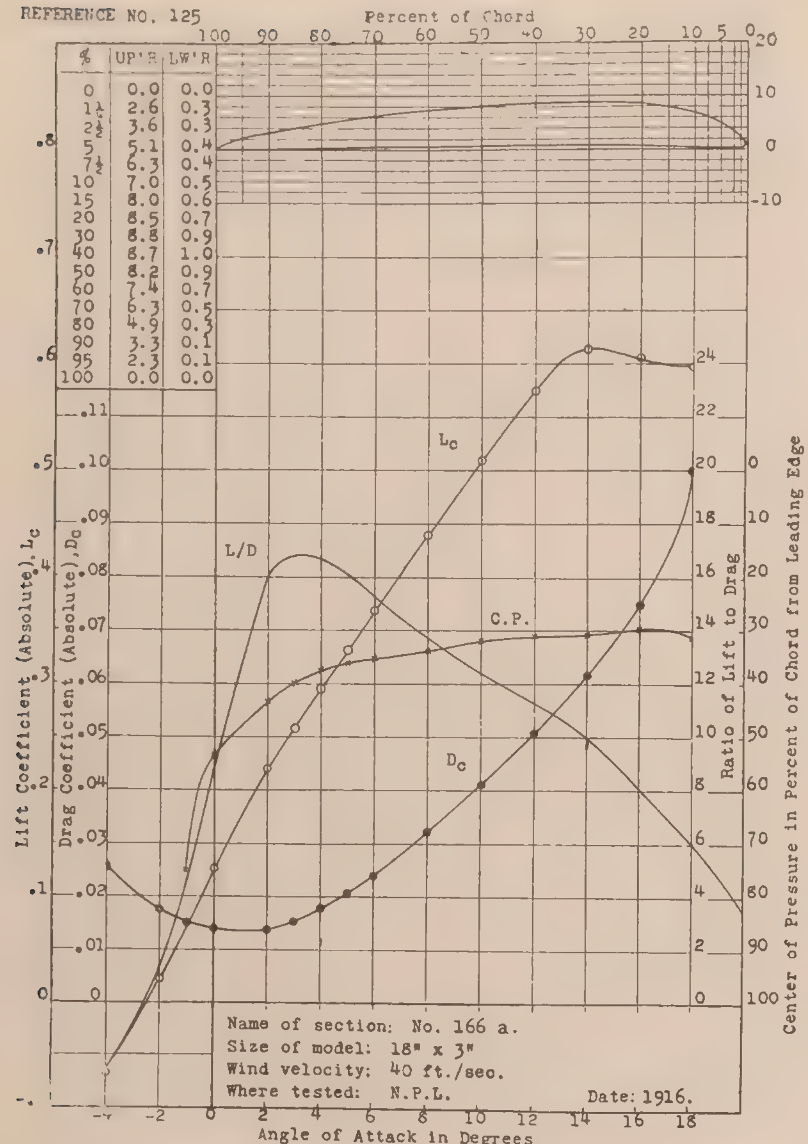

REFERENCE NO. 121
Percent of Chord
% UP'R LW'R
0 2.3 2.3
1½ 3.2 1.6
2½ 3.8 1.1
5 4.7 0.5
7½ 5.3 0.2
10 5.6 0.1
15 6.1 0.0
20 6.4 0.1
30 6.5 0.4
40 6.6 0.6
50 6.5 0.6
60 6.0 0.5
70 5.4 0.4
80 4.2 0.2
90 2.7 0.0
95 1.9 0.1
100 0.7 0.7
L/D
Lc
Dc
C.P.
Lift Coefficient (Absolute), Lc
Drag Coefficient (Absolute), Dc
Ratio of Lift to Drag
Center of Pressure in Percent of Chord from Leading Edge
Name of section: DeH.3.
Size of model: 18" x 3".
Wind velocity: 40 ft./sec.
Where tested: N.P.L.
Date: 1916.
Angle of Attack in Degrees


REFERENCE NO. 122
Percent of Chord
% UP'R LW'R
0 0.8 0.8
1½ 1.9 0.3
2½ 2.7 0.1
5 4.0 0.0
7½ 4.9 0.0
10 5.7 0.1
15 6.8 0.5
20 7.3 1.0
30 7.6 1.7
40 7.6 1.6
50 7.3 1.4
60 6.1 1.1
70 5.8 0.8
80 4.5 0.4
90 2.9 0.1
95 1.8 0.0
100 0.3 0.3
L/D
Lc
Dc
C.P.
Lift Coefficient (Absolute)
Drag Coefficient (Absolute), Dc
Ratio of Lift to Drag
Center of Pressure in Percent of Chord from Leading Edge
Name of section: F.2.B.
Size of model: 18" x 3".
Wind velocity: 40 ft./sec.
Where tested: N.P.L.
Date: 1917.
Angle of Attack in Degrees


REFERENCE NO. 123
Percent of Chord
% UP'R LW'R
0 0.1 0.1
1½ 1.9 0.1
2½ 3.1 0.0
5 4.9 0.1
7½ 6.2 0.4
10 7.1 0.7
15 8.1 1.1
20 8.9 1.5
30 9.4 1.8
40 9.2 1.8
50 8.7 1.7
60 7.8 1.5
70 6.5 1.1
80 4.9 0.7
90 2.9 0.3
95 1.8 0.2
100 0.7 0.0
L/D
Lc
Dc
C.P.
Lift Coefficient (Absolute), Lc
Drag Coefficient (Absolute), Dc
Ratio of Lift to Drag
Center of Pressure in Percent of Chord from Leading Edge
Name of section: Fairey.
Size of model: 18" x 3".
Wind velocity: 40 ft./sec.
Where tested: N.P.L.
Date: 1918.
Angle of Attack in Degrees

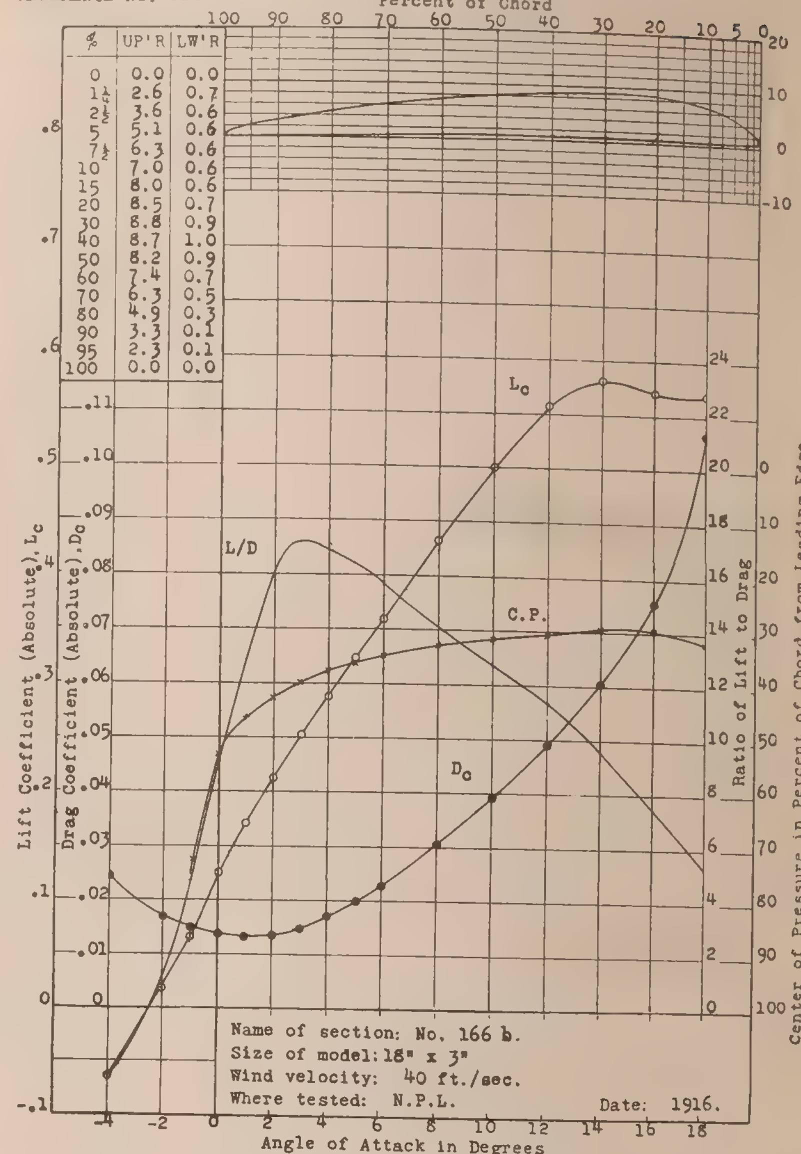

REFERENCE NO. 124
Percent of Chord
% UP'R LW'R
0 0.7 0.7
1½ 2.6 0.0
2½ 3.6 0.1
5 5.1 0.2
7½ 6.3 0.3
10 7.0 0.4
15 8.0 0.6
20 8.5 0.7
30 8.8 0.9
40 8.7 1.0
50 8.2 0.9
60 7.4 0.7
70 6.3 0.5
80 4.9 0.3
90 3.3 0.1
95 2.3 0.1
100 0.0 0.0
L/D
Lc
Dc
C.P.
Lift Coefficient (Absolute), Lc
Drag Coefficient (Absolute), Dc
Ratio of Lift to Drag
Name of section: No. 166 (Handley Page)
Size of model: 18" x 3"
Wind velocity: 40 ft./sec.
Where tested: N.P.L.
Date: 1916.
Angle of Attack in Degrees

REFERENCE NO. 125



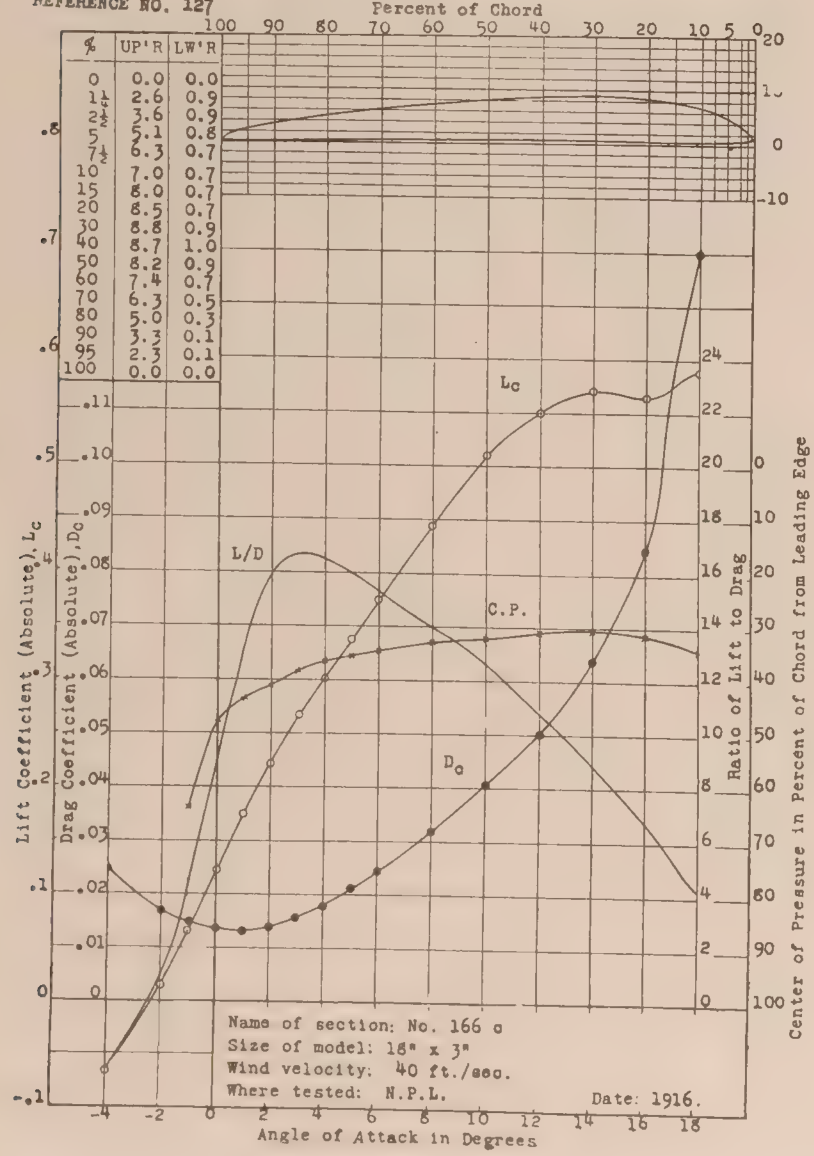
%	UP'R	LW'R
0	0.0	0.0
1¼	2.6	0.3
2½	3.6	0.3
5	5.1	0.4
7½	6.3	0.4
10	7.0	0.5
15	8.0	0.6
20	8.5	0.7
30	8.8	0.9
40	8.7	1.0
50	8.2	0.9
60	7.4	0.7
70	6.3	0.5
80	4.9	0.3
90	3.3	0.1
95	2.3	0.1
100	0.0	0.0

REFERENCE NO. 126



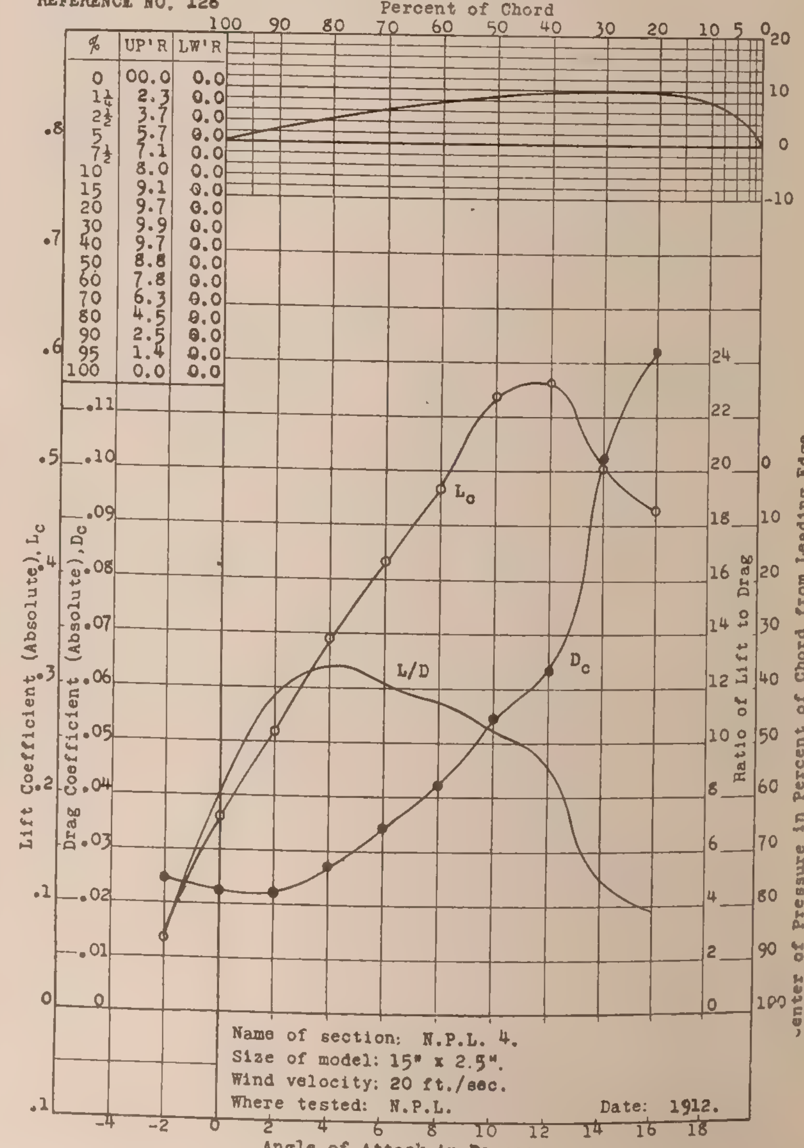
%	UP'R	LW'R
0	0.0	0.0
1¼	2.6	0.7
2½	3.6	0.6
5	5.1	0.6
7½	6.3	0.6
10	7.0	0.6
15	8.0	0.6
20	8.5	0.7
30	8.8	0.9
40	8.7	1.0
50	8.2	0.9
60	7.4	0.7
70	6.3	0.5
80	4.9	0.3
90	3.3	0.1
95	2.3	0.1
100	0.0	0.0

REFERENCE NO. 127



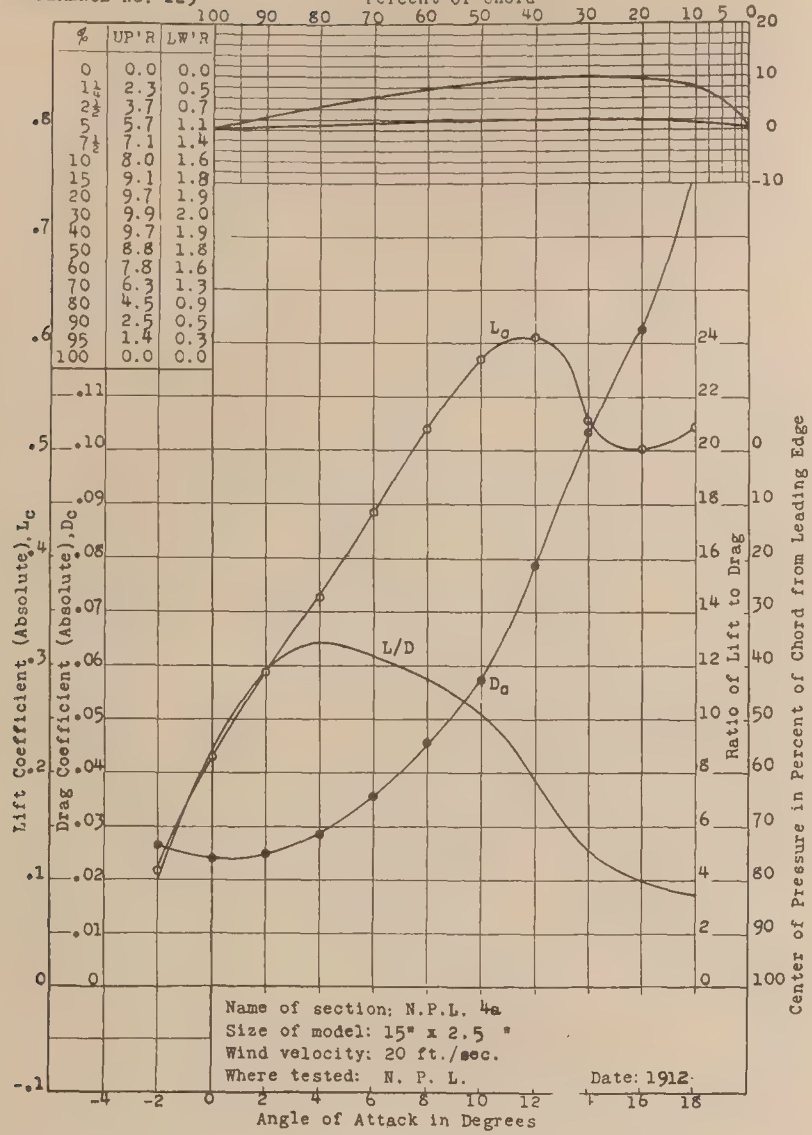
%	UP'R	LW'R
0	0.0	0.0
1¼	2.6	0.9
2½	3.6	0.9
5	5.1	0.8
7½	6.3	0.7
10	7.0	0.7
15	8.0	0.7
20	8.5	0.7
30	8.8	0.9
40	8.7	1.0
50	8.2	0.9
60	7.4	0.7
70	6.3	0.5
80	5.0	0.3
90	3.3	0.1
95	2.3	0.1
100	0.0	0.0

REFERENCE NO. 128



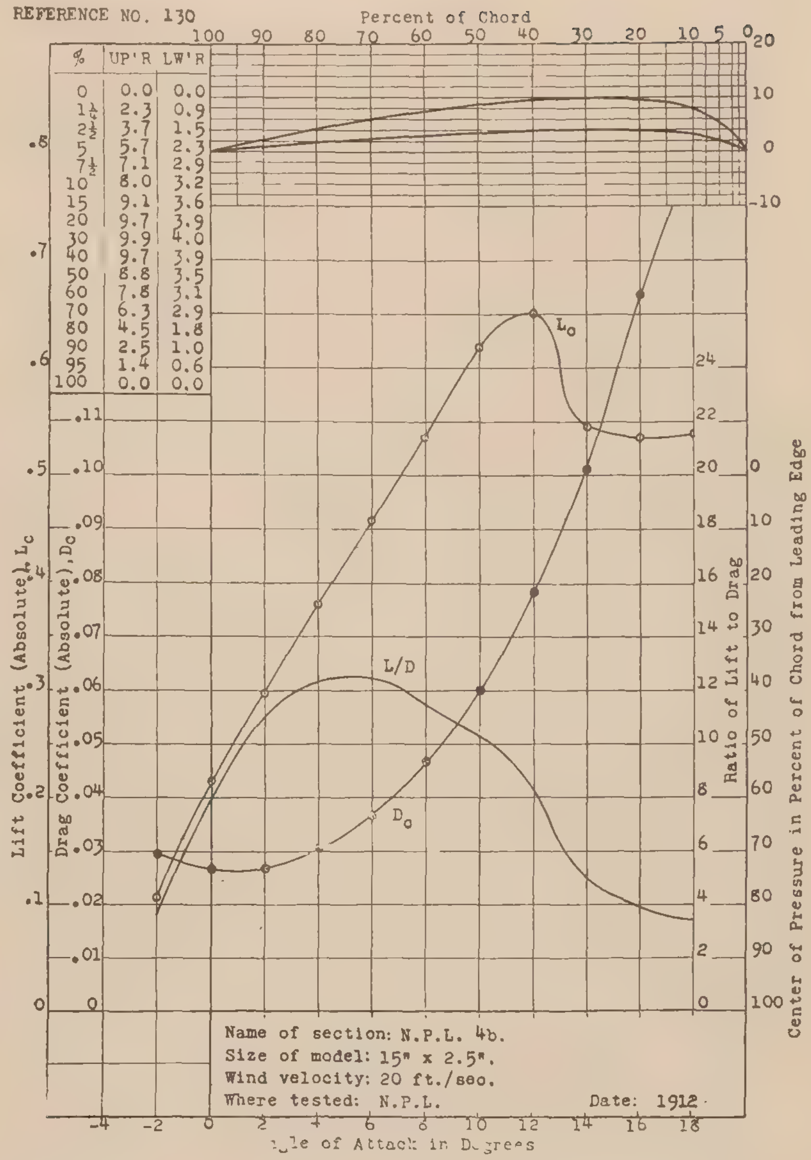
%	UP'R	LW'R
0	00.0	0.0
1¼	2.3	0.0
2½	3.7	0.0
5	5.7	0.0
7½	7.1	0.0
10	8.0	0.0
15	9.1	0.0
20	9.7	0.0
30	9.9	0.0
40	9.7	0.0
50	8.8	0.0
60	7.8	0.0
70	6.3	0.0
80	4.5	0.0
90	2.5	0.0
95	1.4	0.0
100	0.0	0.0

REFERENCE NO. 129



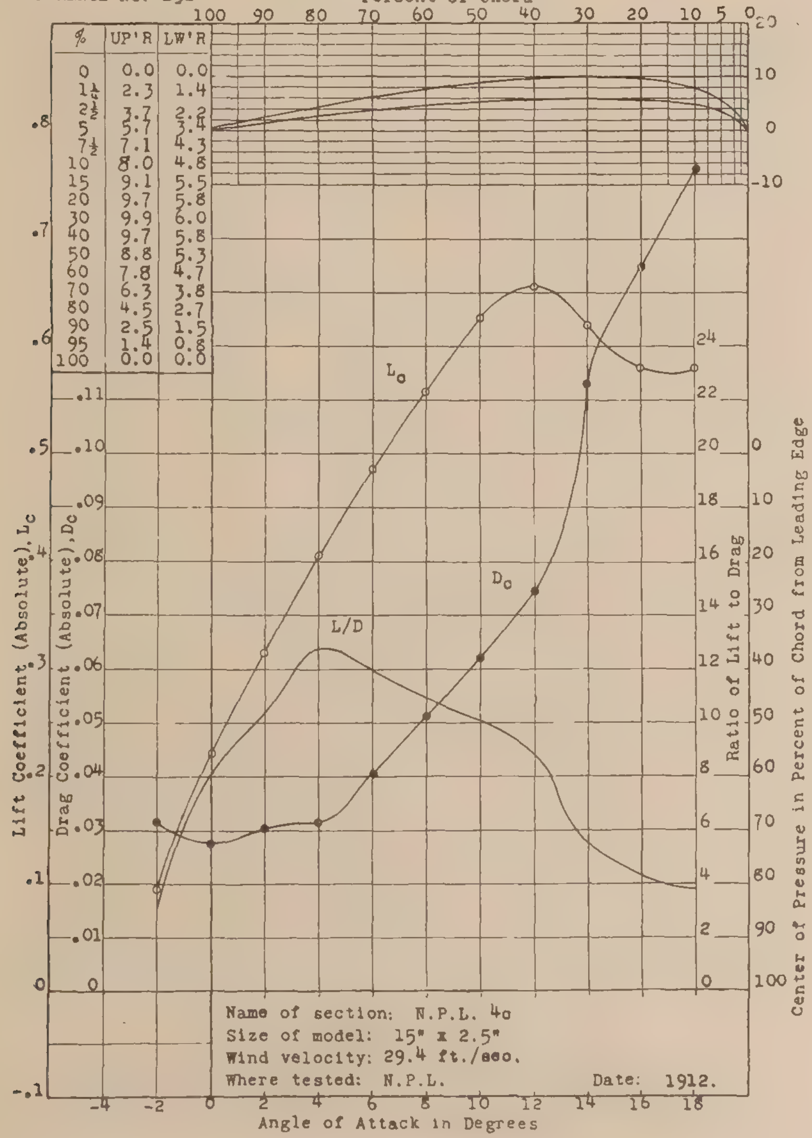
%	UP'R	LW'R
0	0.0	0.0
1¼	2.3	0.5
2½	3.7	0.7
5	5.7	1.1
7½	7.1	1.4
10	8.0	1.6
15	9.1	1.8
20	9.7	1.9
30	9.9	2.0
40	9.7	1.9
50	8.8	1.8
60	7.8	1.6
70	6.3	1.3
80	4.5	0.9
90	2.5	0.5
95	1.4	0.3
100	0.0	0.0

REFERENCE NO. 130



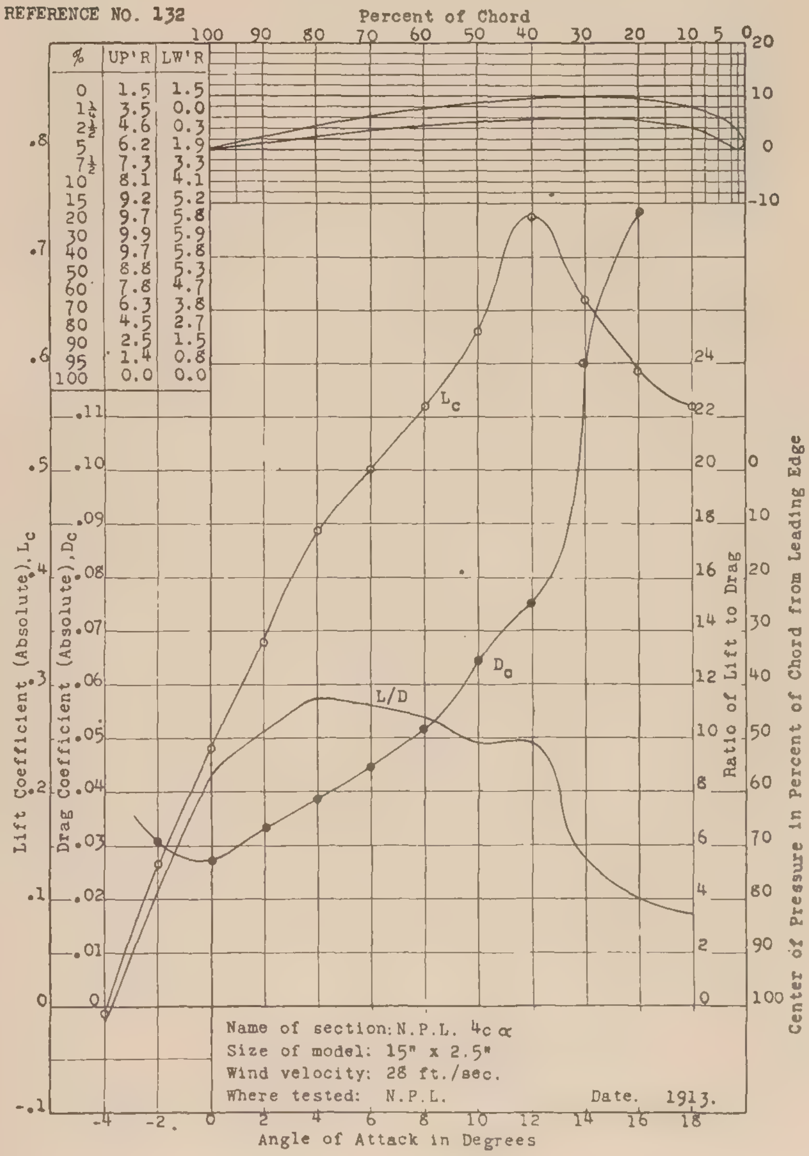
%	UP'R	LW'R
0	0.0	0.0
1¼	2.3	0.9
2½	3.7	1.5
5	5.7	2.3
7½	7.1	2.9
10	8.0	3.2
15	9.1	3.6
20	9.7	3.9
30	9.9	4.0
40	9.7	3.9
50	8.8	3.5
60	7.8	3.1
70	6.3	2.9
80	4.5	1.8
90	2.5	1.0
95	1.4	0.6
100	0.0	0.0

REFERENCE NO. 131



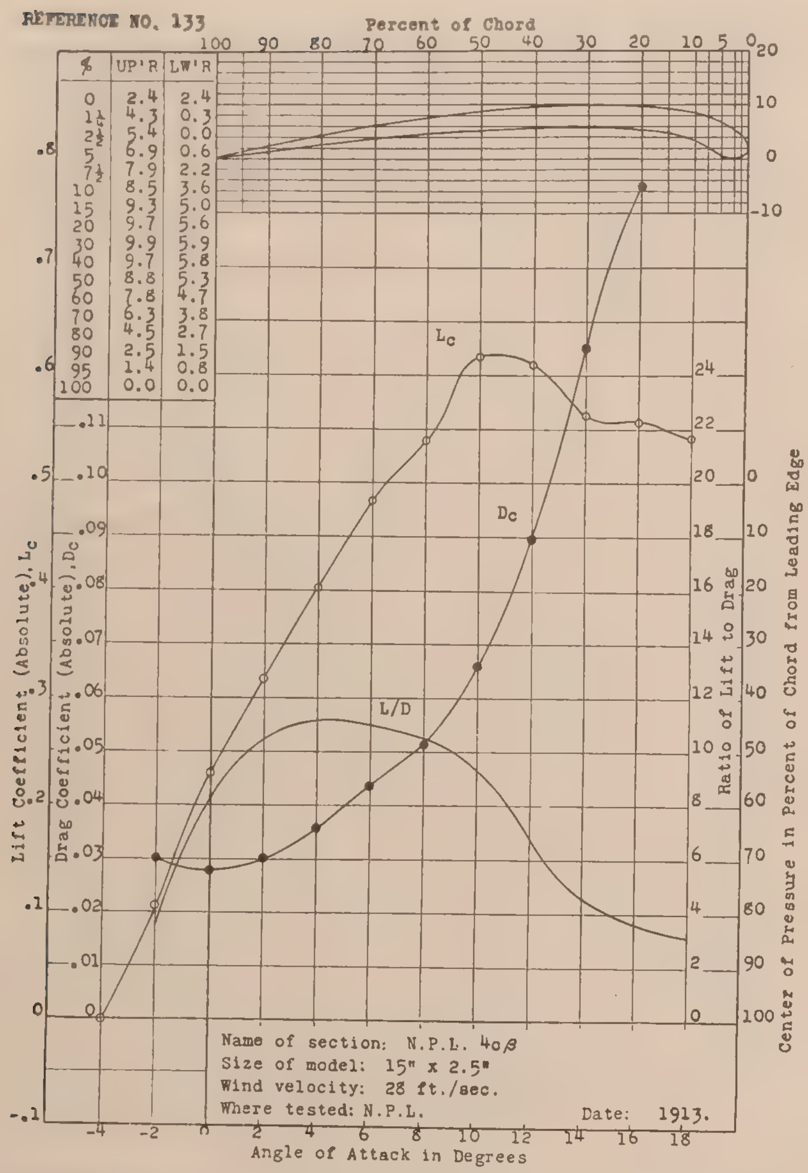
%	UP'R	LW'R
0	0.0	0.0
1¼	2.3	1.4
2½	3.7	2.2
5	5.7	3.4
7½	7.1	4.3
10	8.0	4.8
15	9.1	5.5
20	9.7	5.8
30	9.9	6.0
40	9.7	5.8
50	8.8	5.3
60	7.8	4.7
70	6.3	3.8
80	4.5	2.7
90	2.5	1.5
95	1.4	0.8
100	0.0	0.0

REFERENCE NO. 132



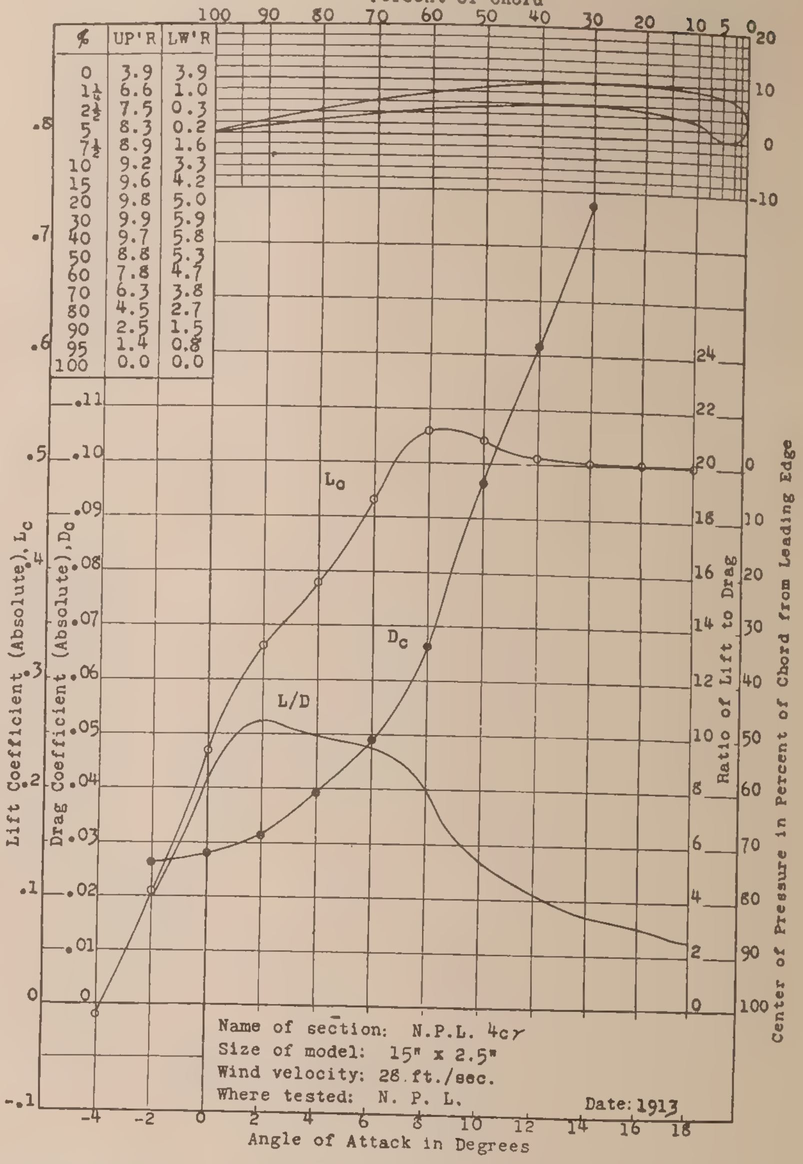
%	UP'R	LW'R
0	1.5	1.5
1¼	3.5	0.0
2½	4.6	0.3
5	6.2	1.9
7½	7.3	3.3
10	8.1	4.1
15	9.2	5.2
20	9.7	5.8
30	9.9	5.9
40	9.7	5.8
50	8.8	5.3
60	7.8	4.7
70	6.3	3.8
80	4.5	2.7
90	2.5	1.5
95	1.4	0.8
100	0.0	0.0

REFERENCE NO. 133



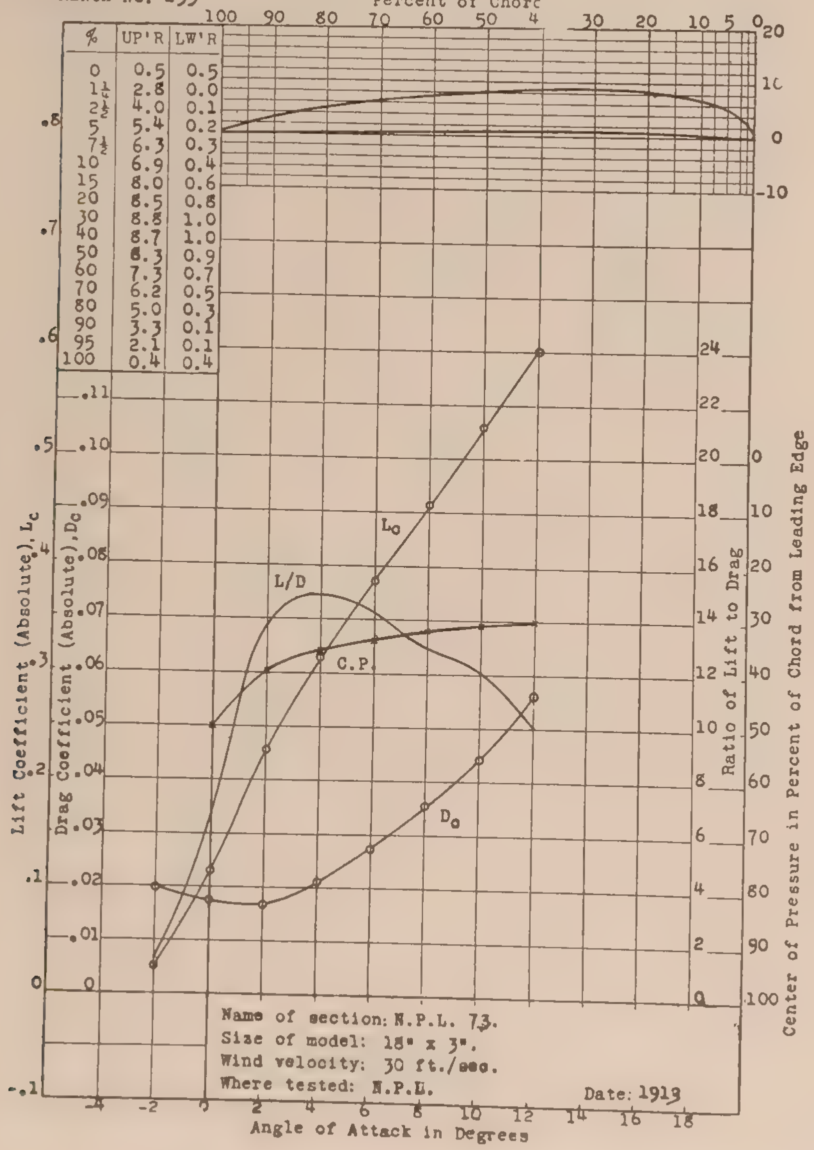
%	UP'R	LW'R
0	2.4	2.4
1¼	4.3	0.3
2½	5.4	0.0
5	6.9	0.6
7½	7.9	2.2
10	8.5	3.6
15	9.3	5.0
20	9.7	5.6
30	9.9	5.9
40	9.7	5.8
50	8.8	5.3
60	7.8	4.7
70	6.3	3.8
80	4.5	2.7
90	2.5	1.5
95	1.4	0.8
100	0.0	0.0

REFERENCE NO. 134



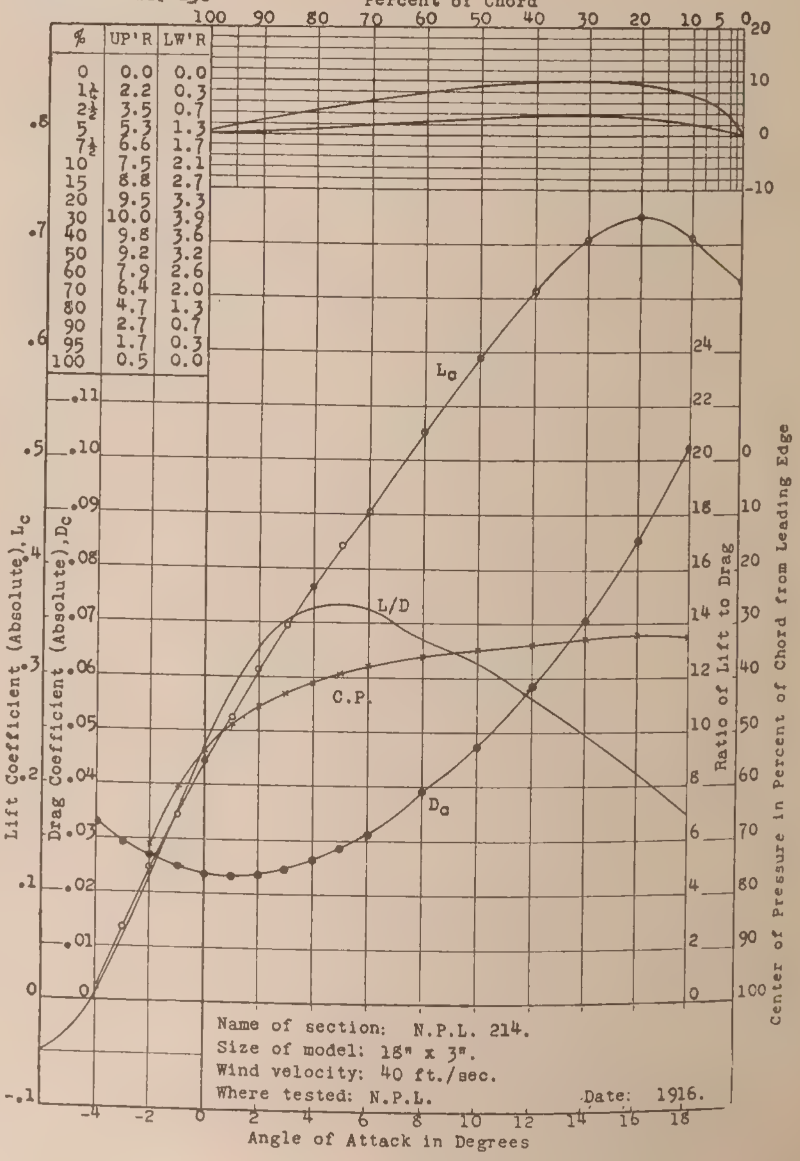
%	UP'R	LW'R
0	3.9	3.9
1¼	6.6	1.0
2½	7.5	0.3
5	8.3	0.2
7½	8.9	1.6
10	9.2	3.3
15	9.6	4.2
20	9.8	5.0
30	9.9	5.9
40	9.7	5.8
50	8.8	5.3
60	7.8	4.7
70	6.3	3.8
80	4.5	2.7
90	2.5	1.5
95	1.4	0.8
100	0.0	0.0

REFERENCE NO. 135



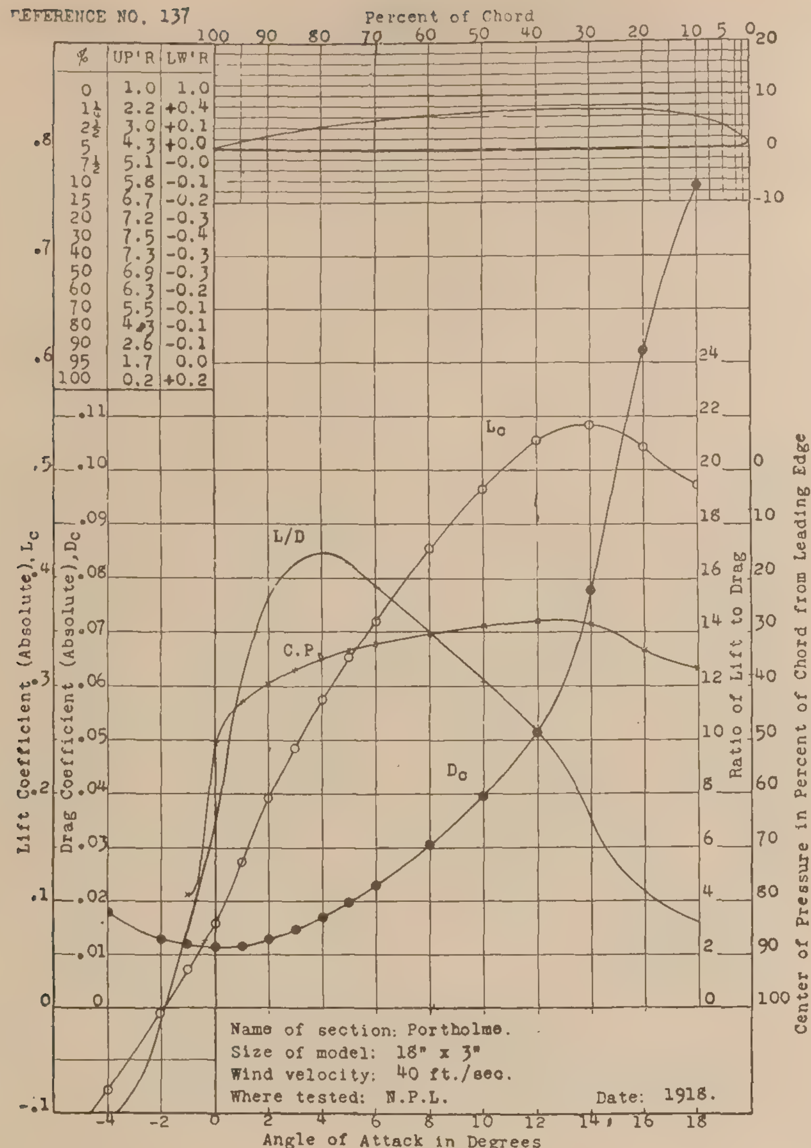
%	UP'R	LW'R
0	0.5	0.5
1¼	2.8	0.0
2½	4.0	0.1
5	5.4	0.2
7½	6.3	0.3
10	6.9	0.4
15	8.0	0.6
20	8.5	0.8
30	8.8	1.0
40	8.7	1.0
50	8.3	0.9
60	7.3	0.7
70	6.2	0.5
80	5.0	0.3
90	3.3	0.1
95	2.1	0.1
100	0.4	0.4

REFERENCE NO. 136



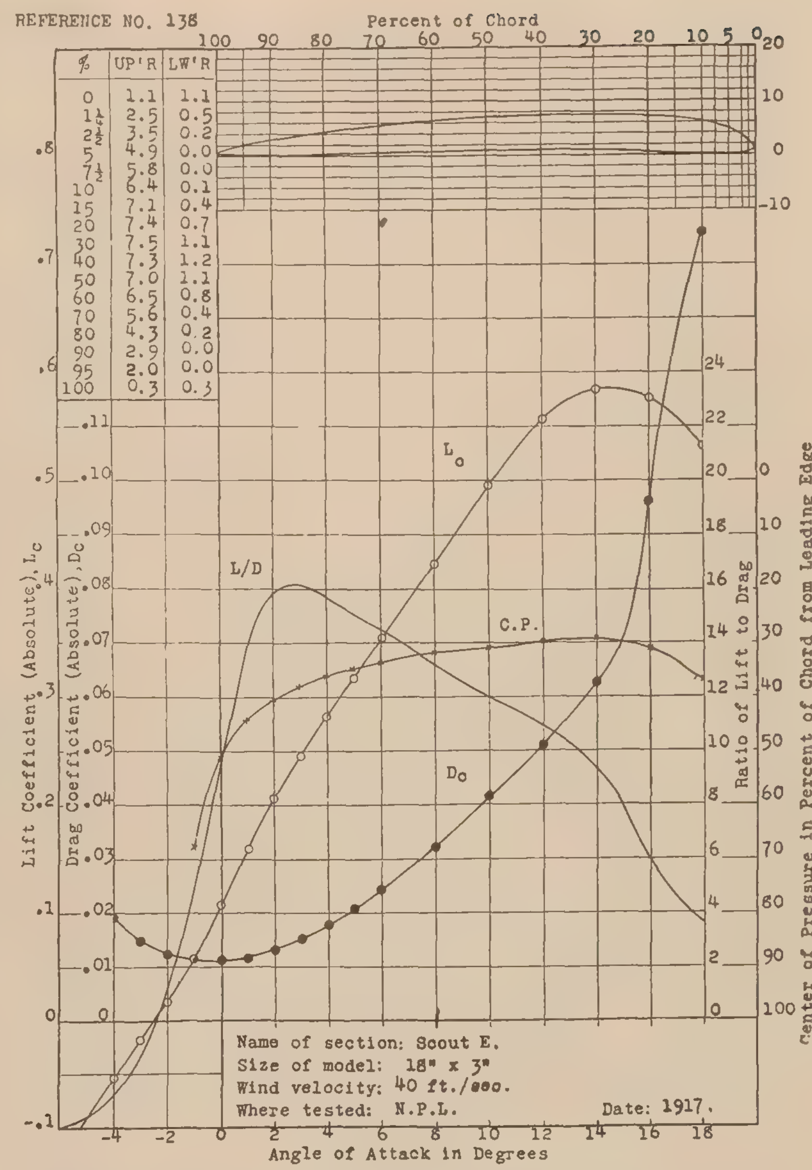
%	UP'R	LW'R
0	0.0	0.0
1¼	2.2	0.3
2½	3.5	0.7
5	5.3	1.3
7½	6.6	1.7
10	7.5	2.1
15	8.8	2.7
20	9.5	3.3
30	10.0	3.9
40	9.8	3.6
50	9.2	3.2
60	7.9	2.6
70	6.4	2.0
80	4.7	1.3
90	2.7	0.7
95	1.7	0.3
100	0.5	0.0

REFERENCE NO. 137



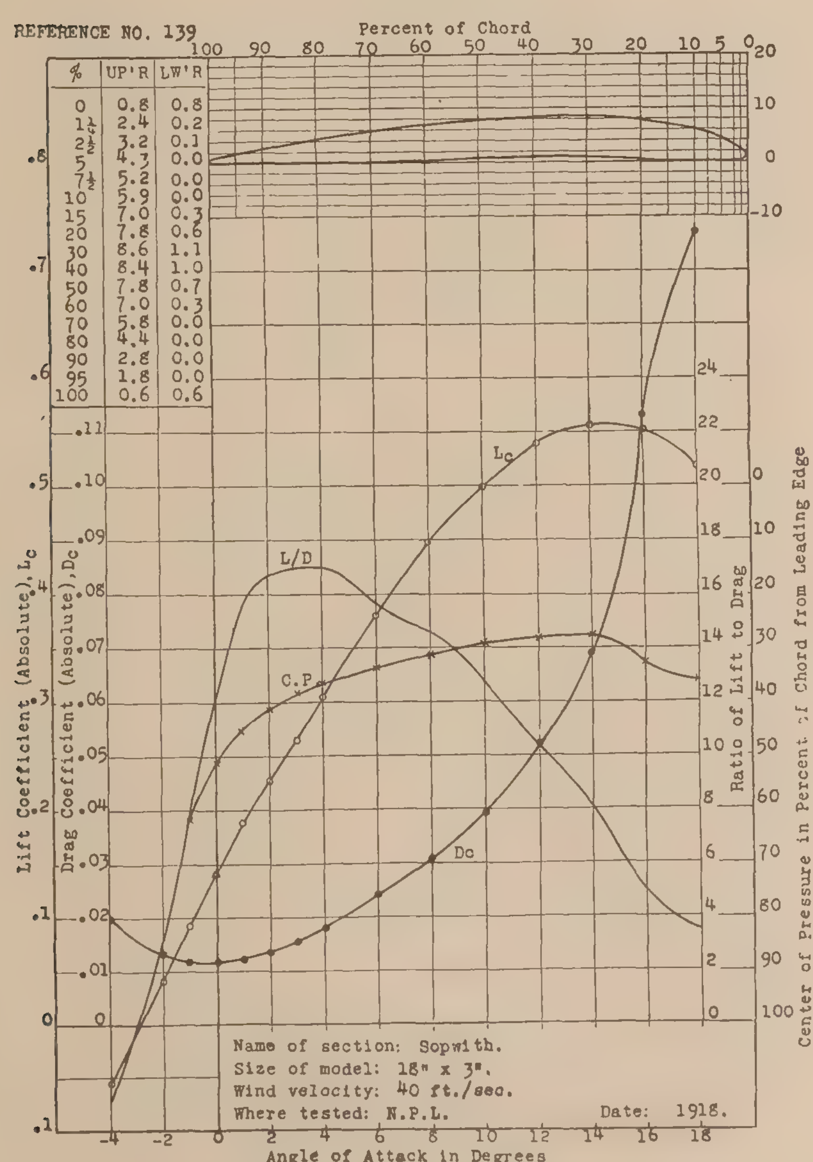
%	UP'R	LW'R
0	1.0	1.0
1¼	2.2	+0.4
2½	3.0	+0.1
5	4.3	+0.0
7½	5.1	-0.0
10	5.8	-0.1
15	6.7	-0.2
20	7.2	-0.3
30	7.5	-0.4
40	7.3	-0.3
50	6.9	-0.3
60	6.3	-0.2
70	5.5	-0.1
80	4.3	-0.1
90	2.6	-0.1
95	1.7	0.0
100	0.2	+0.2

REFERENCE NO. 138



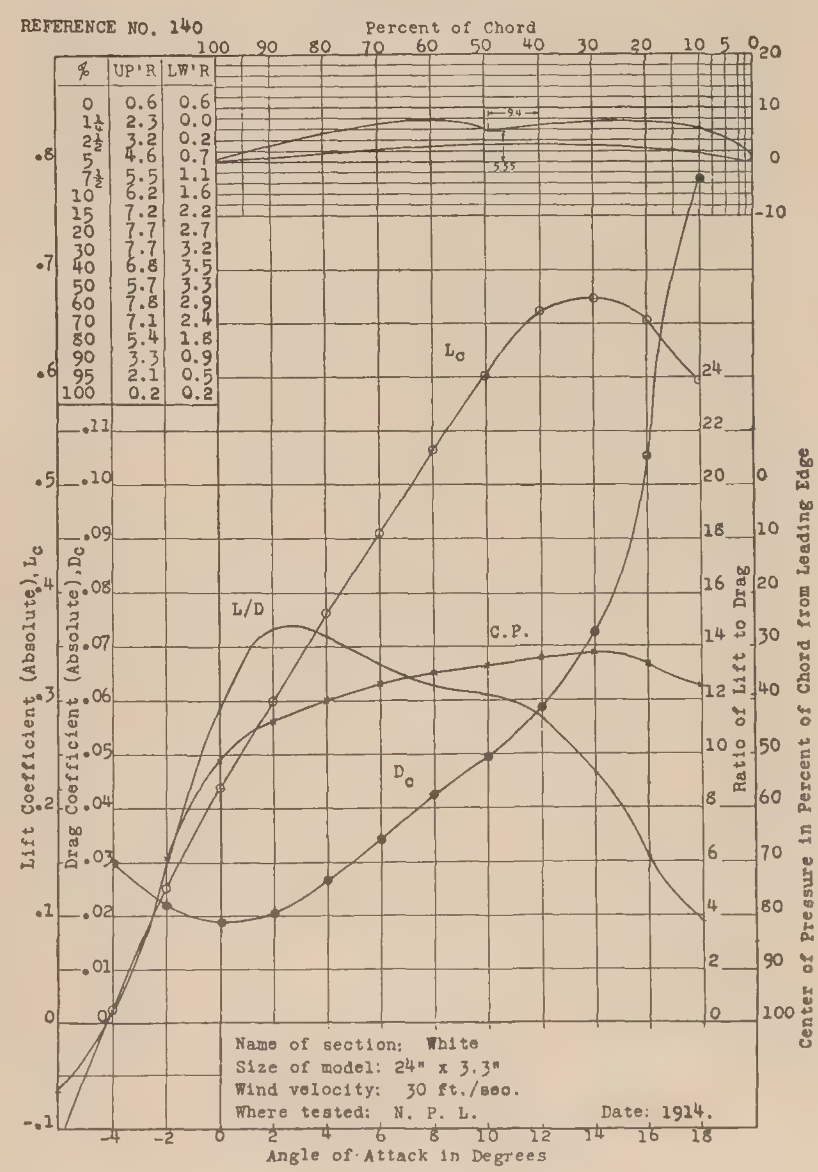
%	UP'R	LW'R
0	1.1	1.1
1¼	2.5	0.5
2½	3.5	0.2
5	4.9	0.0
7½	5.8	0.0
10	6.4	0.1
15	7.1	0.4
20	7.4	0.7
30	7.5	1.1
40	7.3	1.2
50	7.0	1.1
60	6.5	0.8
70	5.6	0.4
80	4.3	0.2
90	2.9	0.0
95	2.0	0.0
100	0.3	0.3

REFERENCE NO. 139



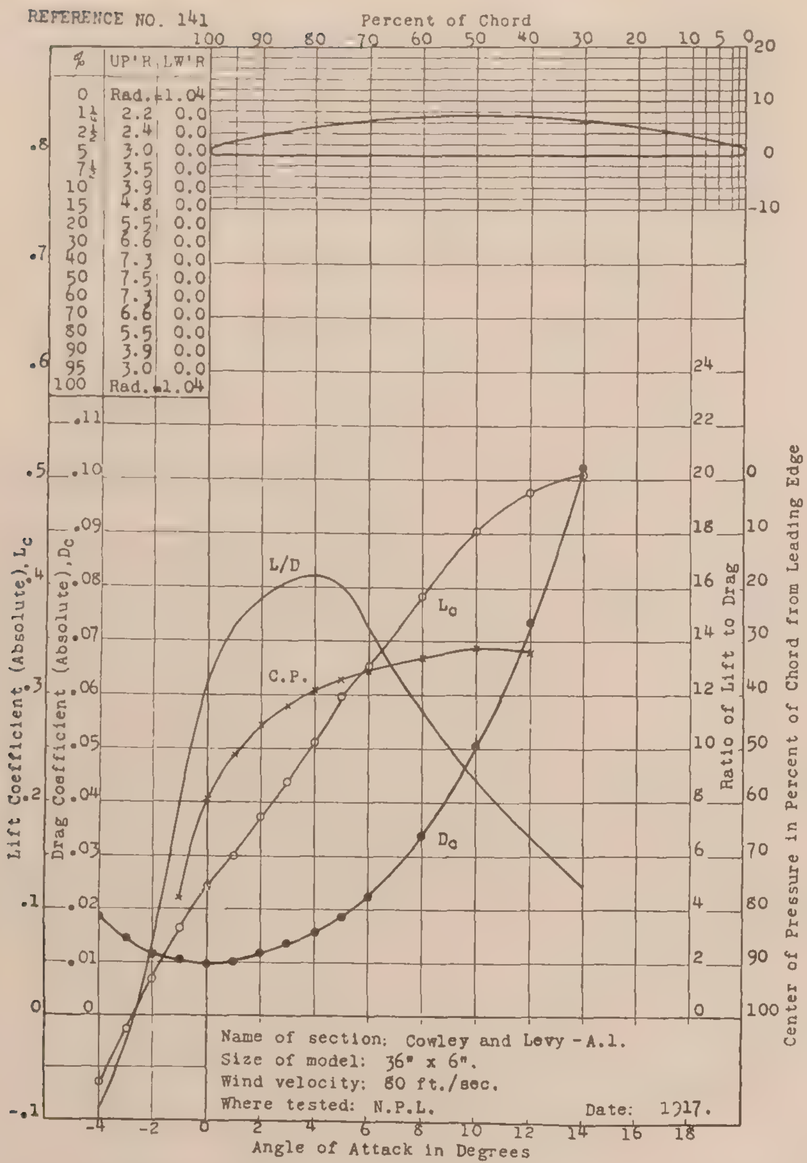
%	UP'R	LW'R
0	0.8	0.8
1¼	2.4	0.2
2½	3.2	0.1
5	4.3	0.0
7½	5.2	0.0
10	5.9	0.0
15	7.0	0.3
20	7.8	0.6
30	8.6	1.1
40	8.4	1.0
50	7.8	0.7
60	7.0	0.3
70	5.8	0.0
80	4.4	0.0
90	2.8	0.0
95	1.8	0.0
100	0.6	0.6

REFERENCE NO. 140



%	UP'R	LW'R
0	0.6	0.6
1¼	2.3	0.0
2½	3.2	0.2
5	4.6	0.7
7½	5.5	1.1
10	6.2	1.6
15	7.2	2.2
20	7.7	2.7
30	7.7	3.2
40	6.8	3.5
50	5.7	3.3
60	7.8	2.9
70	7.1	2.4
80	5.4	1.8
90	3.3	0.9
95	2.1	0.5
100	0.2	0.2

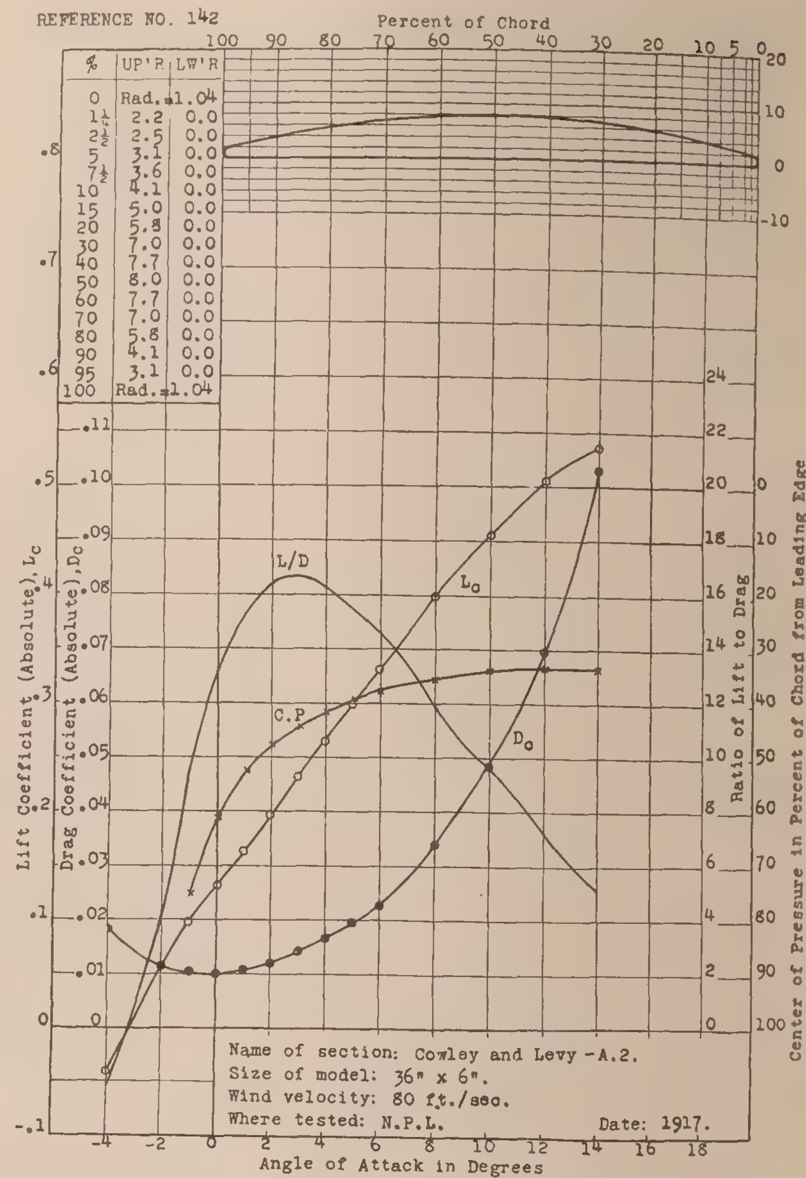
REFERENCE NO. 141



%	UP'R	LW'R
0	Rad.=1.04	
1¼	2.2	0.0
2½	2.4	0.0
5	3.0	0.0
7½	3.5	0.0
10	3.9	0.0
15	4.8	0.0
20	5.5	0.0
30	6.6	0.0
40	7.3	0.0
50	7.5	0.0
60	7.3	0.0
70	6.6	0.0
80	5.5	0.0
90	3.9	0.0
95	3.0	0.0
100	Rad.=1.04	

Name of section: Cowley and Levy - A.1.
Size of model: 36" x 6".
Wind velocity: 80 ft./sec.
Where tested: N.P.L. Date: 1917.

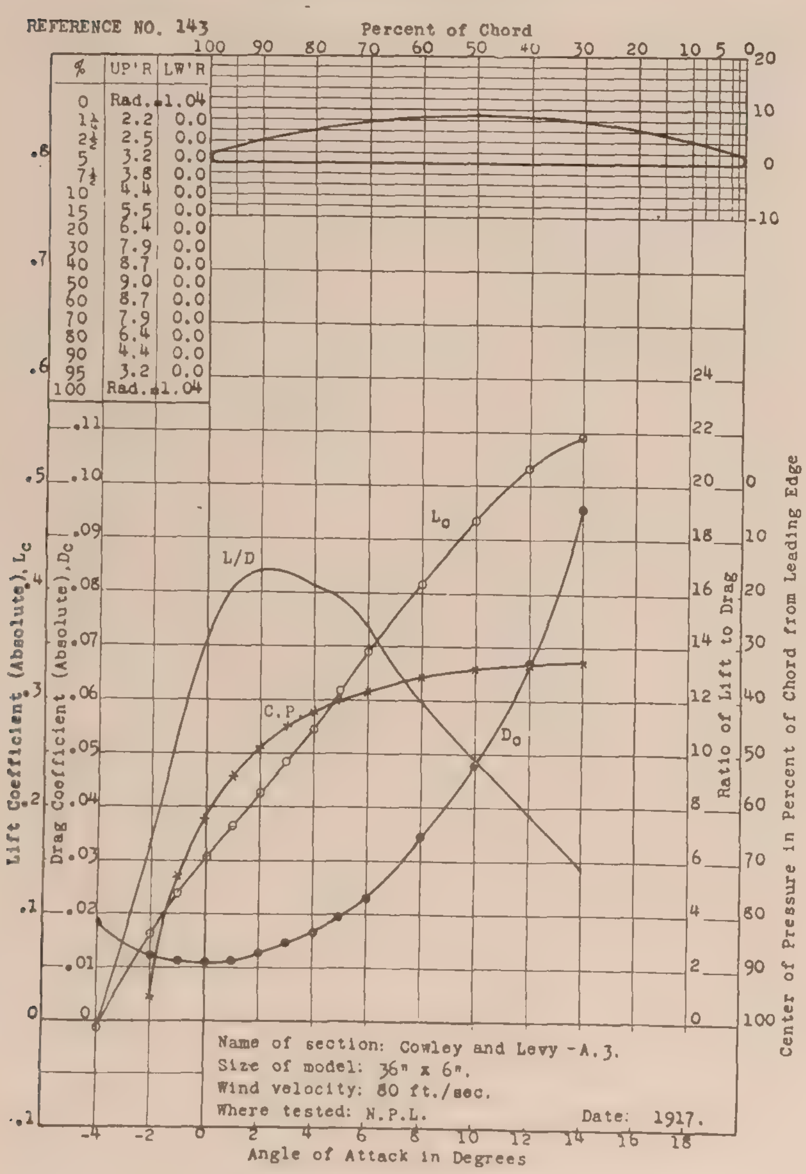
REFERENCE NO. 142



%	UP'R	LW'R
0	Rad.=1.04	
1¼	2.2	0.0
2½	2.5	0.0
5	3.1	0.0
7½	3.6	0.0
10	4.1	0.0
15	5.0	0.0
20	5.8	0.0
30	7.0	0.0
40	7.7	0.0
50	8.0	0.0
60	7.7	0.0
70	7.0	0.0
80	5.8	0.0
90	4.1	0.0
95	3.1	0.0
100	Rad.=1.04	

Name of section: Cowley and Levy - A.2.
Size of model: 36" x 6".
Wind velocity: 80 ft./sec.
Where tested: N.P.L. Date: 1917.

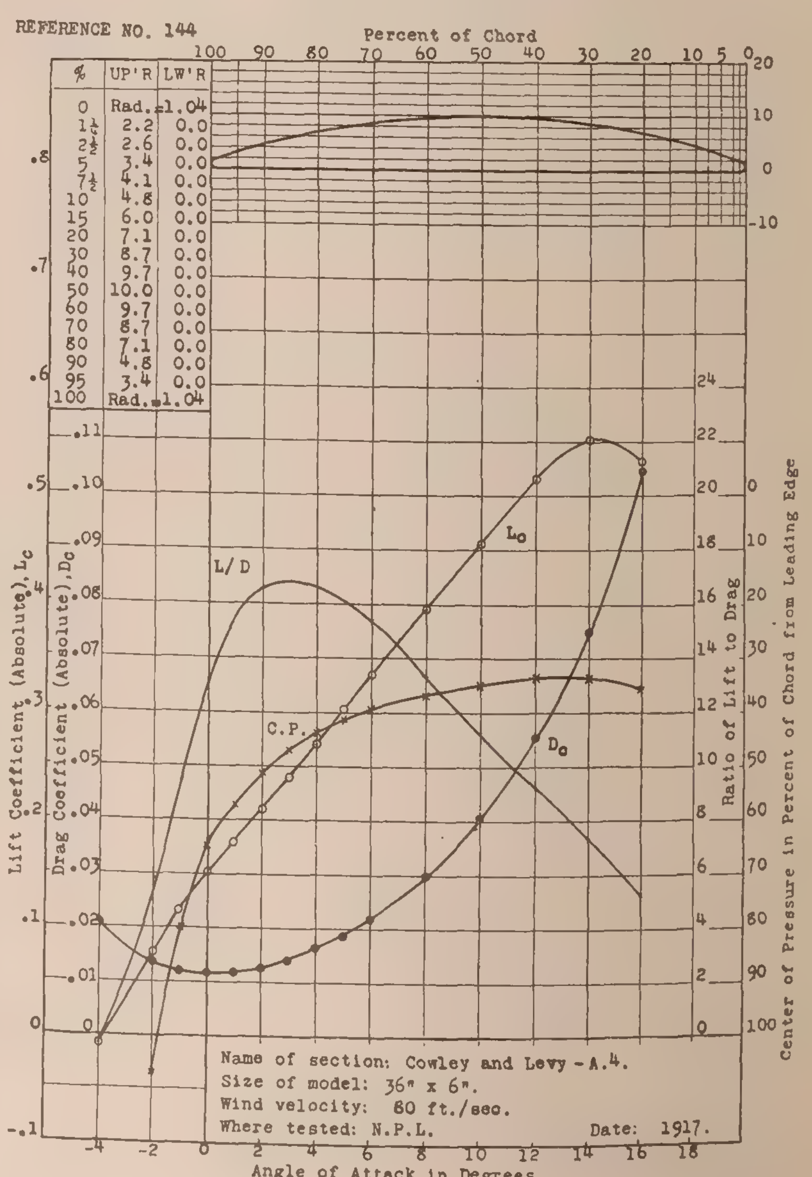
REFERENCE NO. 143



%	UP'R	LW'R
0	Rad.=1.04	
1¼	2.2	0.0
2½	2.5	0.0
5	3.2	0.0
7½	3.8	0.0
10	4.4	0.0
15	5.5	0.0
20	6.4	0.0
30	7.9	0.0
40	8.7	0.0
50	9.0	0.0
60	8.7	0.0
70	7.9	0.0
80	6.4	0.0
90	4.4	0.0
95	3.2	0.0
100	Rad.=1.04	

Name of section: Cowley and Levy - A.3.
Size of model: 36" x 6".
Wind velocity: 80 ft./sec.
Where tested: N.P.L. Date: 1917.

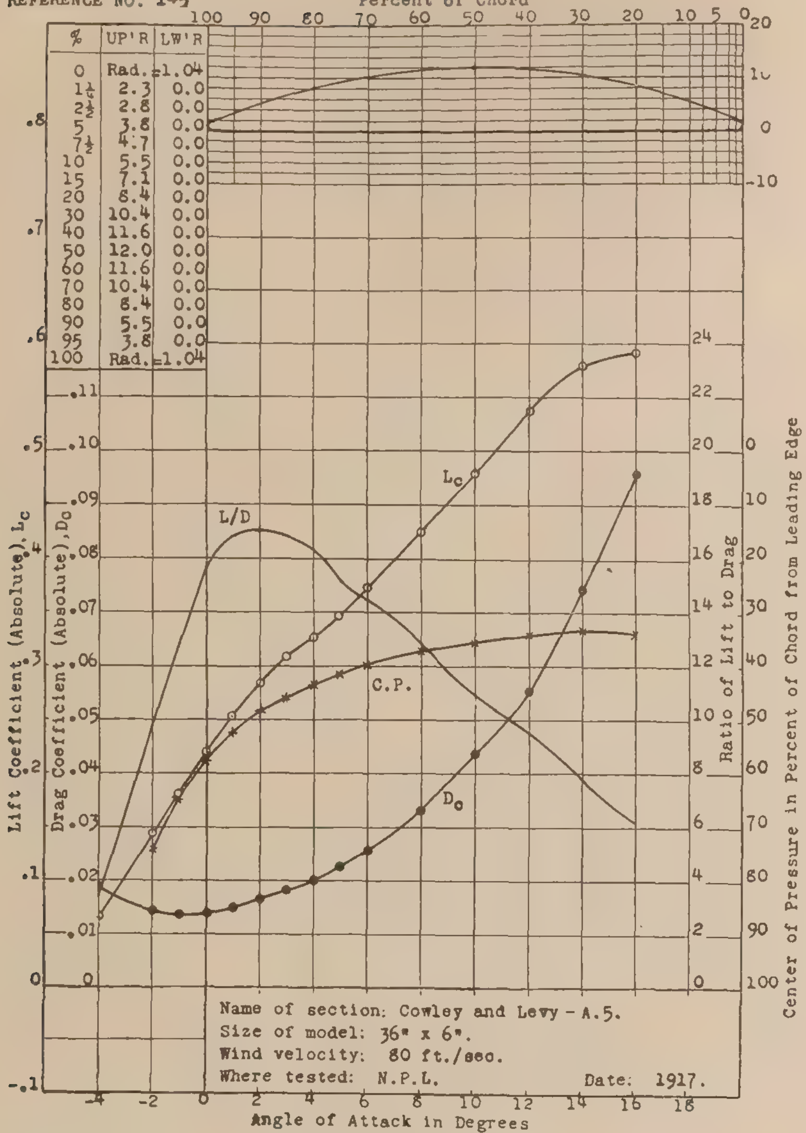
REFERENCE NO. 144



%	UP'R	LW'R
0	Rad.=1.04	
1¼	2.2	0.0
2½	2.6	0.0
5	3.4	0.0
7½	4.1	0.0
10	4.8	0.0
15	6.0	0.0
20	7.1	0.0
30	8.7	0.0
40	9.7	0.0
50	10.0	0.0
60	9.7	0.0
70	8.7	0.0
80	7.1	0.0
90	4.8	0.0
95	3.4	0.0
100	Rad.=1.04	

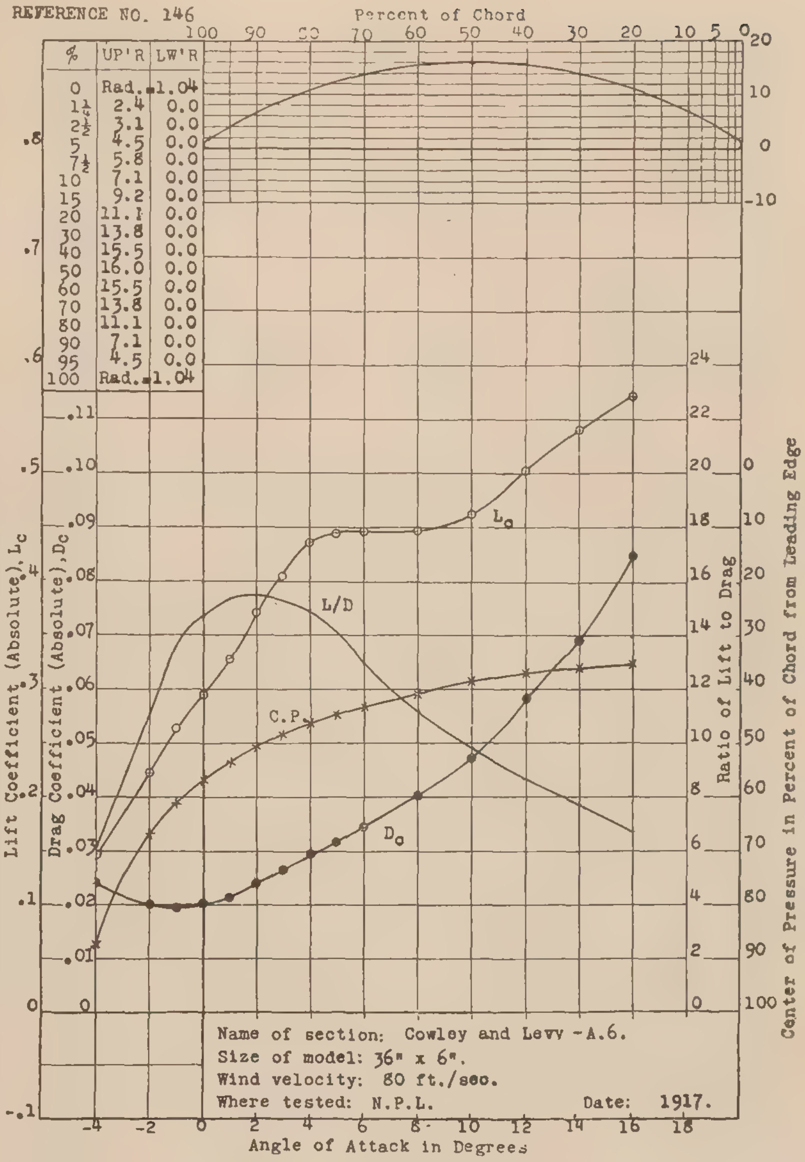
Name of section: Cowley and Levy - A.4.
Size of model: 36" x 6".
Wind velocity: 80 ft./sec.
Where tested: N.P.L. Date: 1917.

REFERENCE NO. 145



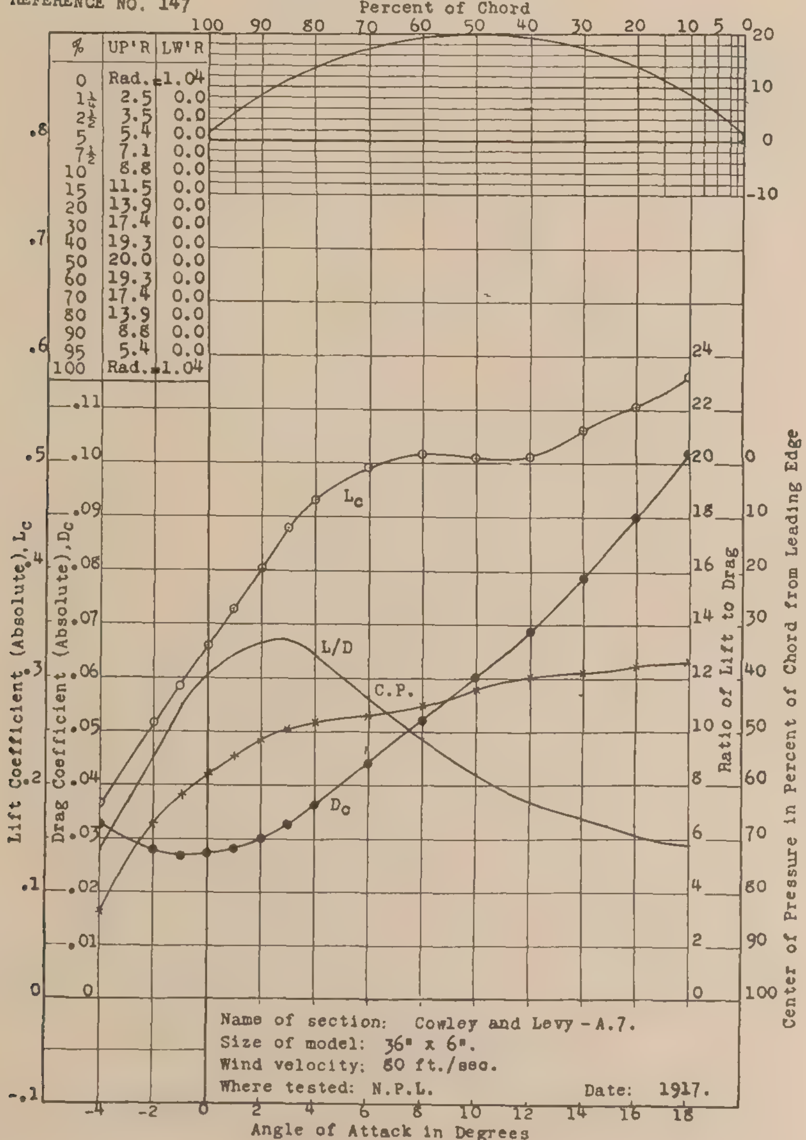
%	UP'R	LW'R
0	Rad. = 1.04	
1¼	2.3	0.0
2½	2.8	0.0
5	3.8	0.0
7½	4.7	0.0
10	5.5	0.0
15	7.1	0.0
20	8.4	0.0
30	10.4	0.0
40	11.6	0.0
50	12.0	0.0
60	11.6	0.0
70	10.4	0.0
80	8.4	0.0
90	5.5	0.0
95	3.8	0.0
100	Rad. = 1.04	

REFERENCE NO. 146



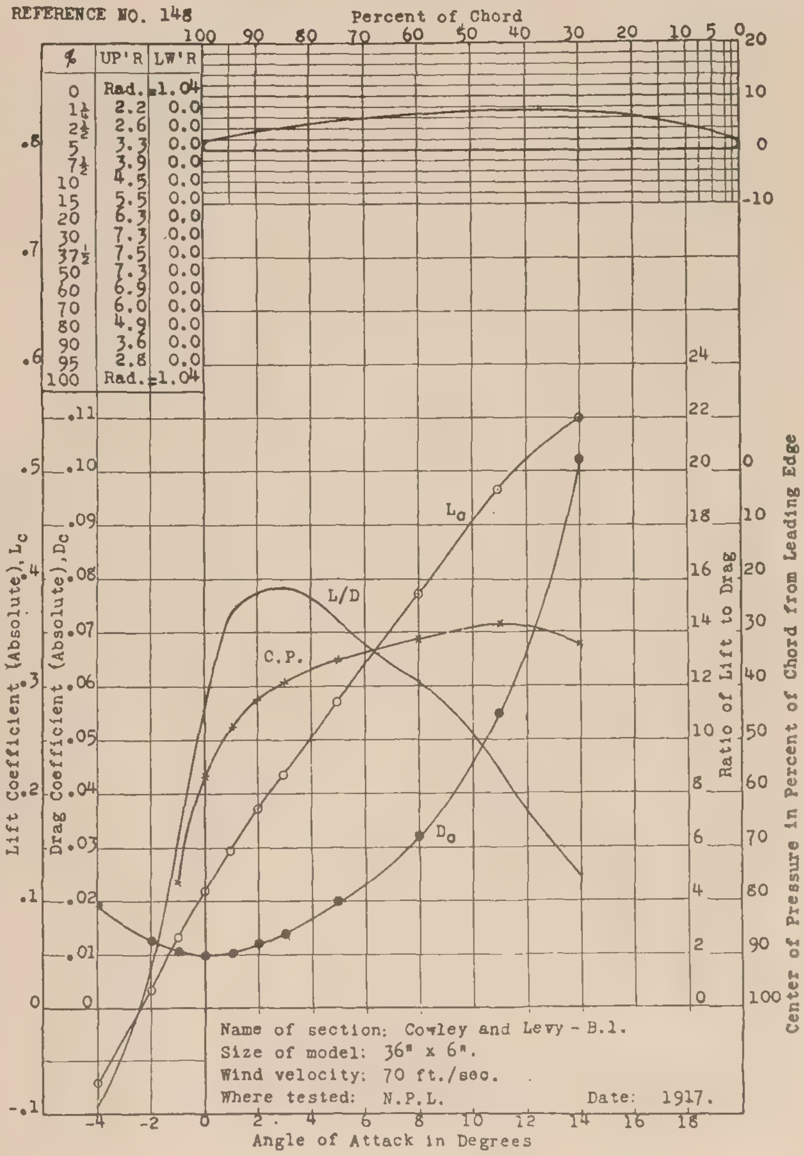
%	UP'R	LW'R
0	Rad. = 1.04	
1¼	2.4	0.0
2½	3.1	0.0
5	4.5	0.0
7½	5.8	0.0
10	7.1	0.0
15	9.2	0.0
20	11.1	0.0
30	13.8	0.0
40	15.5	0.0
50	16.0	0.0
60	15.5	0.0
70	13.8	0.0
80	11.1	0.0
90	7.1	0.0
95	4.5	0.0
100	Rad. = 1.04	

REFERENCE NO. 147



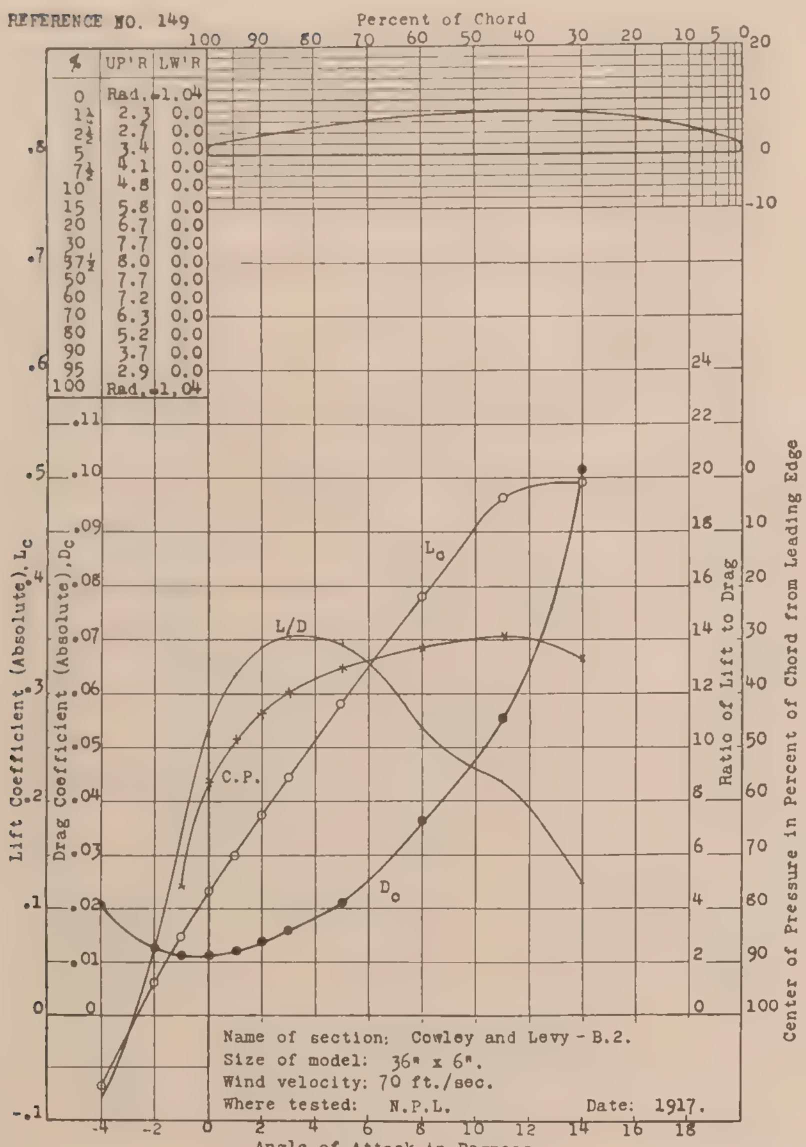
%	UP'R	LW'R
0	Rad. = 1.04	
1¼	2.5	0.0
2½	3.5	0.0
5	5.4	0.0
7½	7.1	0.0
10	8.8	0.0
15	11.5	0.0
20	13.9	0.0
30	17.4	0.0
40	19.3	0.0
50	20.0	0.0
60	19.3	0.0
70	17.4	0.0
80	13.9	0.0
90	8.8	0.0
95	5.4	0.0
100	Rad. = 1.04	

REFERENCE NO. 148



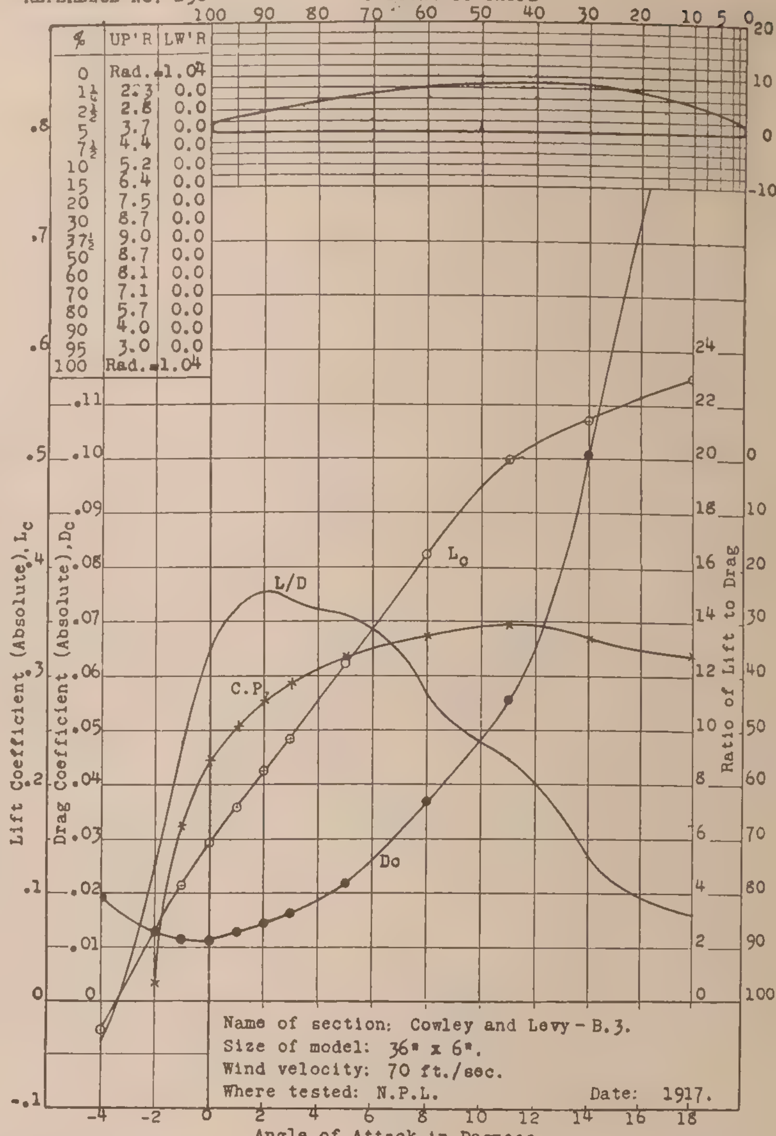
%	UP'R	LW'R
0	Rad. = 1.04	
1¼	2.2	0.0
2½	2.6	0.0
5	3.3	0.0
7½	3.9	0.0
10	4.5	0.0
15	5.5	0.0
20	6.3	0.0
30	7.3	0.0
37½	7.5	0.0
50	7.3	0.0
60	6.9	0.0
70	6.0	0.0
80	4.9	0.0
90	3.6	0.0
95	2.8	0.0
100	Rad. = 1.04	

REFERENCE NO. 149



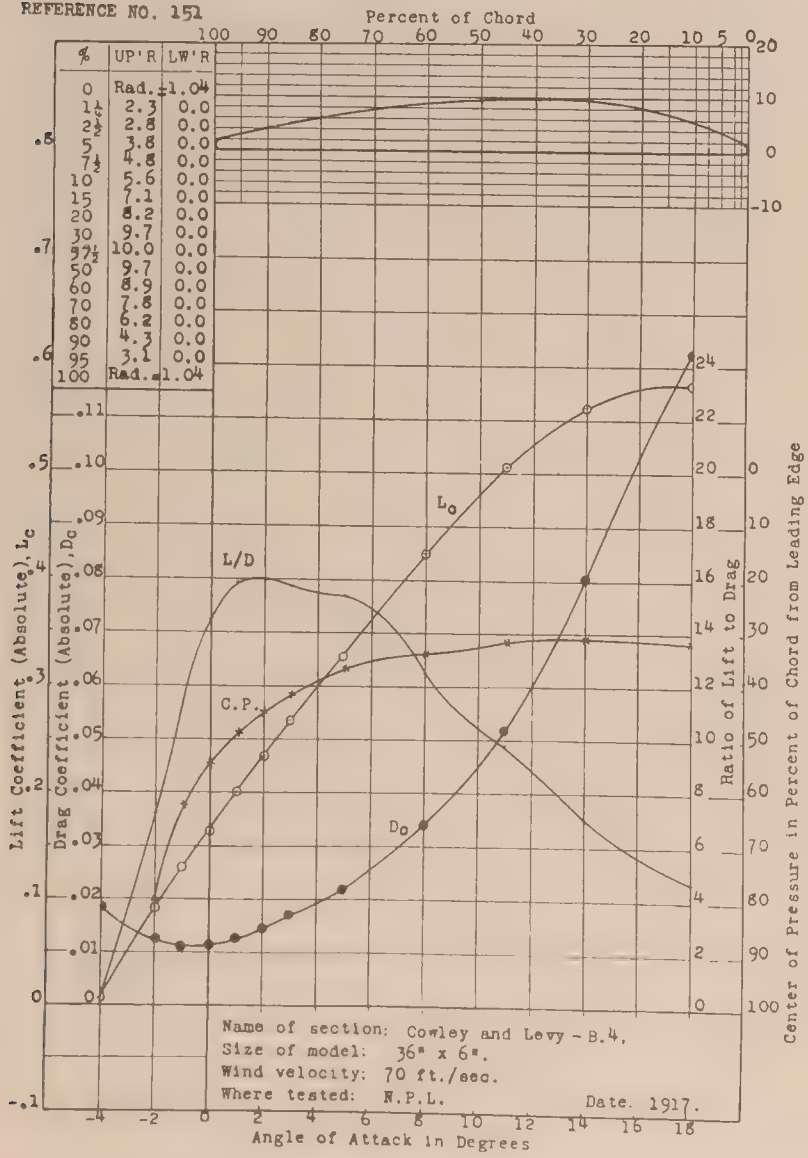
%	UP'R	LW'R
0	Rad.=1.04	
1¼	2.3	0.0
2½	2.7	0.0
5	3.4	0.0
7½	4.1	0.0
10	4.8	0.0
15	5.8	0.0
20	6.7	0.0
30	7.7	0.0
37½	8.0	0.0
50	7.7	0.0
60	7.2	0.0
70	6.3	0.0
80	5.2	0.0
90	3.7	0.0
95	2.9	0.0
100	Rad.=1.04	

REFERENCE NO. 150



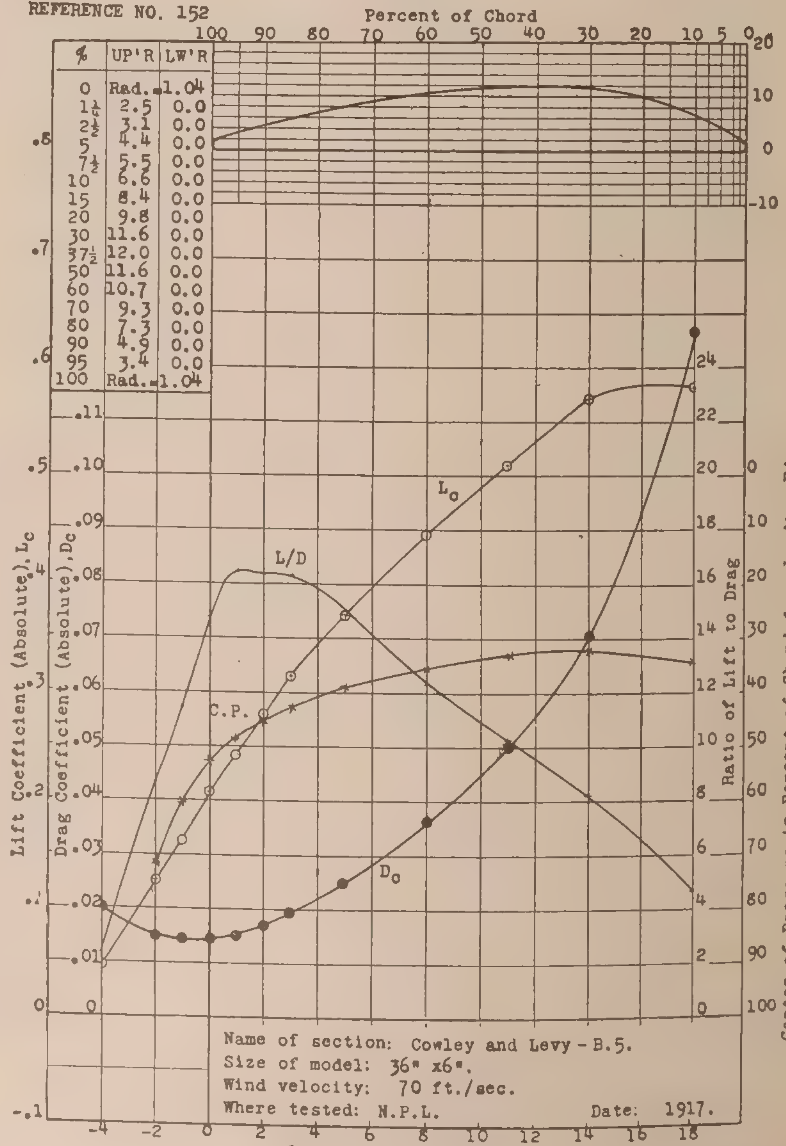
%	UP'R	LW'R
0	Rad.=1.04	
1¼	2.3	0.0
2½	2.8	0.0
5	3.7	0.0
7½	4.4	0.0
10	5.2	0.0
15	6.4	0.0
20	7.5	0.0
30	8.7	0.0
37½	9.0	0.0
50	8.7	0.0
60	8.1	0.0
70	7.1	0.0
80	5.7	0.0
90	4.0	0.0
95	3.0	0.0
100	Rad.=1.04	

REFERENCE NO. 151



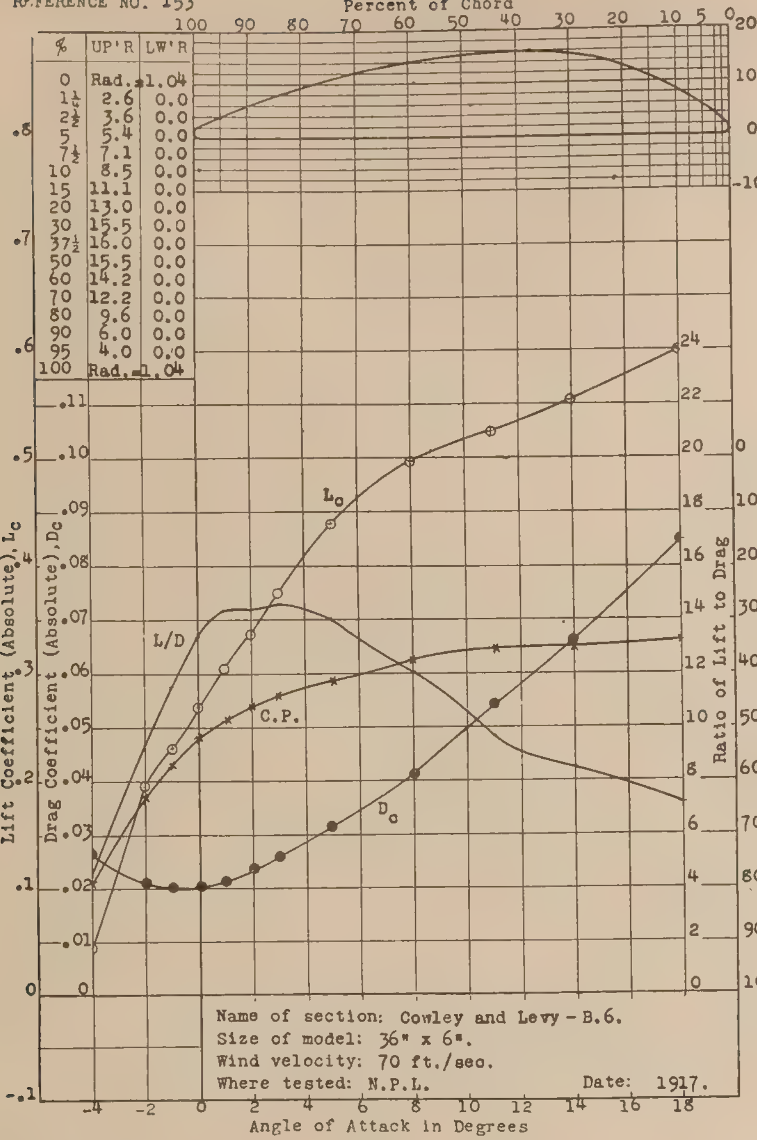
%	UP'R	LW'R
0	Rad.=1.04	
1¼	2.3	0.0
2½	2.8	0.0
5	3.8	0.0
7½	4.8	0.0
10	5.6	0.0
15	7.1	0.0
20	8.2	0.0
30	9.7	0.0
37½	10.0	0.0
50	9.7	0.0
60	8.9	0.0
70	7.8	0.0
80	6.2	0.0
90	4.3	0.0
95	3.1	0.0
100	Rad.=1.04	

REFERENCE NO. 152



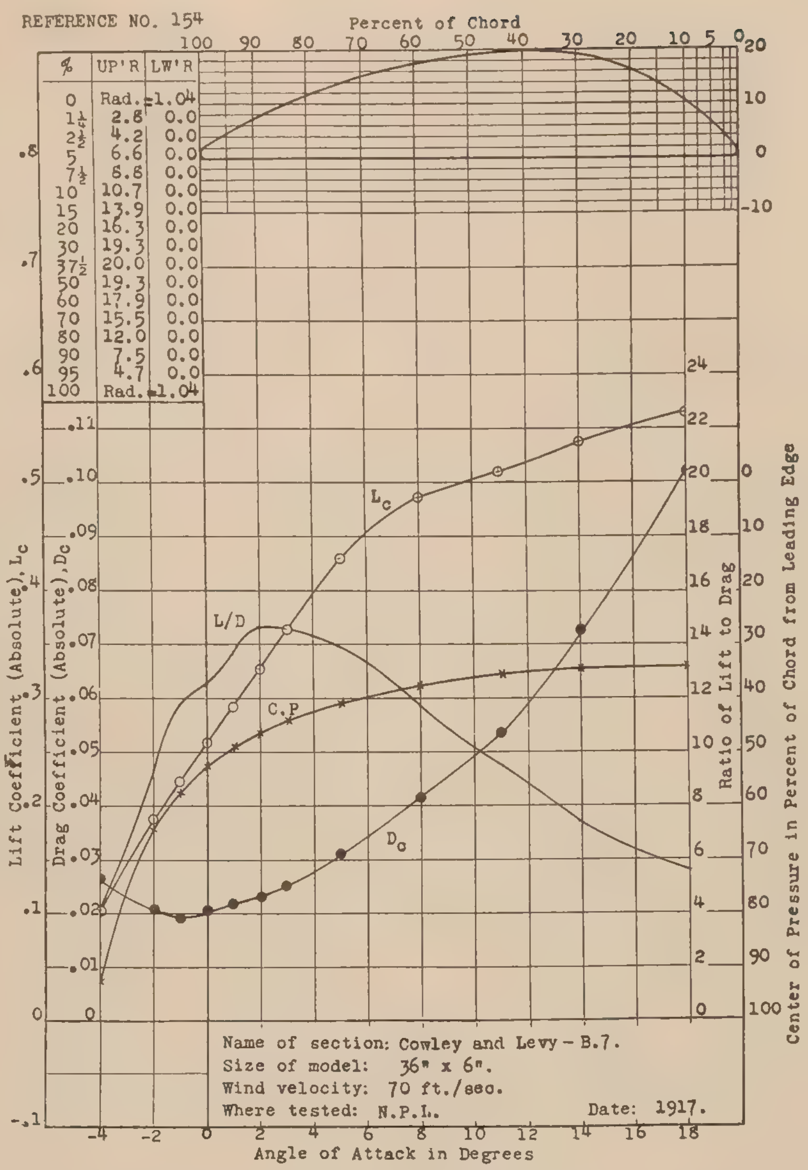
%	UP'R	LW'R
0	Rad.=1.04	
1¼	2.5	0.0
2½	3.1	0.0
5	4.4	0.0
7½	5.5	0.0
10	6.6	0.0
15	8.4	0.0
20	9.8	0.0
30	11.6	0.0
37½	12.0	0.0
50	11.6	0.0
60	10.7	0.0
70	9.3	0.0
80	7.3	0.0
90	4.5	0.0
95	3.4	0.0
100	Rad.=1.04	

REFERENCE NO. 153



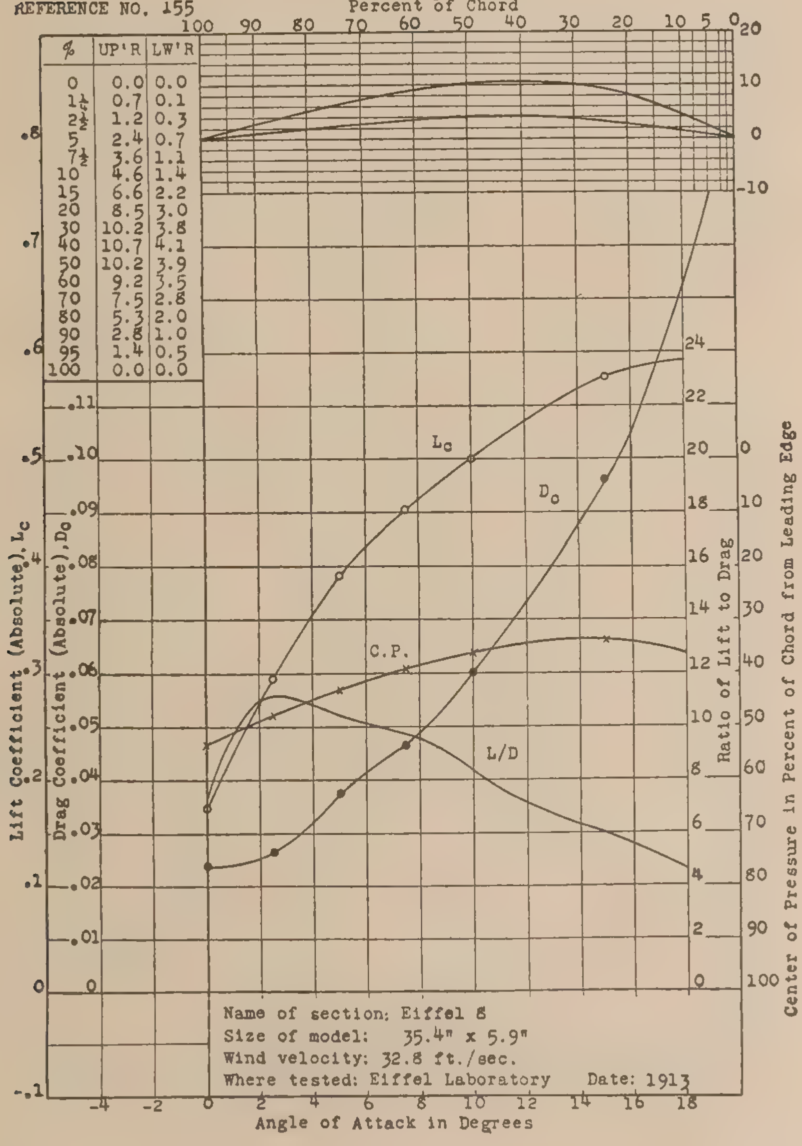
%	UP'R	LW'R
0	Rad.=1.04	
1¼	2.6	0.0
2½	3.6	0.0
5	5.4	0.0
7½	7.1	0.0
10	8.5	0.0
15	11.1	0.0
20	13.0	0.0
30	15.5	0.0
37½	16.0	0.0
50	15.5	0.0
60	14.2	0.0
70	12.2	0.0
80	9.6	0.0
90	6.0	0.0
95	4.0	0.0
100	Rad.=1.04	

REFERENCE NO. 154



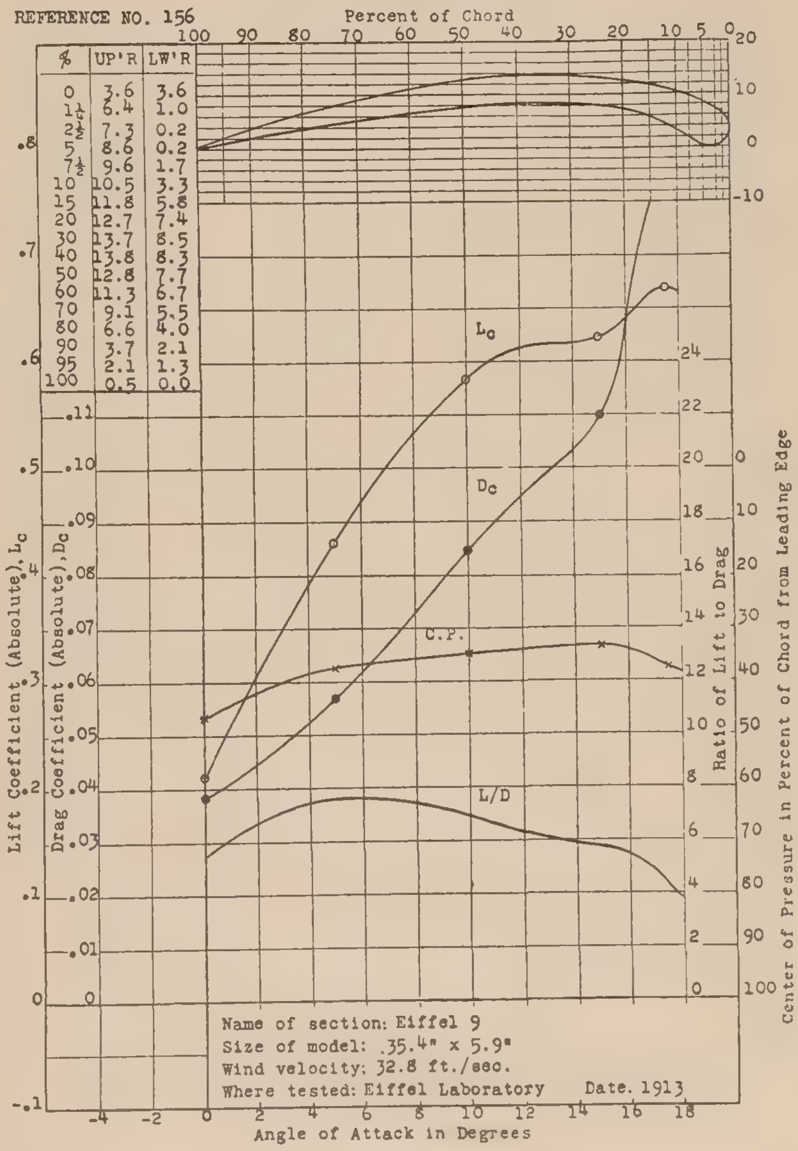
%	UP'R	LW'R
0	Rad.=1.04	
1¼	2.8	0.0
2½	4.2	0.0
5	6.6	0.0
7½	8.8	0.0
10	10.7	0.0
15	13.9	0.0
20	16.3	0.0
30	19.3	0.0
37½	20.0	0.0
50	19.3	0.0
60	17.9	0.0
70	15.5	0.0
80	12.0	0.0
90	7.5	0.0
95	4.7	0.0
100	Rad.=1.04	

REFERENCE NO. 155



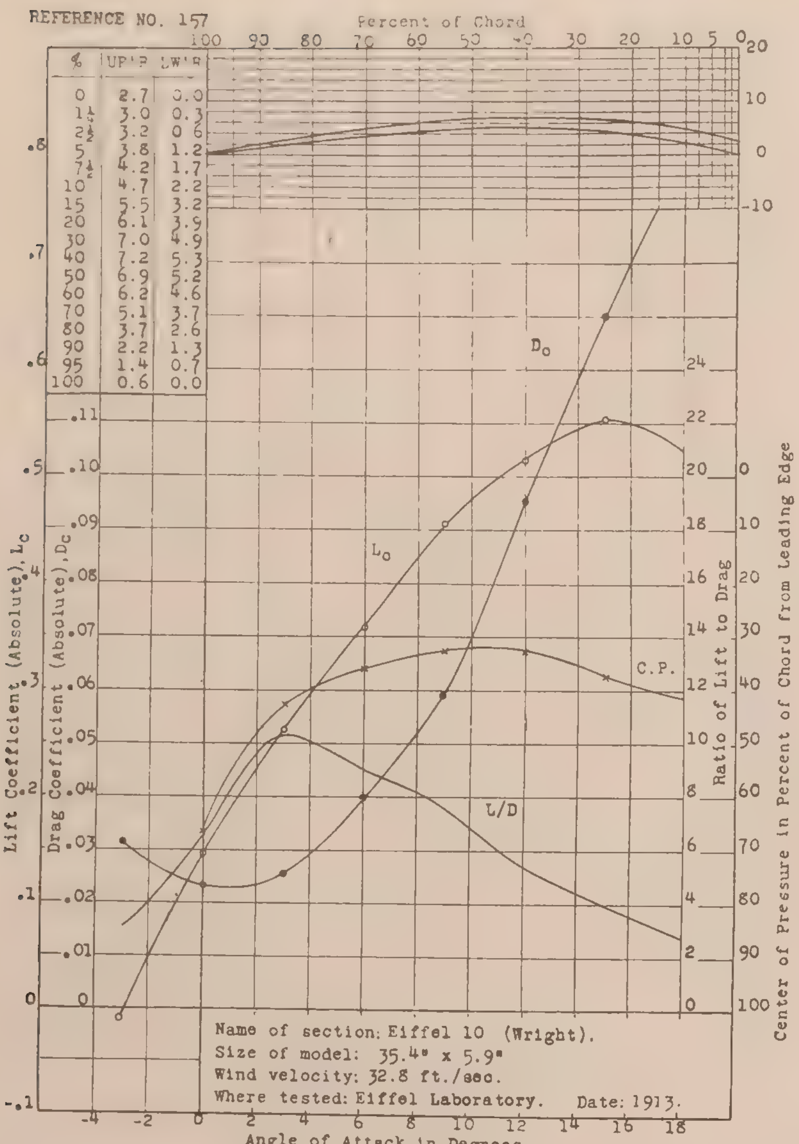
%	UP'R	LW'R
0	0.0	0.0
1¼	0.7	0.1
2½	1.2	0.3
5	2.4	0.7
7½	3.6	1.1
10	4.6	1.4
15	6.6	2.2
20	8.5	3.0
30	10.2	3.8
40	10.7	4.1
50	10.2	3.9
60	9.2	3.5
70	7.5	2.8
80	5.3	2.0
90	2.8	1.0
95	1.4	0.5
100	0.0	0.0

REFERENCE NO. 156



%	UP'R	LW'R
0	3.6	3.6
1¼	6.4	1.0
2½	7.3	0.2
5	8.6	0.2
7½	9.6	1.7
10	10.5	3.3
15	11.8	5.8
20	12.7	7.4
30	13.7	8.5
40	13.8	8.3
50	12.8	7.7
60	11.3	6.7
70	9.1	5.5
80	6.6	4.0
90	3.7	2.1
95	2.1	1.3
100	0.5	0.0

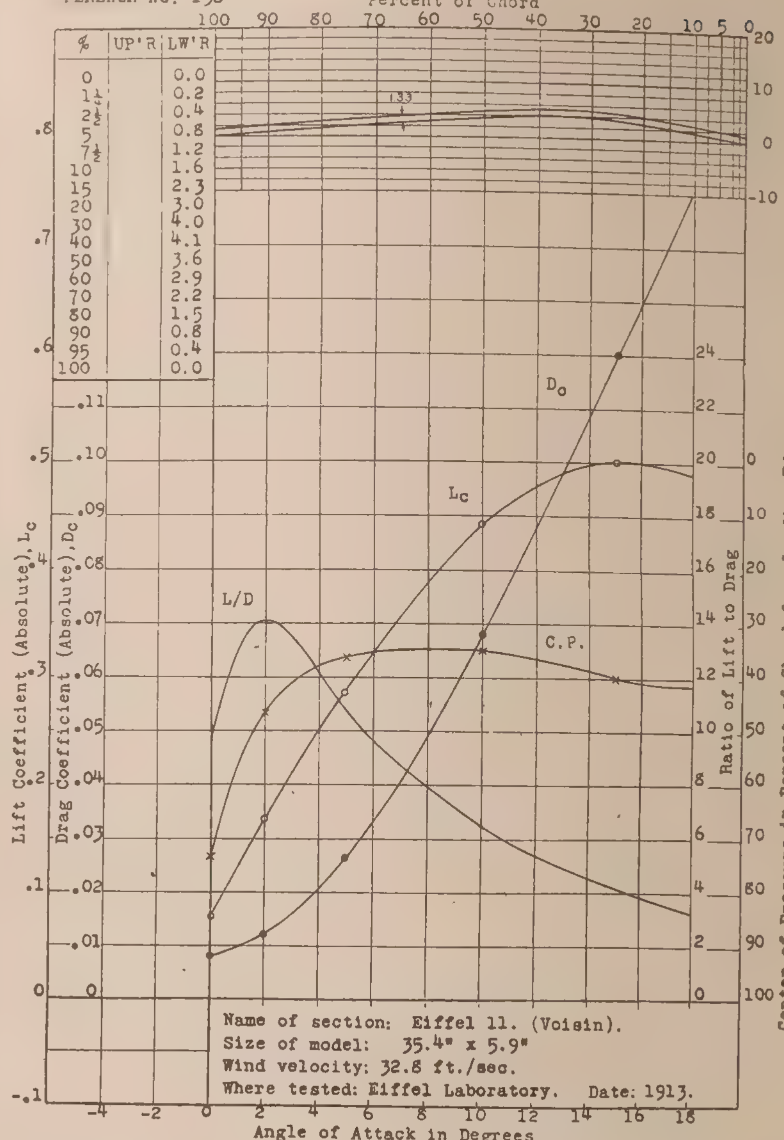
REFERENCE NO. 157



%	UP'R	LW'R
0	2.7	0.0
1¼	3.0	0.3
2½	3.2	0.6
5	3.8	1.2
7½	4.2	1.7
10	4.7	2.2
15	5.5	3.2
20	6.1	3.9
30	7.0	4.9
40	7.2	5.3
50	6.9	5.2
60	6.2	4.6
70	5.1	3.7
80	3.7	2.6
90	2.2	1.3
95	1.4	0.7
100	0.6	0.0

Name of section: Eiffel 10 (Wright).
Size of model: 35.4" x 5.9"
Wind velocity: 32.8 ft./sec.
Where tested: Eiffel Laboratory. Date: 1913.

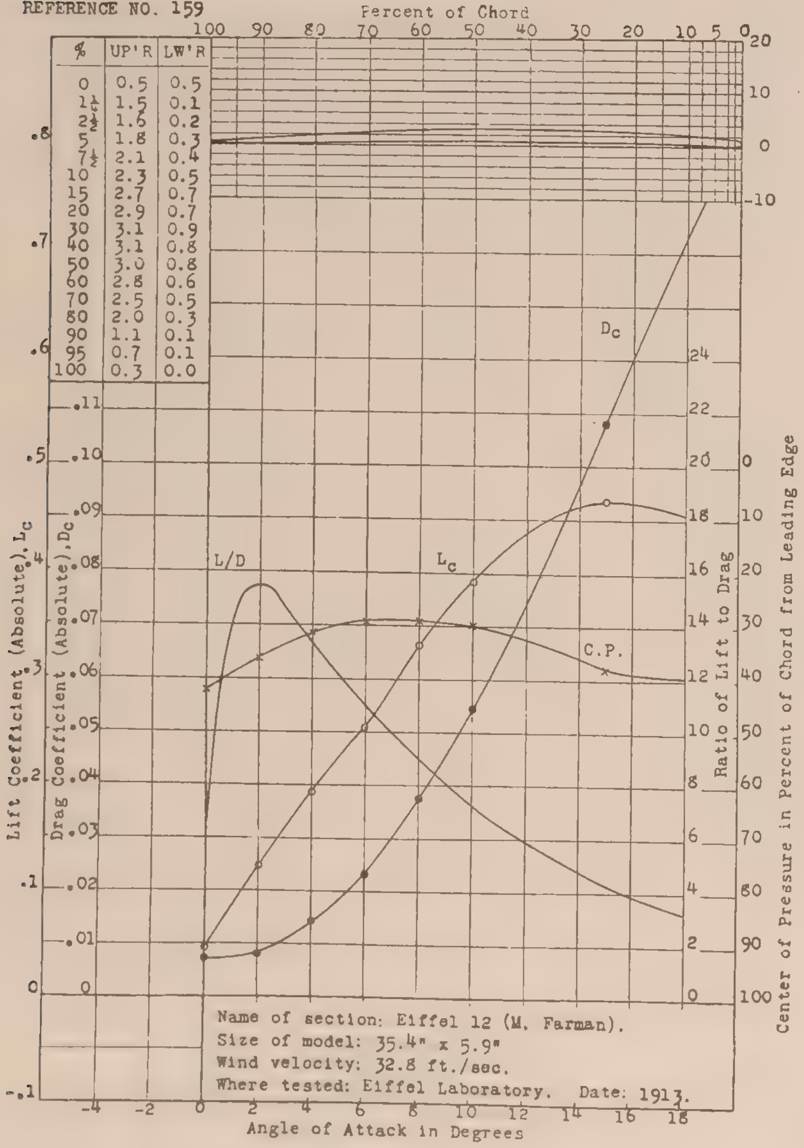
REFERENCE NO. 158



%	UP'R	LW'R
0		0.0
1¼		0.2
2½		0.4
5		0.8
7½		1.2
10		1.6
15		2.3
20		3.0
30		4.0
40		4.1
50		3.6
60		2.9
70		2.2
80		1.5
90		0.8
95		0.4
100		0.0

Name of section: Eiffel 11. (Voisin).
Size of model: 35.4" x 5.9"
Wind velocity: 32.8 ft./sec.
Where tested: Eiffel Laboratory. Date: 1913.

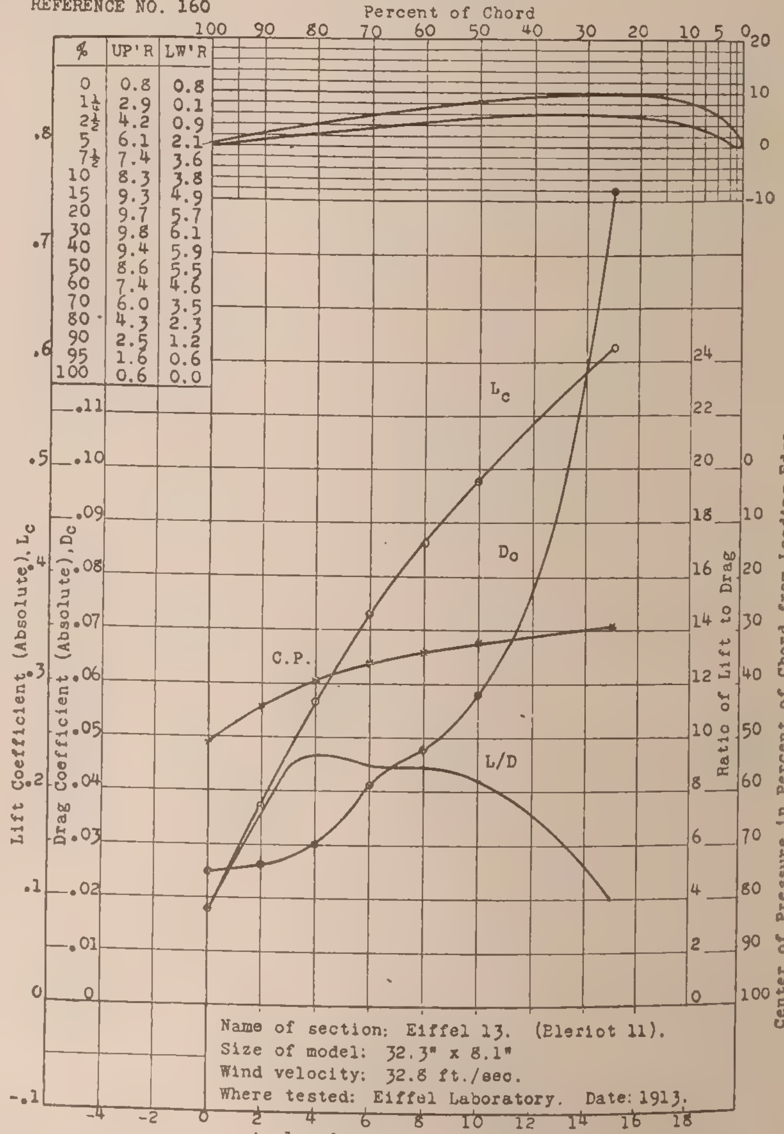
REFERENCE NO. 159



%	UP'R	LW'R
0	0.5	0.5
1¼	1.5	0.1
2½	1.6	0.2
5	1.8	0.3
7½	2.1	0.4
10	2.3	0.5
15	2.7	0.7
20	2.9	0.7
30	3.1	0.9
40	3.1	0.8
50	3.0	0.8
60	2.8	0.6
70	2.5	0.5
80	2.0	0.3
90	1.1	0.1
95	0.7	0.1
100	0.3	0.0

Name of section: Eiffel 12 (M. Farman).
Size of model: 35.4" x 5.9"
Wind velocity: 32.8 ft./sec.
Where tested: Eiffel Laboratory. Date: 1913.

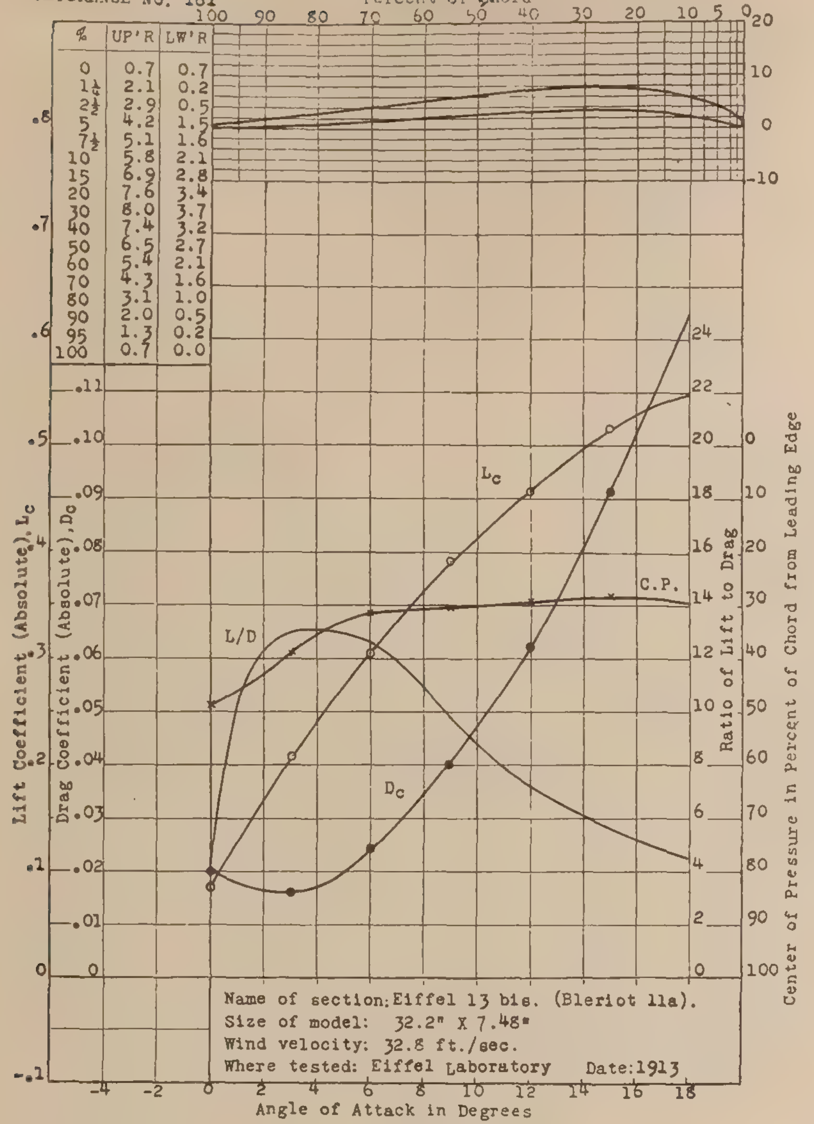
REFERENCE NO. 160



%	UP'R	LW'R
0	0.8	0.8
1¼	2.9	0.1
2½	4.2	0.9
5	6.1	2.1
7½	7.4	3.6
10	8.3	3.8
15	9.3	4.9
20	9.7	5.7
30	9.8	6.1
40	9.4	5.9
50	8.6	5.5
60	7.4	4.6
70	6.0	3.5
80	4.3	2.3
90	2.5	1.2
95	1.6	0.6
100	0.6	0.0

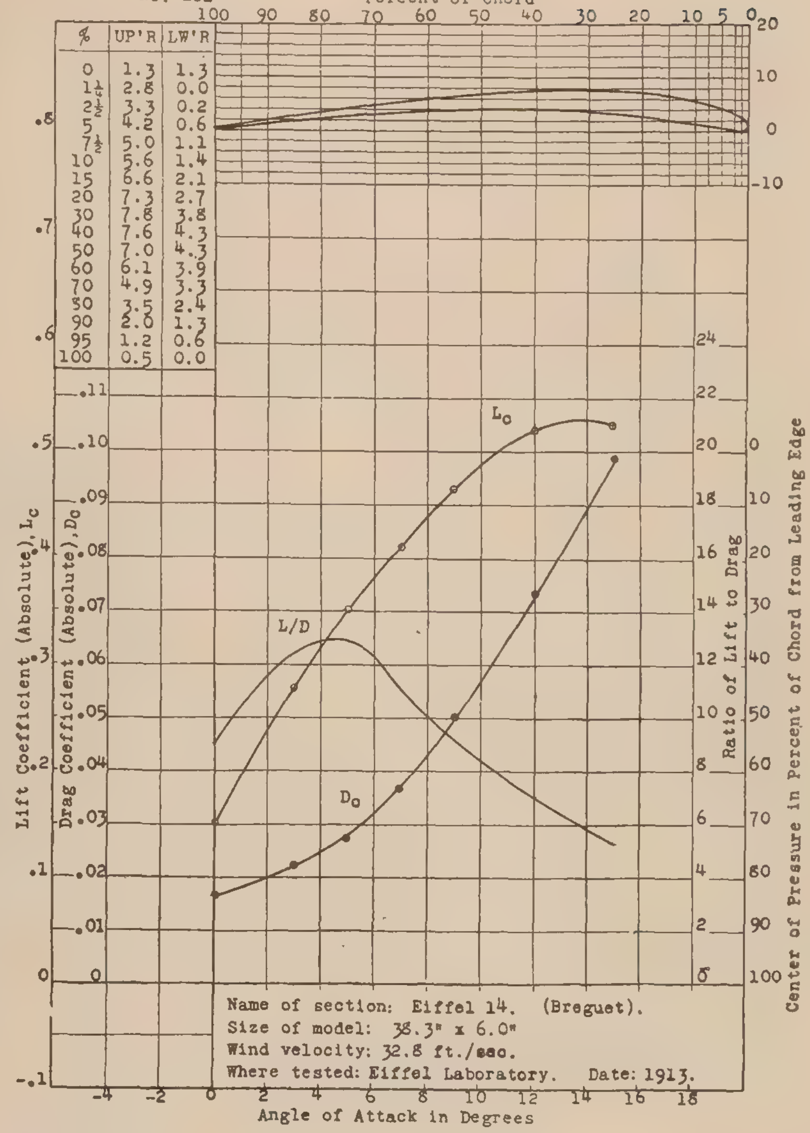
Name of section: Eiffel 13. (Bleriot 11).
Size of model: 32.3" x 8.1"
Wind velocity: 32.8 ft./sec.
Where tested: Eiffel Laboratory. Date: 1913.

REFERENCE NO. 161



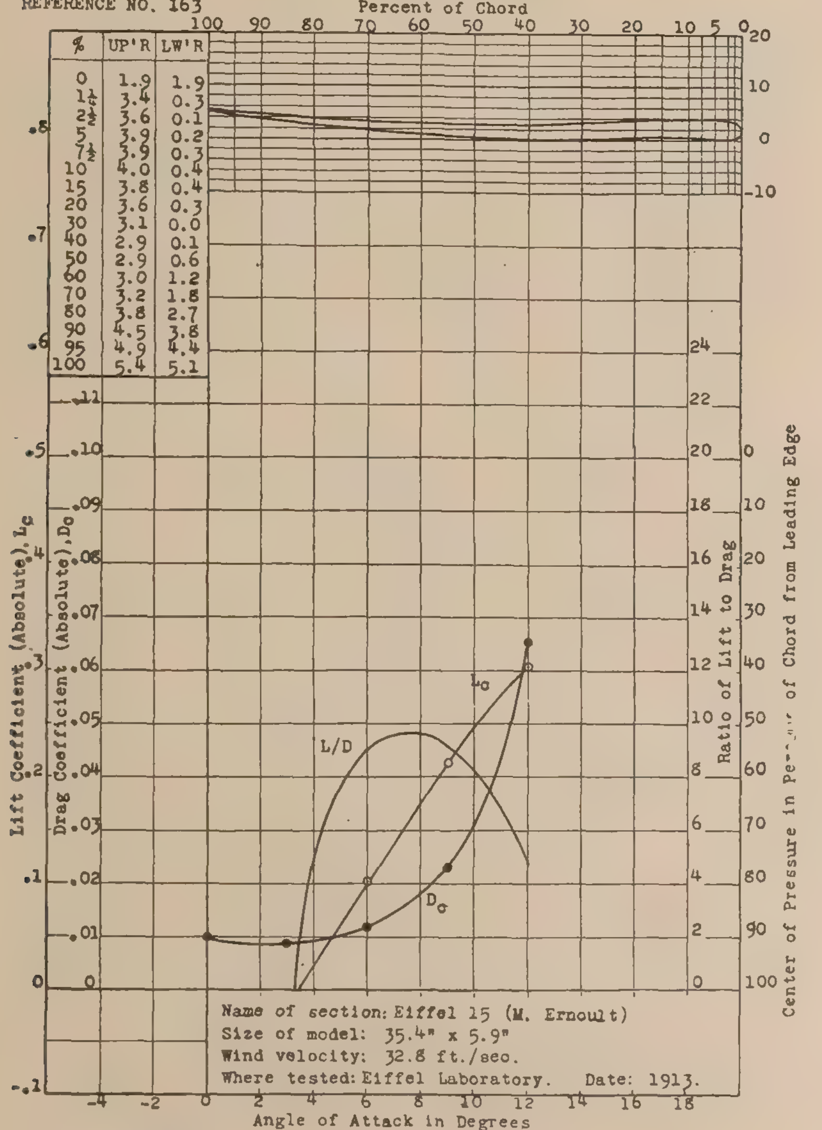
%	UP'R	LW'R
0	0.7	0.7
1¼	2.1	0.2
2½	2.9	0.5
5	4.2	1.5
7½	5.1	1.6
10	5.8	2.1
15	6.9	2.8
20	7.6	3.4
30	8.0	3.7
40	7.4	3.2
50	6.5	2.7
60	5.4	2.1
70	4.3	1.6
80	3.1	1.0
90	2.0	0.5
95	1.3	0.2
100	0.7	0.0

REFERENCE NO. 162



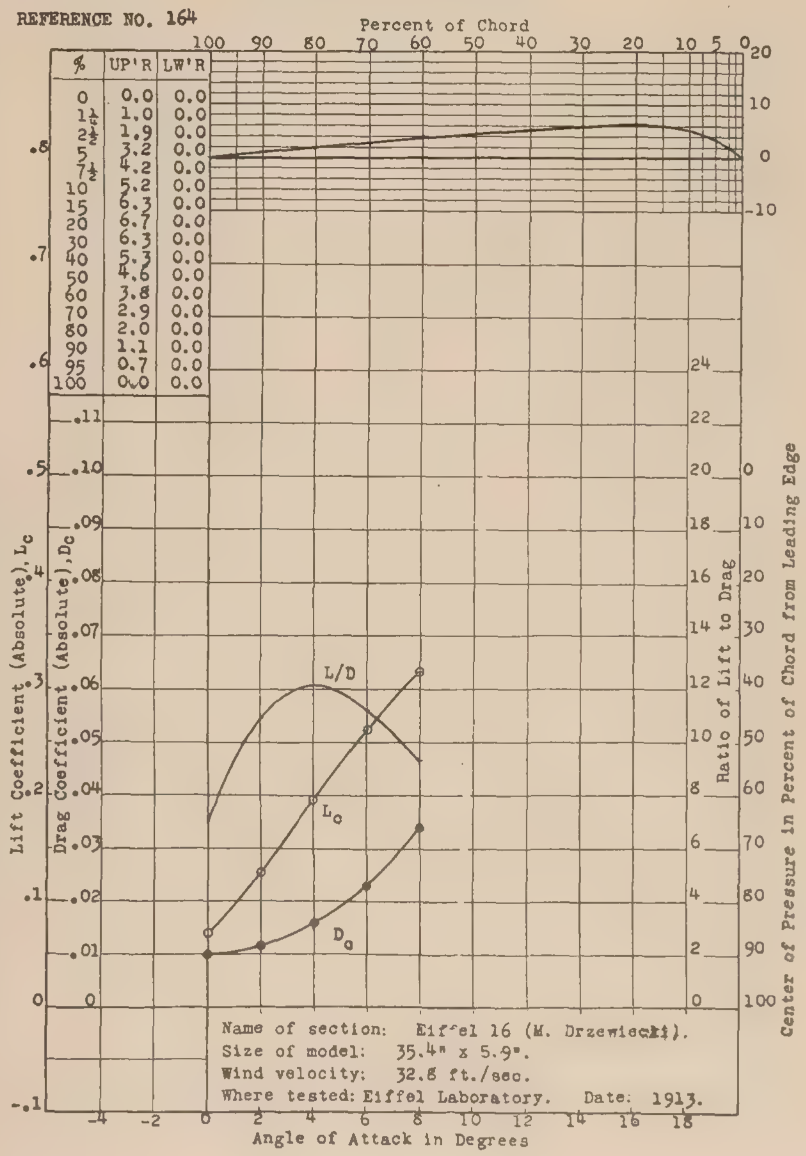
%	UP'R	LW'R
0	1.3	1.3
1¼	2.8	0.0
2½	3.3	0.2
5	4.2	0.6
7½	5.0	1.1
10	5.6	1.4
15	6.6	2.1
20	7.3	2.7
30	7.8	3.8
40	7.6	4.3
50	7.0	4.3
60	6.1	3.9
70	4.9	3.3
80	3.5	2.4
90	2.0	1.3
95	1.2	0.6
100	0.5	0.0

REFERENCE NO. 163



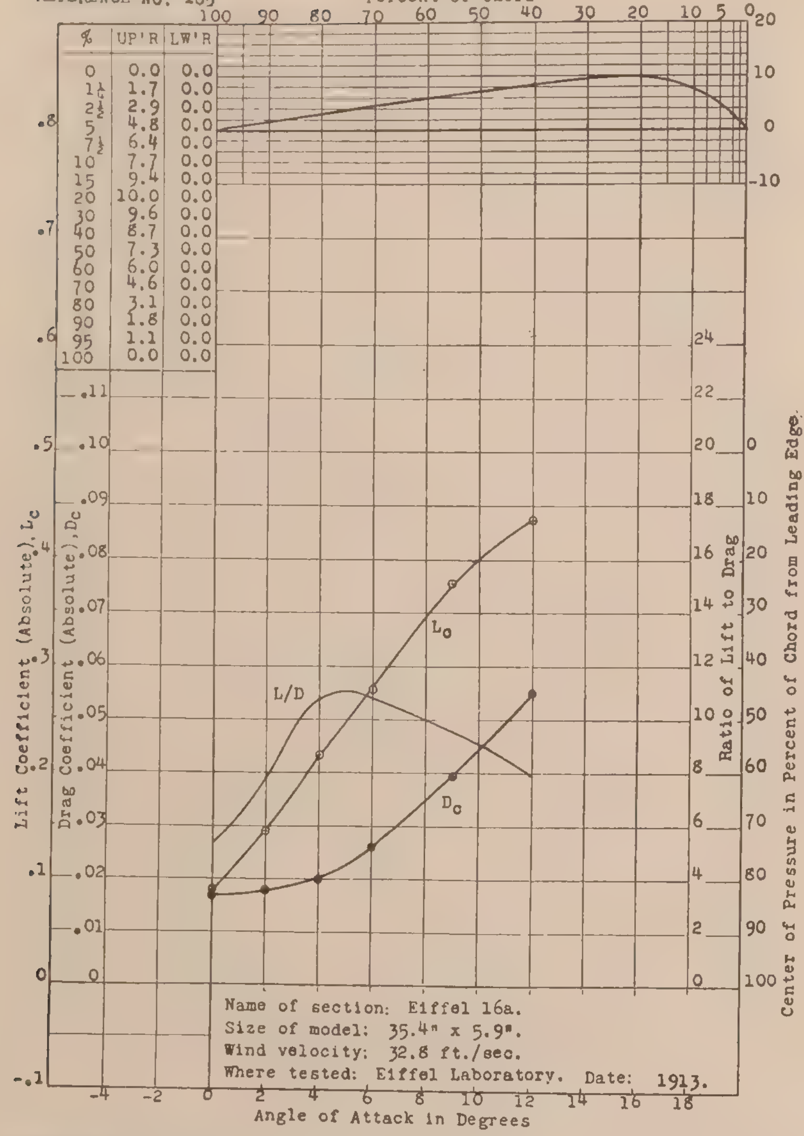
%	UP'R	LW'R
0	1.9	1.9
1¼	3.4	0.3
2½	3.6	0.1
5	3.9	0.2
7½	3.9	0.3
10	4.0	0.4
15	3.8	0.4
20	3.6	0.3
30	3.1	0.0
40	2.9	0.1
50	2.9	0.6
60	3.0	1.2
70	3.2	1.8
80	3.8	2.7
90	4.5	3.8
95	4.9	4.4
100	5.4	5.1

REFERENCE NO. 164



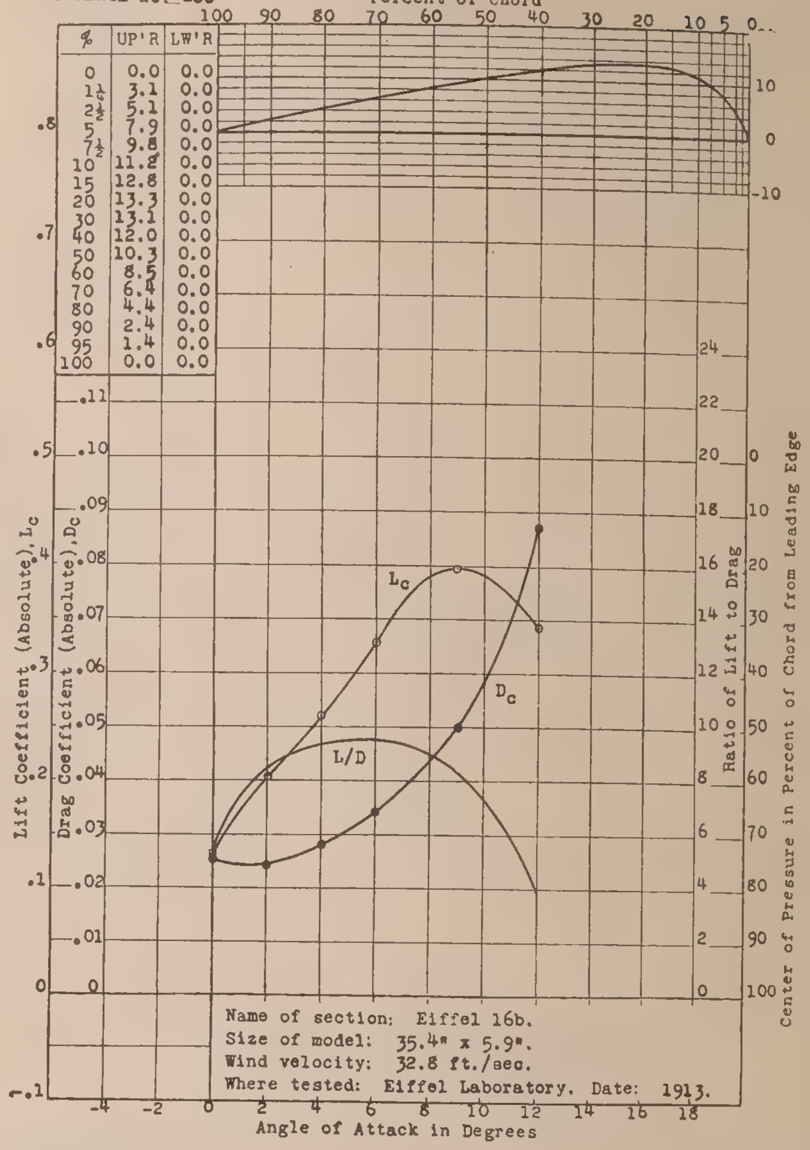
%	UP'R	LW'R
0	0.0	0.0
1¼	1.0	0.0
2½	1.9	0.0
5	3.2	0.0
7½	4.2	0.0
10	5.2	0.0
15	6.3	0.0
20	6.7	0.0
30	6.3	0.0
40	5.3	0.0
50	4.6	0.0
60	3.8	0.0
70	2.9	0.0
80	2.0	0.0
90	1.1	0.0
95	0.7	0.0
100	0.0	0.0

REFERENCE NO. 165



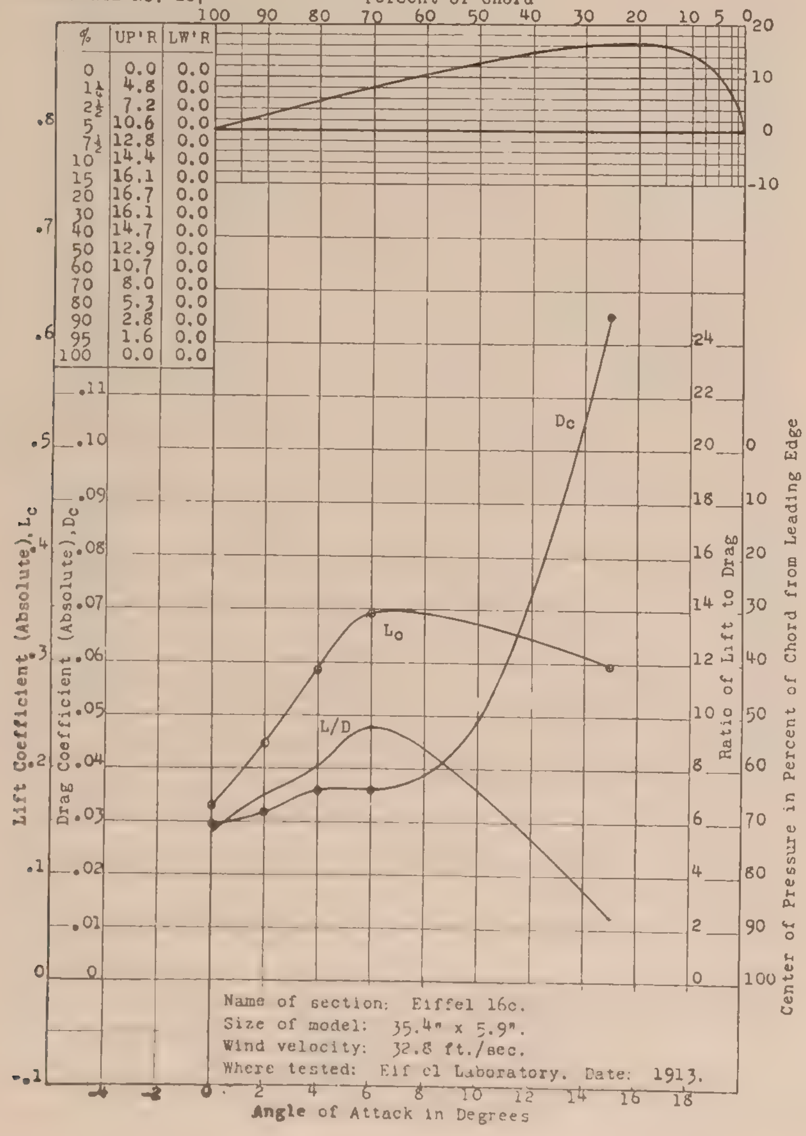
%	UP'R	LW'R
0	0.0	0.0
$1\frac{1}{4}$	1.7	0.0
$2\frac{1}{2}$	2.9	0.0
5	4.8	0.0
$7\frac{1}{2}$	6.4	0.0
10	7.7	0.0
15	9.4	0.0
20	10.0	0.0
30	9.6	0.0
40	8.7	0.0
50	7.3	0.0
60	6.0	0.0
70	4.6	0.0
80	3.1	0.0
90	1.8	0.0
95	1.1	0.0
100	0.0	0.0

REFERENCE NO. 166



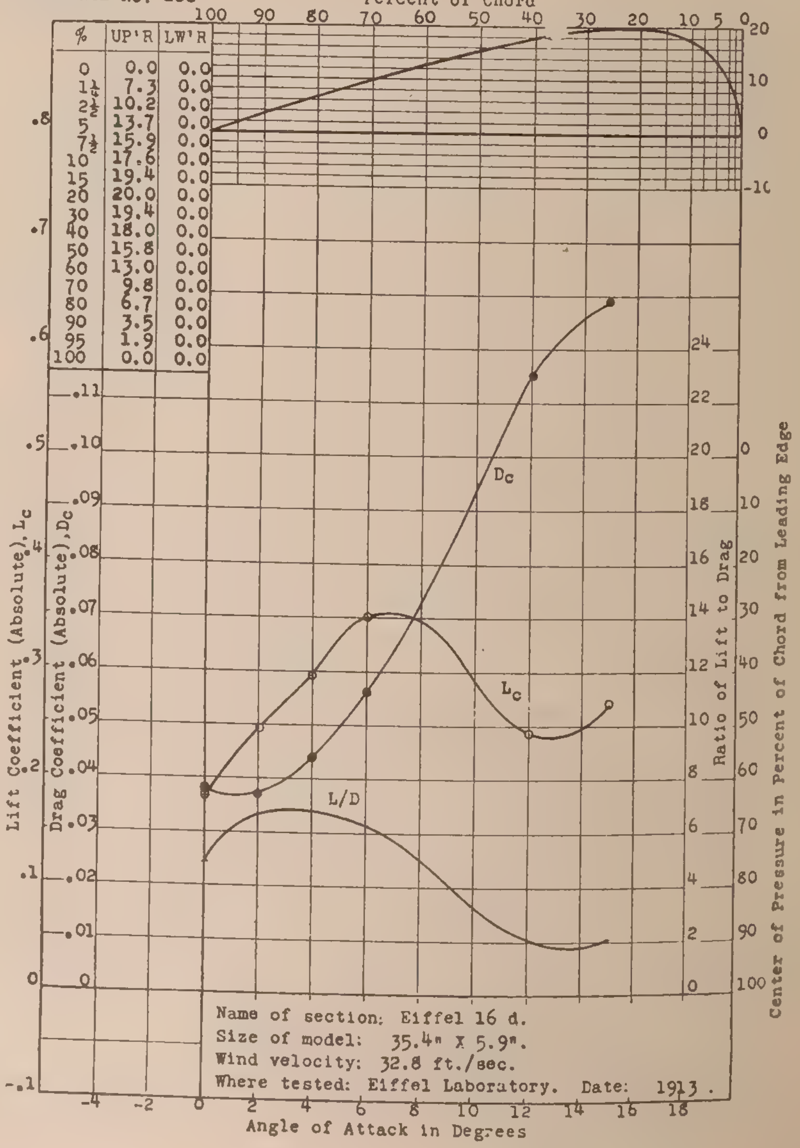
%	UP'R	LW'R
0	0.0	0.0
$1\frac{1}{4}$	3.1	0.0
$2\frac{1}{2}$	5.1	0.0
5	7.9	0.0
$7\frac{1}{2}$	9.8	0.0
10	11.2	0.0
15	12.8	0.0
20	13.3	0.0
30	13.1	0.0
40	12.0	0.0
50	10.3	0.0
60	8.5	0.0
70	6.4	0.0
80	4.4	0.0
90	2.4	0.0
95	1.4	0.0
100	0.0	0.0

REFERENCE NO. 167



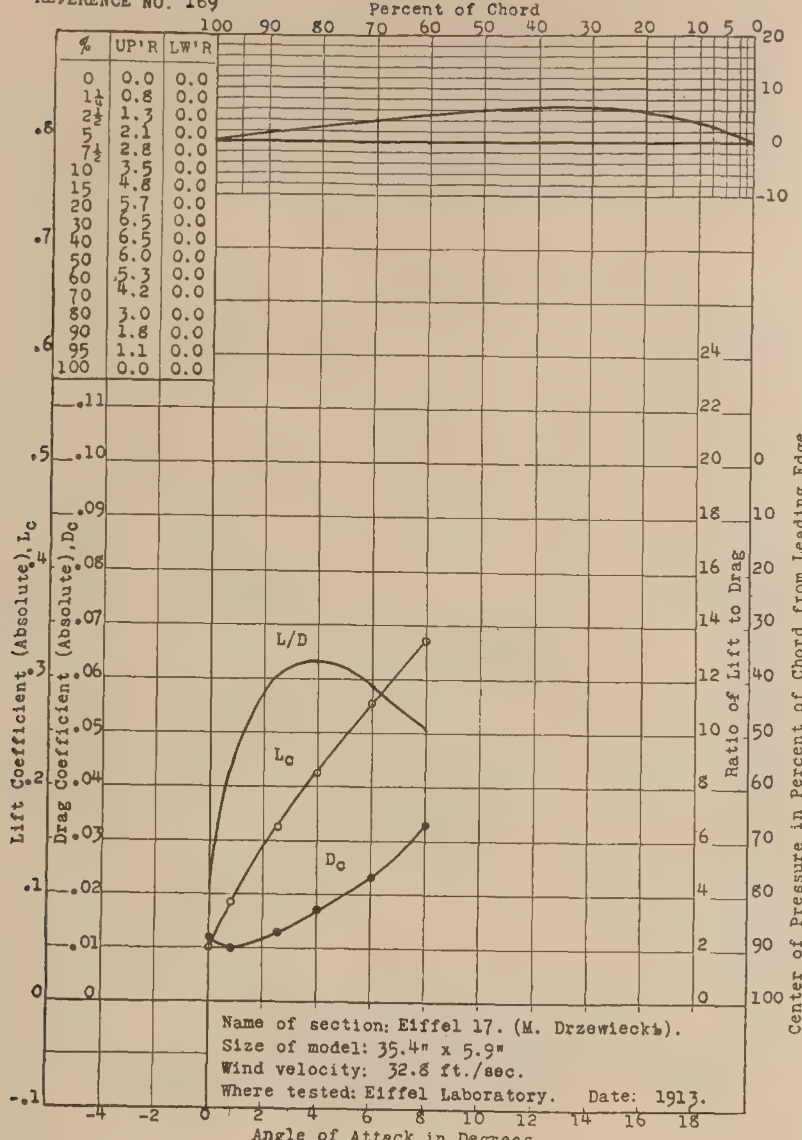
%	UP'R	LW'R
0	0.0	0.0
$1\frac{1}{4}$	4.8	0.0
$2\frac{1}{2}$	7.2	0.0
5	10.6	0.0
$7\frac{1}{2}$	12.8	0.0
10	14.4	0.0
15	16.1	0.0
20	16.7	0.0
30	16.1	0.0
40	14.7	0.0
50	12.9	0.0
60	10.7	0.0
70	8.0	0.0
80	5.3	0.0
90	2.8	0.0
95	1.6	0.0
100	0.0	0.0

REFERENCE NO. 168



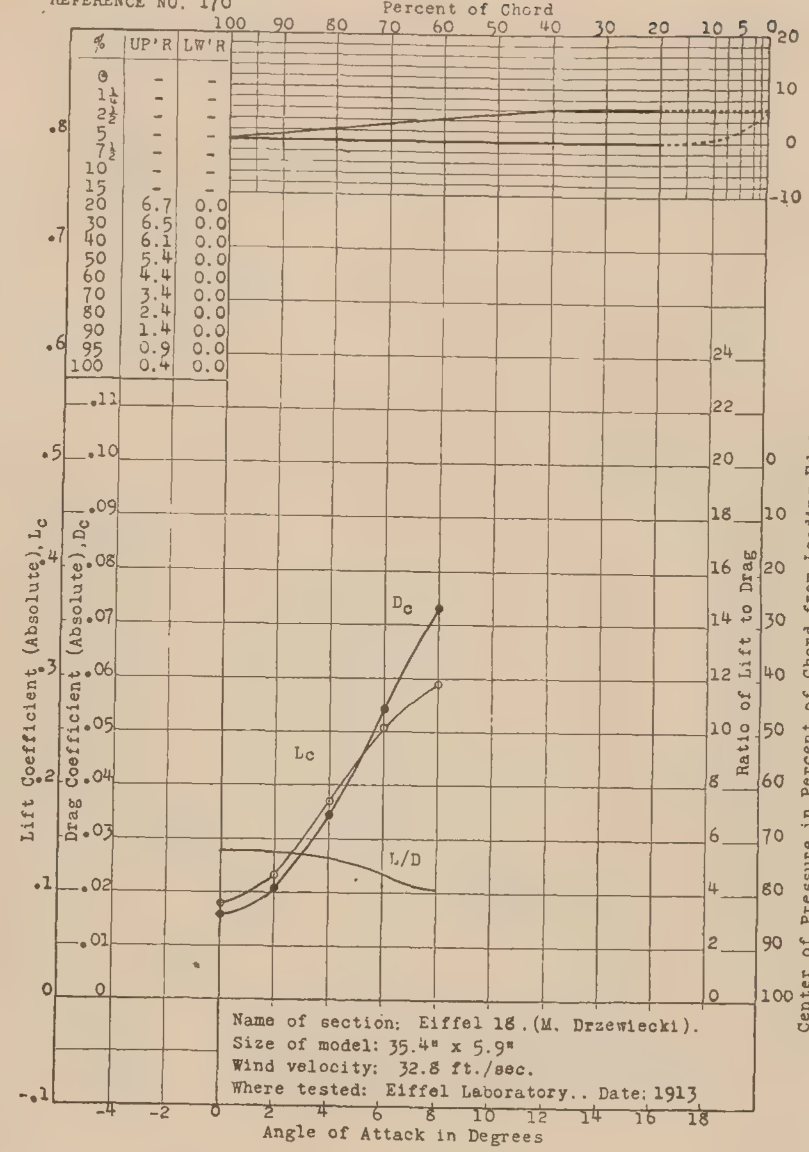
%	UP'R	LW'R
0	0.0	0.0
$1\frac{1}{4}$	7.3	0.0
$2\frac{1}{2}$	10.2	0.0
5	13.7	0.0
$7\frac{1}{2}$	15.9	0.0
10	17.6	0.0
15	19.4	0.0
20	20.0	0.0
30	19.4	0.0
40	18.0	0.0
50	15.8	0.0
60	13.0	0.0
70	9.8	0.0
80	6.7	0.0
90	3.5	0.0
95	1.9	0.0
100	0.0	0.0

REFERENCE NO. 169



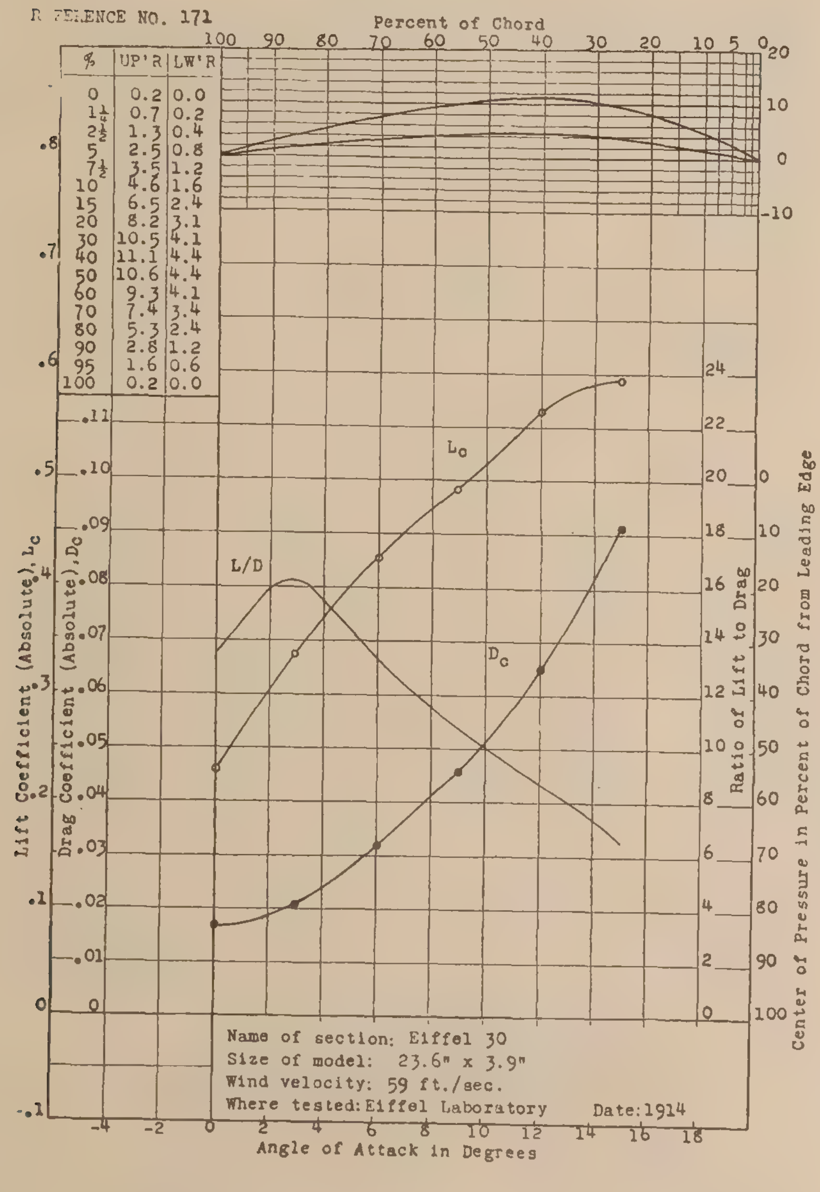
%	UP'R	LW'R
0	0.0	0.0
1¼	0.8	0.0
2½	1.3	0.0
5	2.1	0.0
7½	2.8	0.0
10	3.5	0.0
15	4.8	0.0
20	5.7	0.0
30	6.5	0.0
40	6.5	0.0
50	6.0	0.0
60	5.3	0.0
70	4.2	0.0
80	3.0	0.0
90	1.8	0.0
95	1.1	0.0
100	0.0	0.0

REFERENCE NO. 170



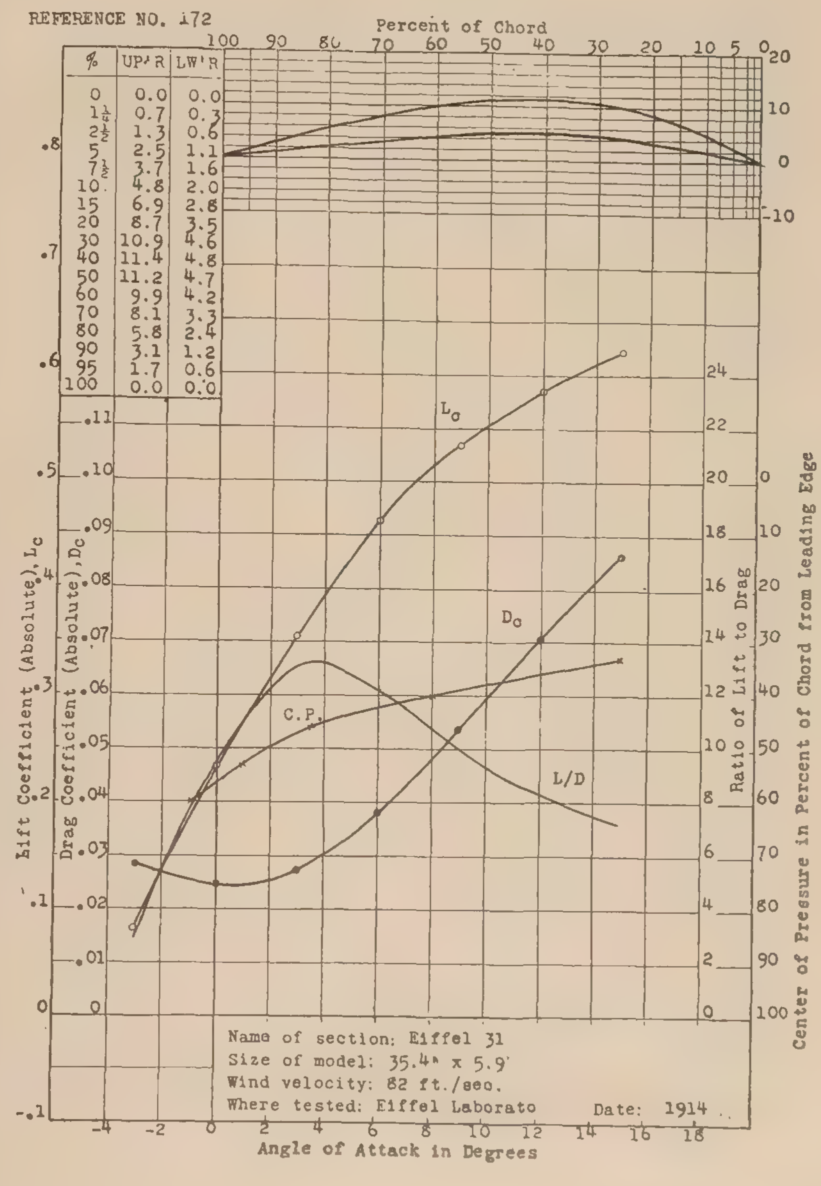
%	UP'R	LW'R
0	-	-
1¼	-	-
2½	-	-
5	-	-
7½	-	-
10	-	-
15	-	-
20	6.7	0.0
30	6.5	0.0
40	6.1	0.0
50	5.4	0.0
60	4.4	0.0
70	3.4	0.0
80	2.4	0.0
90	1.4	0.0
95	0.9	0.0
100	0.4	0.0

REFERENCE NO. 171



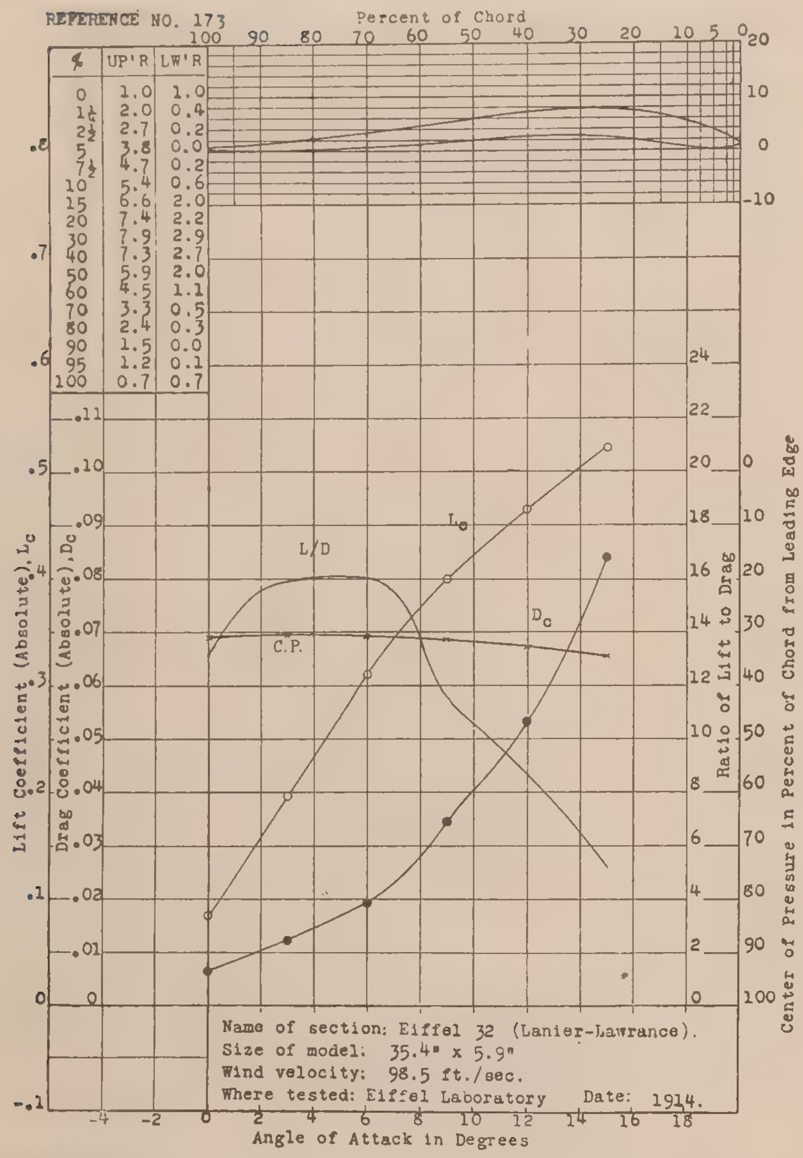
%	UP'R	LW'R
0	0.2	0.0
1¼	0.7	0.2
2½	1.3	0.4
5	2.5	0.8
7½	3.5	1.2
10	4.6	1.6
15	6.5	2.4
20	8.2	3.1
30	10.5	4.1
40	11.1	4.4
50	10.6	4.4
60	9.3	4.1
70	7.4	3.4
80	5.3	2.4
90	2.8	1.2
95	1.6	0.6
100	0.2	0.0

REFERENCE NO. 172



%	UP'R	LW'R
0	0.0	0.0
1¼	0.7	0.3
2½	1.3	0.6
5	2.5	1.1
7½	3.7	1.6
10	4.8	2.0
15	6.9	2.8
20	8.7	3.5
30	10.9	4.6
40	11.4	4.8
50	11.2	4.7
60	9.9	4.2
70	8.1	3.3
80	5.8	2.4
90	3.1	1.2
95	1.7	0.6
100	0.0	0.0

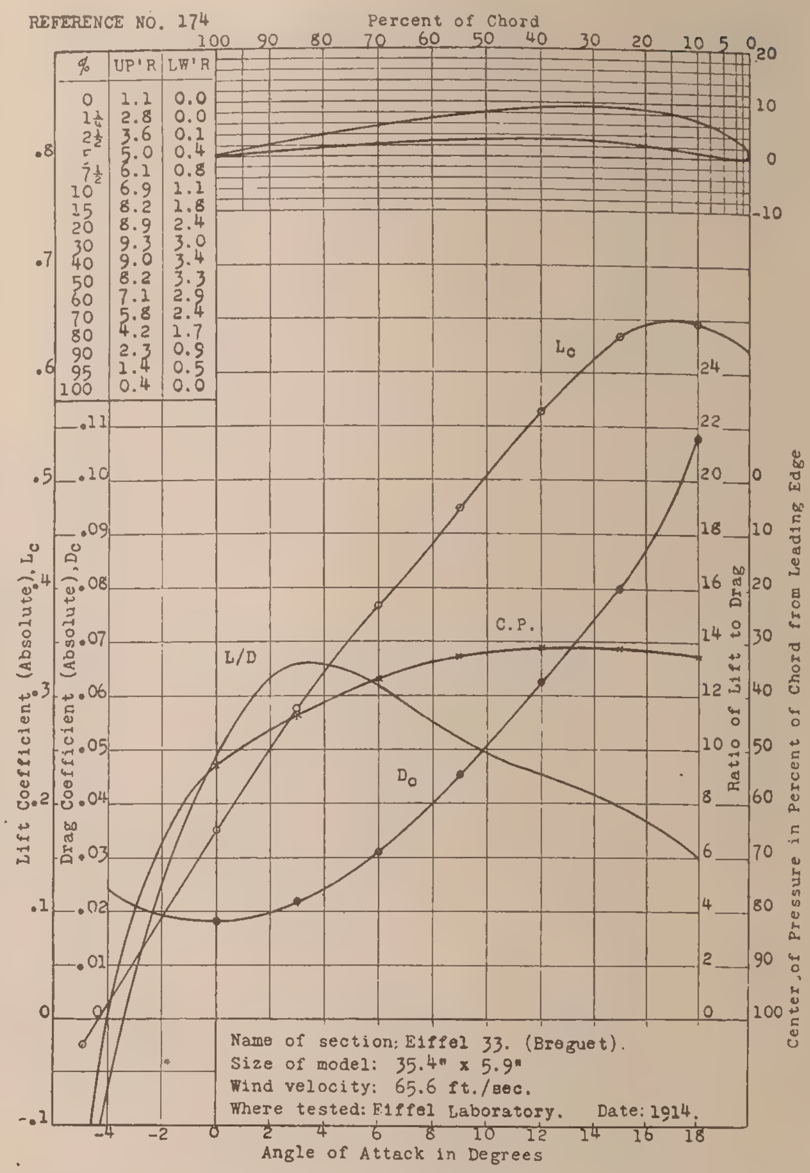
REFERENCE NO. 173



%	UP'R	LW'R
0	1.0	1.0
1½	2.0	0.4
2½	2.7	0.2
5	3.8	0.0
7½	4.7	0.2
10	5.4	0.6
15	6.6	2.0
20	7.4	2.2
30	7.9	2.9
40	7.3	2.7
50	5.9	2.0
60	4.5	1.1
70	3.3	0.5
80	2.4	0.3
90	1.5	0.0
95	1.2	0.1
100	0.7	0.7

Name of section: Eiffel 32 (Lanier-Lawrance).
Size of model: 35.4" x 5.9"
Wind velocity: 98.5 ft./sec.
Where tested: Eiffel Laboratory Date: 1914.

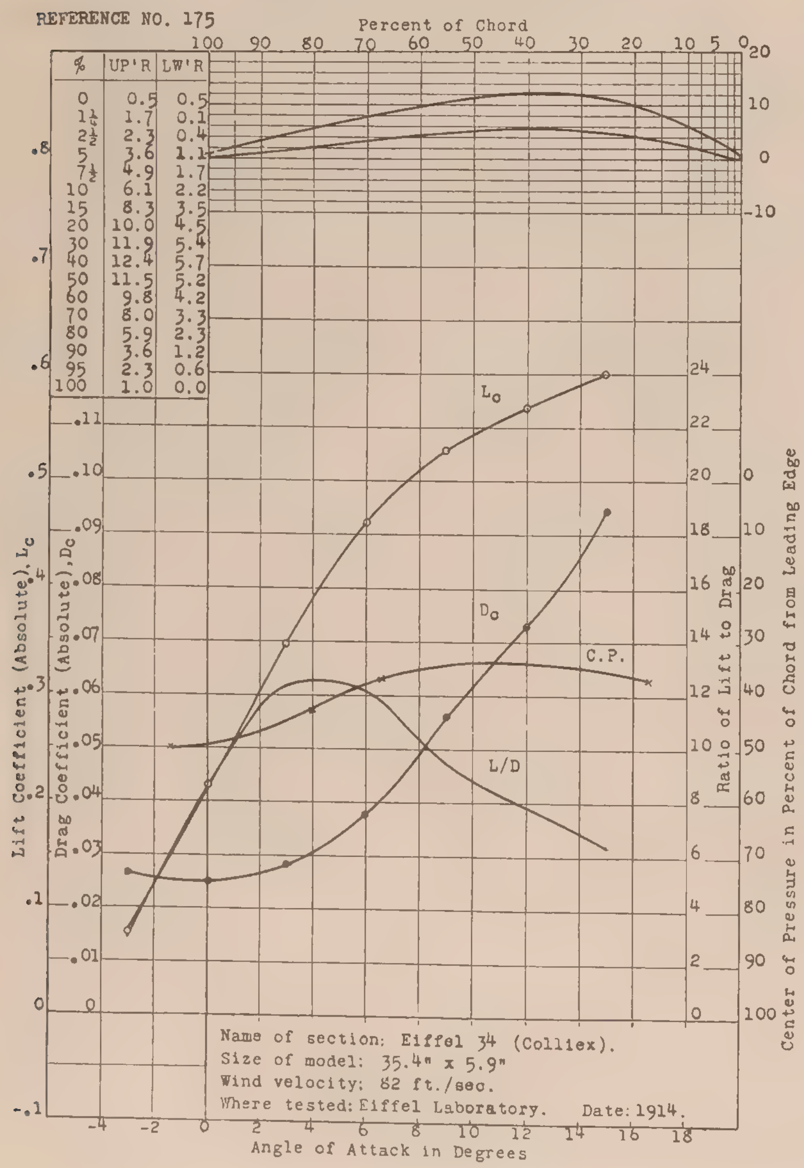
REFERENCE NO. 174



%	UP'R	LW'R
0	1.1	0.0
1½	2.8	0.0
2½	3.6	0.1
5	5.0	0.4
7½	6.1	0.8
10	6.9	1.1
15	8.2	1.8
20	8.9	2.4
30	9.3	3.0
40	9.0	3.4
50	8.2	3.3
60	7.1	2.9
70	5.8	2.4
80	4.2	1.7
90	2.3	0.9
95	1.4	0.5
100	0.4	0.0

Name of section: Eiffel 33. (Breguet).
Size of model: 35.4" x 5.9"
Wind velocity: 65.6 ft./sec.
Where tested: Eiffel Laboratory. Date: 1914.

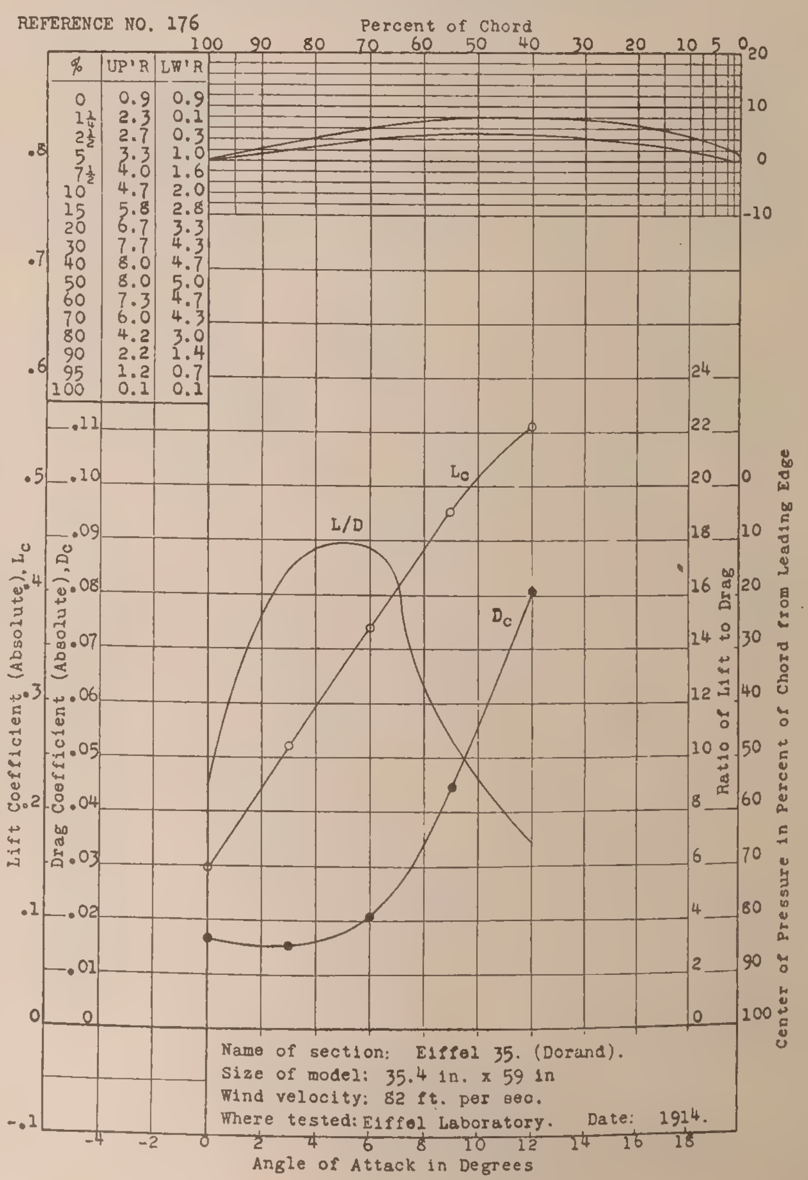
REFERENCE NO. 175



%	UP'R	LW'R
0	0.5	0.5
1½	1.7	0.1
2½	2.3	0.4
5	3.6	1.1
7½	4.9	1.7
10	6.1	2.2
15	8.3	3.5
20	10.0	4.5
30	11.9	5.4
40	12.4	5.7
50	11.5	5.2
60	9.8	4.2
70	8.0	3.3
80	5.9	2.3
90	3.6	1.2
95	2.3	0.6
100	1.0	0.0

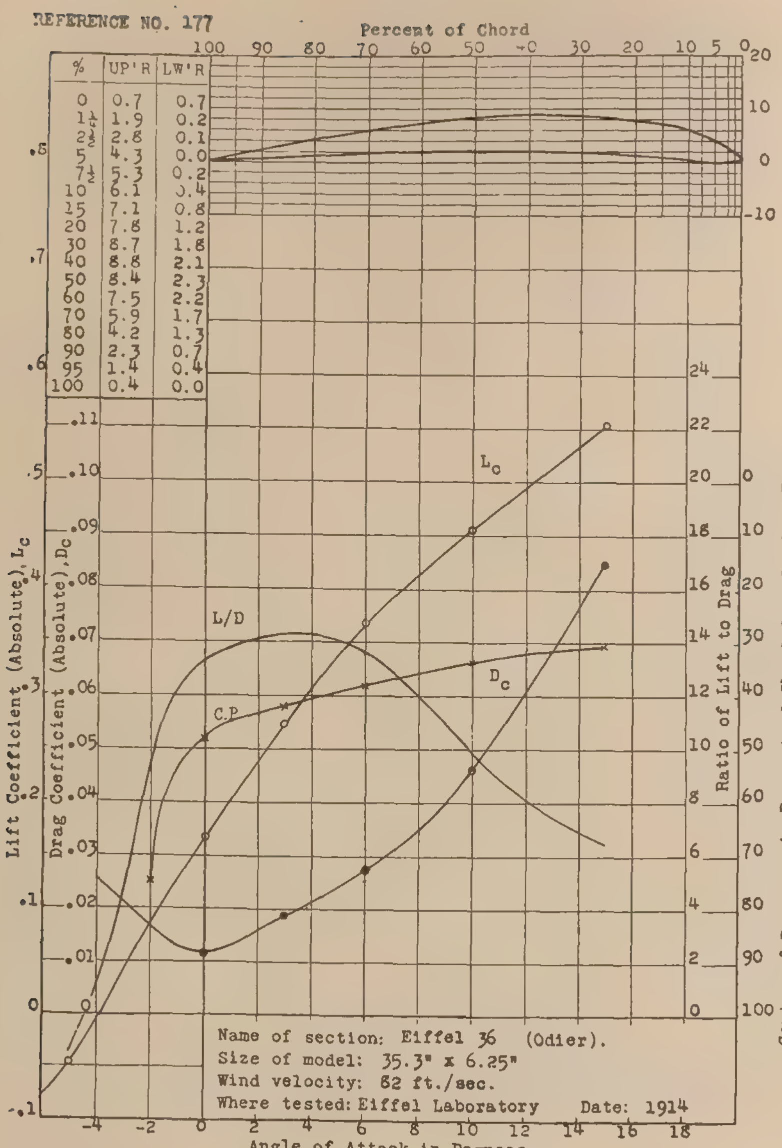
Name of section: Eiffel 34 (Colliex).
Size of model: 35.4" x 5.9"
Wind velocity: 82 ft./sec.
Where tested: Eiffel Laboratory. Date: 1914.

REFERENCE NO. 176

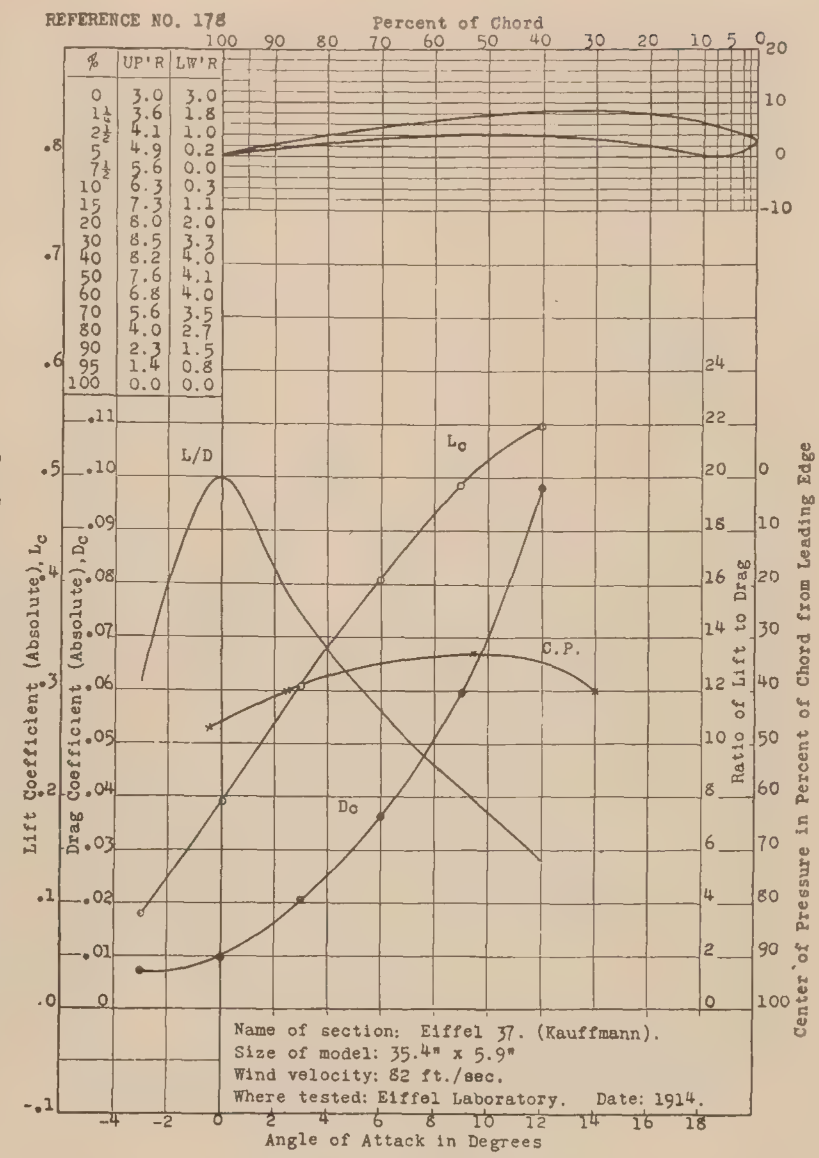


%	UP'R	LW'R
0	0.9	0.9
1½	2.3	0.1
2½	2.7	0.3
5	3.3	1.0
7½	4.0	1.6
10	4.7	2.0
15	5.8	2.8
20	6.7	3.3
30	7.7	4.3
40	8.0	4.7
50	8.0	5.0
60	7.3	4.7
70	6.0	4.3
80	4.2	3.0
90	2.2	1.4
95	1.2	0.7
100	0.1	0.1

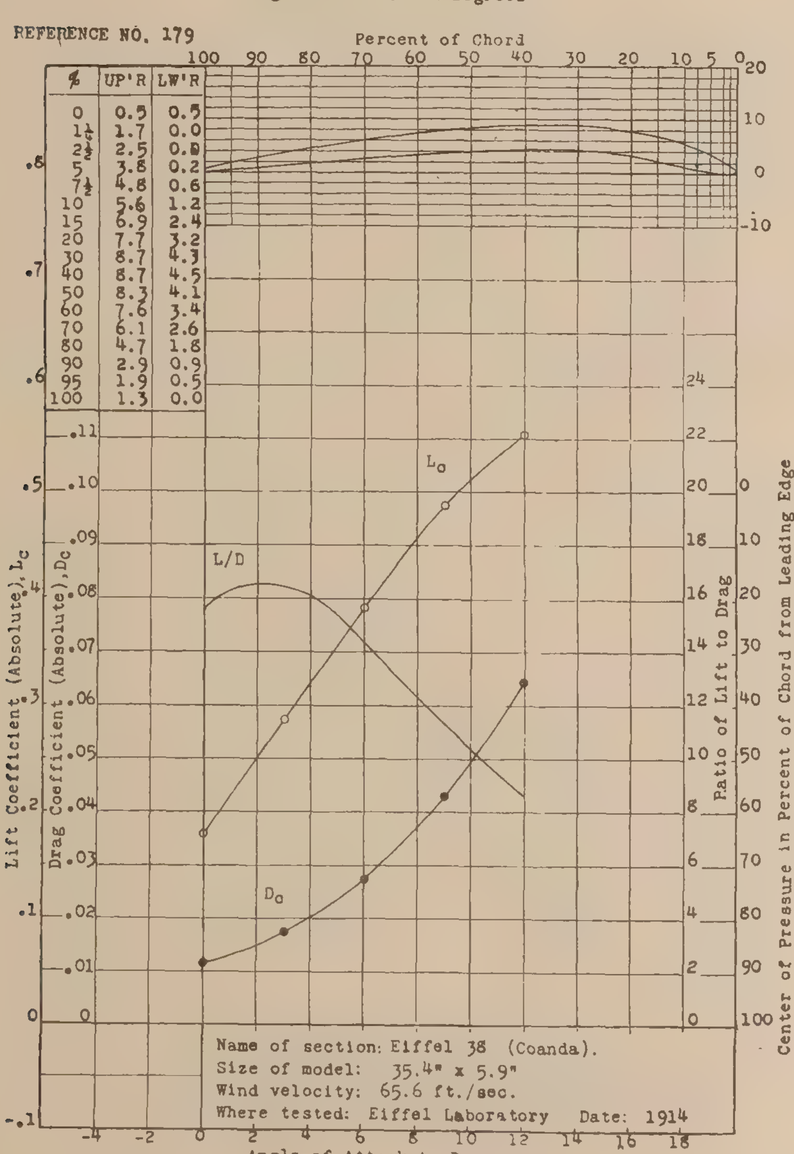
Name of section: Eiffel 35. (Dorand).
Size of model: 35.4 in. x 59 in
Wind velocity: 82 ft. per sec.
Where tested: Eiffel Laboratory. Date: 1914.



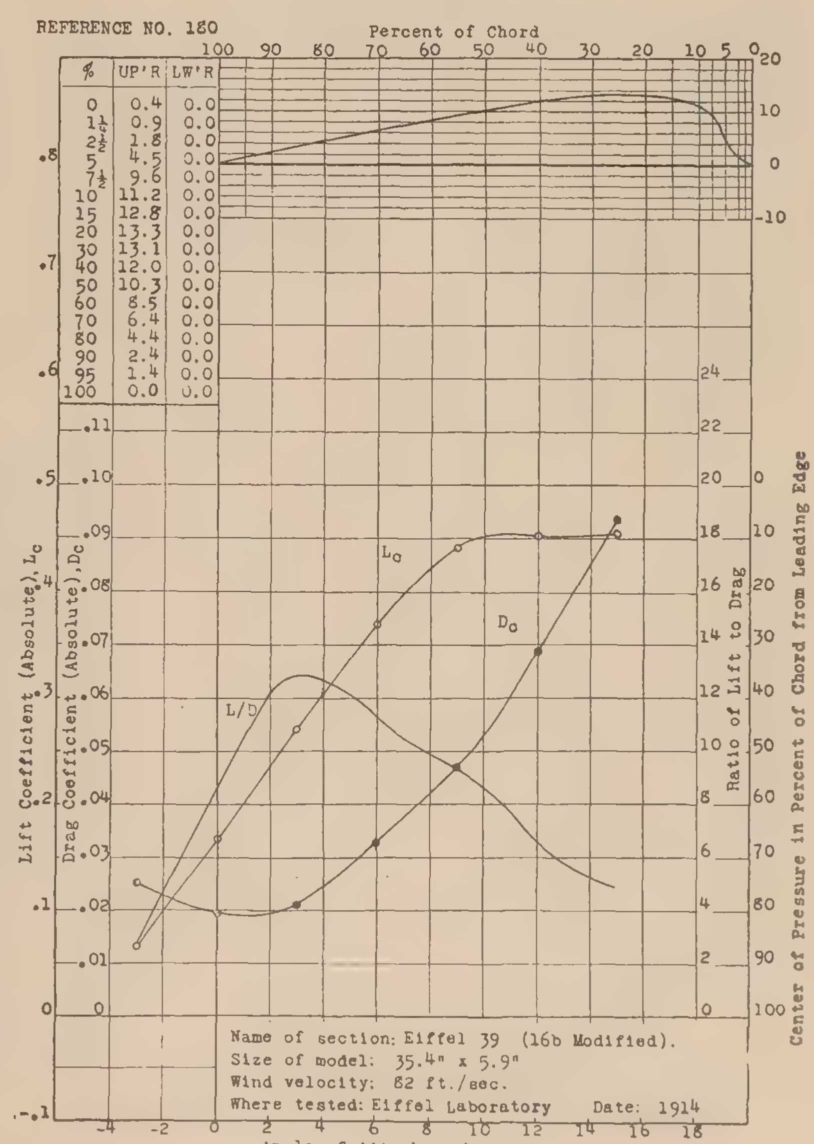
%	UP'R	LW'R
0	0.7	0.7
1½	1.9	0.2
2½	2.8	0.1
5	4.3	0.0
7½	5.3	0.2
10	6.1	0.4
15	7.1	0.8
20	7.8	1.2
30	8.7	1.8
40	8.8	2.1
50	8.4	2.3
60	7.5	2.2
70	5.9	1.7
80	4.2	1.3
90	2.3	0.7
95	1.4	0.4
100	0.4	0.0



%	UP'R	LW'R
0	3.0	3.0
1½	3.6	1.8
2½	4.1	1.0
5	4.9	0.2
7½	5.6	0.0
10	6.3	0.3
15	7.3	1.1
20	8.0	2.0
30	8.5	3.3
40	8.2	4.0
50	7.6	4.1
60	6.8	4.0
70	5.6	3.5
80	4.0	2.7
90	2.3	1.5
95	1.4	0.8
100	0.0	0.0

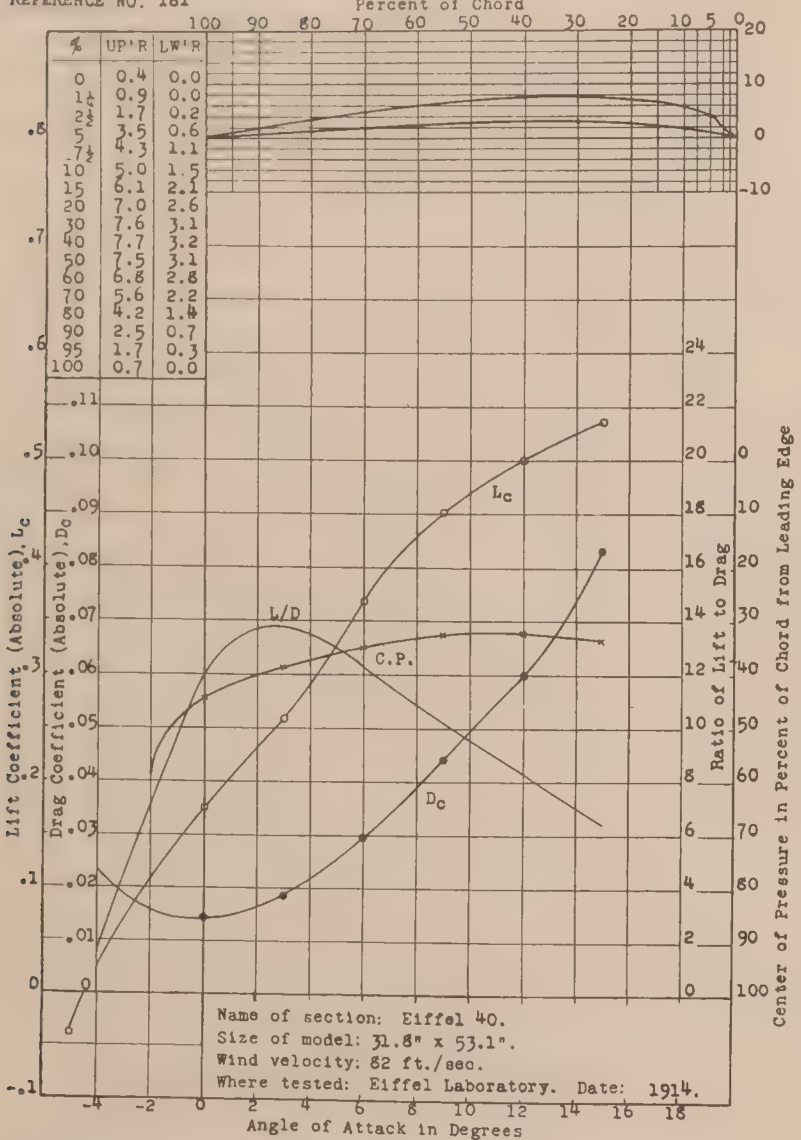


%	UP'R	LW'R
0	0.5	0.5
1½	1.7	0.0
2½	2.5	0.0
5	3.8	0.2
7½	4.8	0.6
10	5.6	1.2
15	6.9	2.4
20	7.7	3.2
30	8.7	4.3
40	8.7	4.5
50	8.3	4.1
60	7.6	3.4
70	6.1	2.6
80	4.7	1.8
90	2.9	0.9
95	1.9	0.5
100	1.3	0.0



%	UP'R	LW'R
0	0.4	0.0
1½	0.9	0.0
2½	1.8	0.0
5	4.5	0.0
7½	9.6	0.0
10	11.2	0.0
15	12.8	0.0
20	13.3	0.0
30	13.1	0.0
40	12.0	0.0
50	10.3	0.0
60	8.5	0.0
70	6.4	0.0
80	4.4	0.0
90	2.4	0.0
95	1.4	0.0
100	0.0	0.0

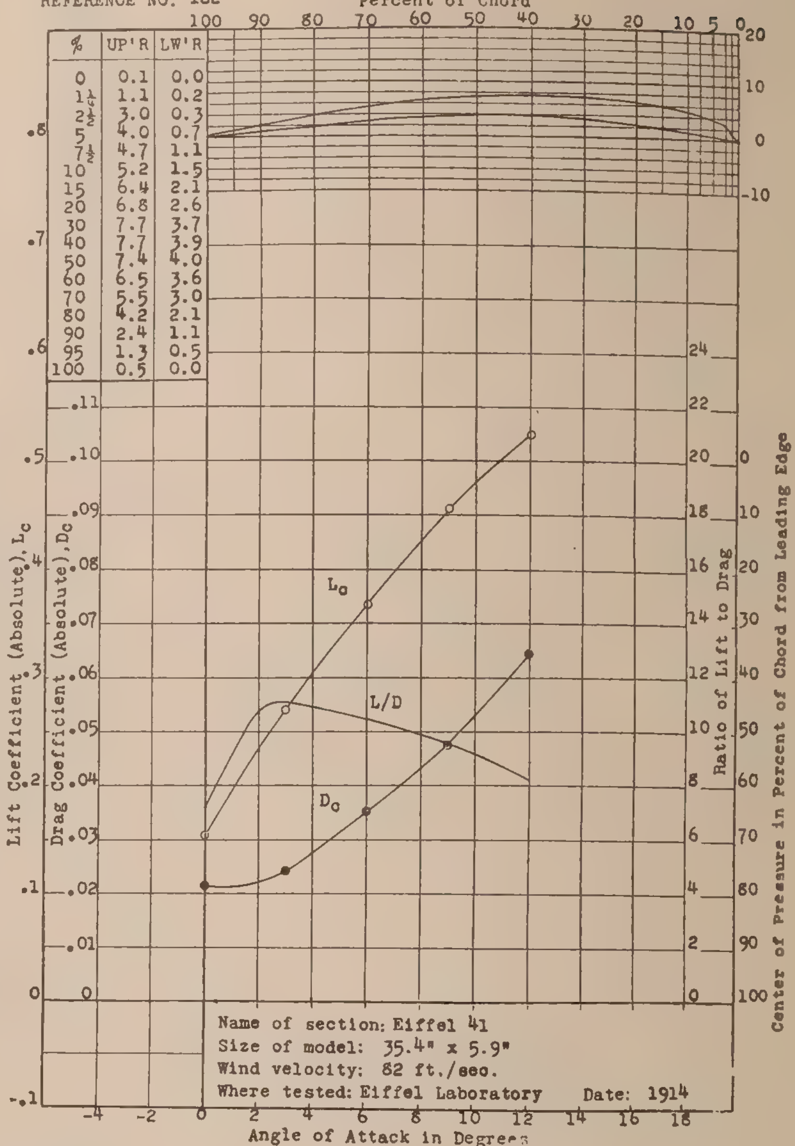
REFERENCE NO. 181



%	UP'R	LW'R
0	0.4	0.0
1¼	0.9	0.0
2½	1.7	0.2
5	3.5	0.6
7½	4.3	1.1
10	5.0	1.5
15	6.1	2.1
20	7.0	2.6
30	7.6	3.1
40	7.7	3.2
50	7.5	3.1
60	6.8	2.8
70	5.6	2.2
80	4.2	1.4
90	2.5	0.7
95	1.7	0.3
100	0.7	0.0

Name of section: Eiffel 40.
Size of model: 31.8" x 53.1".
Wind velocity: 82 ft./sec.
Where tested: Eiffel Laboratory. Date: 1914.

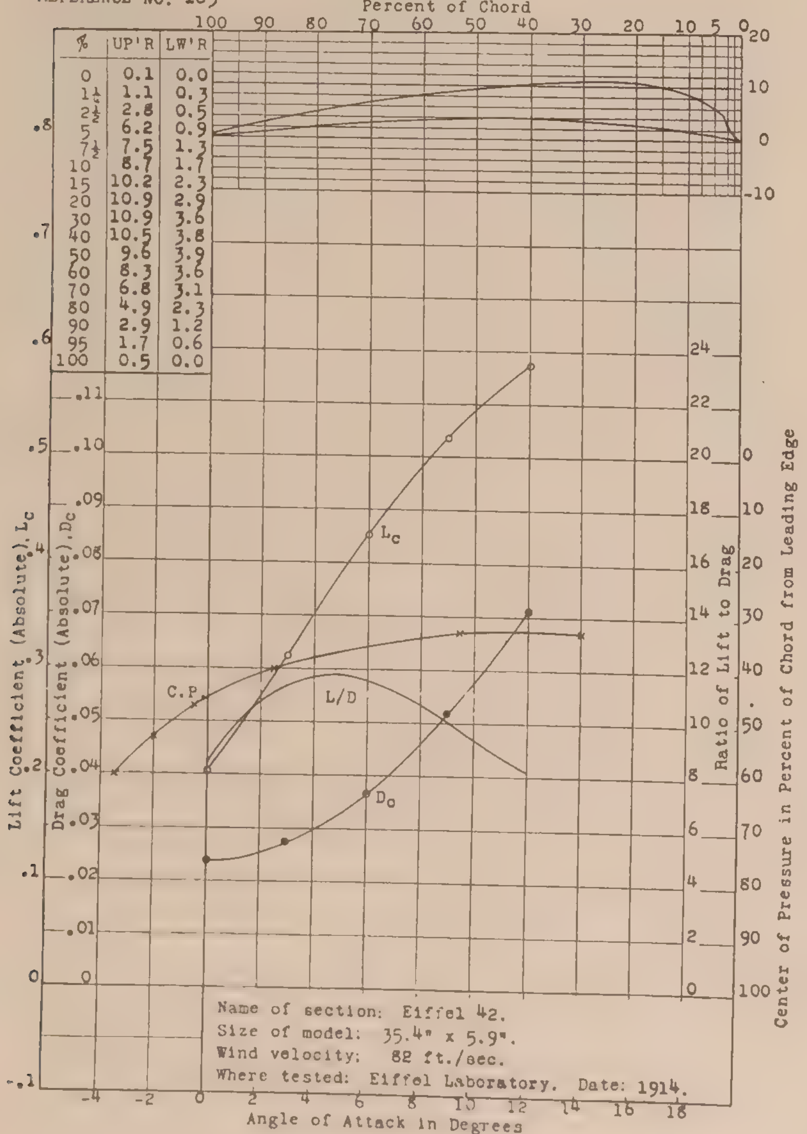
REFERENCE NO. 182



%	UP'R	LW'R
0	0.1	0.0
1¼	1.1	0.2
2½	3.0	0.3
5	4.0	0.7
7½	4.7	1.1
10	5.2	1.5
15	6.4	2.1
20	6.8	2.6
30	7.7	3.7
40	7.7	3.9
50	7.4	4.0
60	6.5	3.6
70	5.5	3.0
80	4.2	2.1
90	2.4	1.1
95	1.3	0.5
100	0.5	0.0

Name of section: Eiffel 41
Size of model: 35.4" x 5.9"
Wind velocity: 82 ft./sec.
Where tested: Eiffel Laboratory Date: 1914

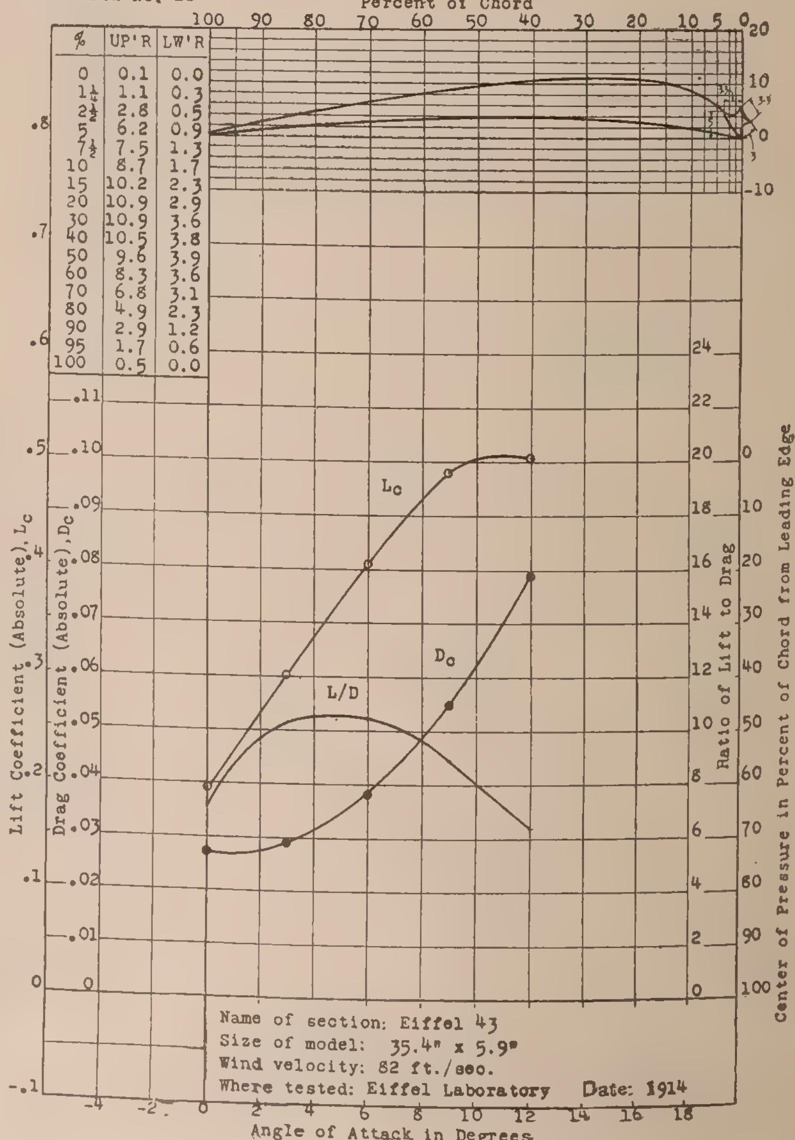
REFERENCE NO. 183



%	UP'R	LW'R
0	0.1	0.0
1¼	1.1	0.3
2½	2.8	0.5
5	6.2	0.9
7½	7.5	1.3
10	8.7	1.7
15	10.2	2.3
20	10.9	2.9
30	10.9	3.6
40	10.5	3.8
50	9.6	3.9
60	8.3	3.6
70	6.8	3.1
80	4.9	2.3
90	2.9	1.2
95	1.7	0.6
100	0.5	0.0

Name of section: Eiffel 42.
Size of model: 35.4" x 5.9".
Wind velocity: 82 ft./sec.
Where tested: Eiffel Laboratory. Date: 1914.

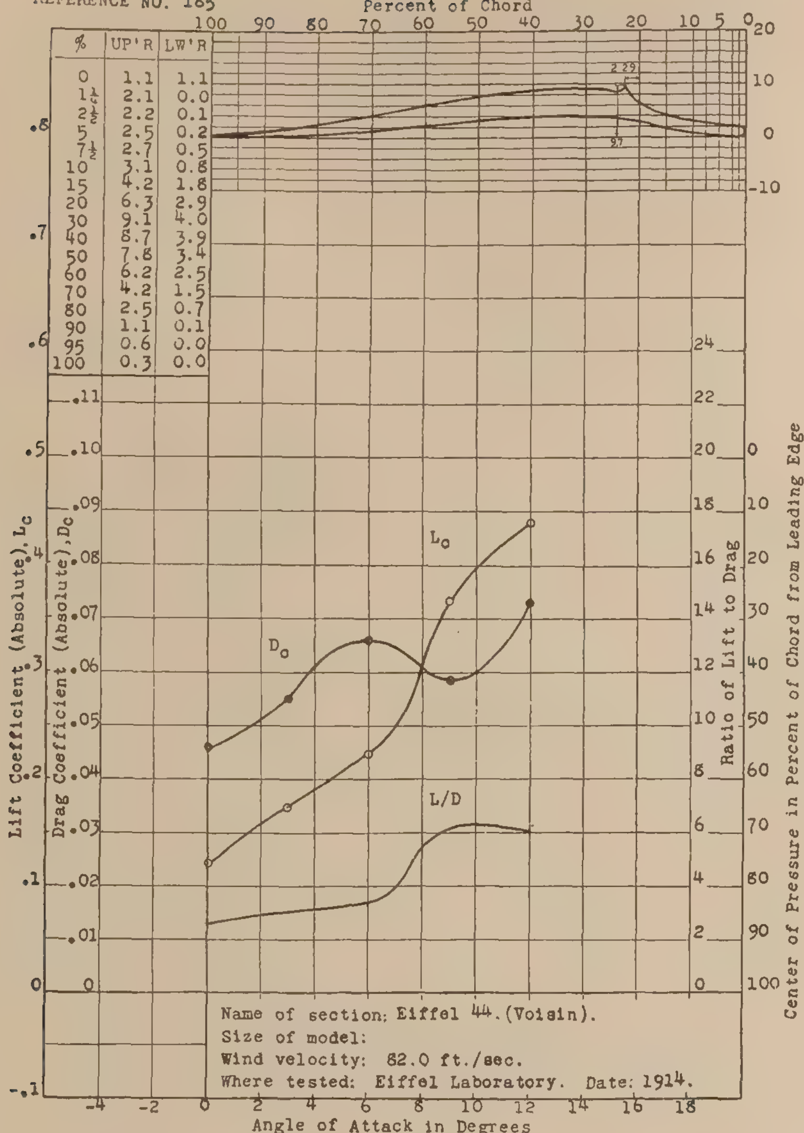
REFERENCE NO. 184



%	UP'R	LW'R
0	0.1	0.0
1¼	1.1	0.3
2½	2.8	0.5
5	6.2	0.9
7½	7.5	1.3
10	8.7	1.7
15	10.2	2.3
20	10.9	2.9
30	10.9	3.6
40	10.5	3.8
50	9.6	3.9
60	8.3	3.6
70	6.8	3.1
80	4.9	2.3
90	2.9	1.2
95	1.7	0.6
100	0.5	0.0

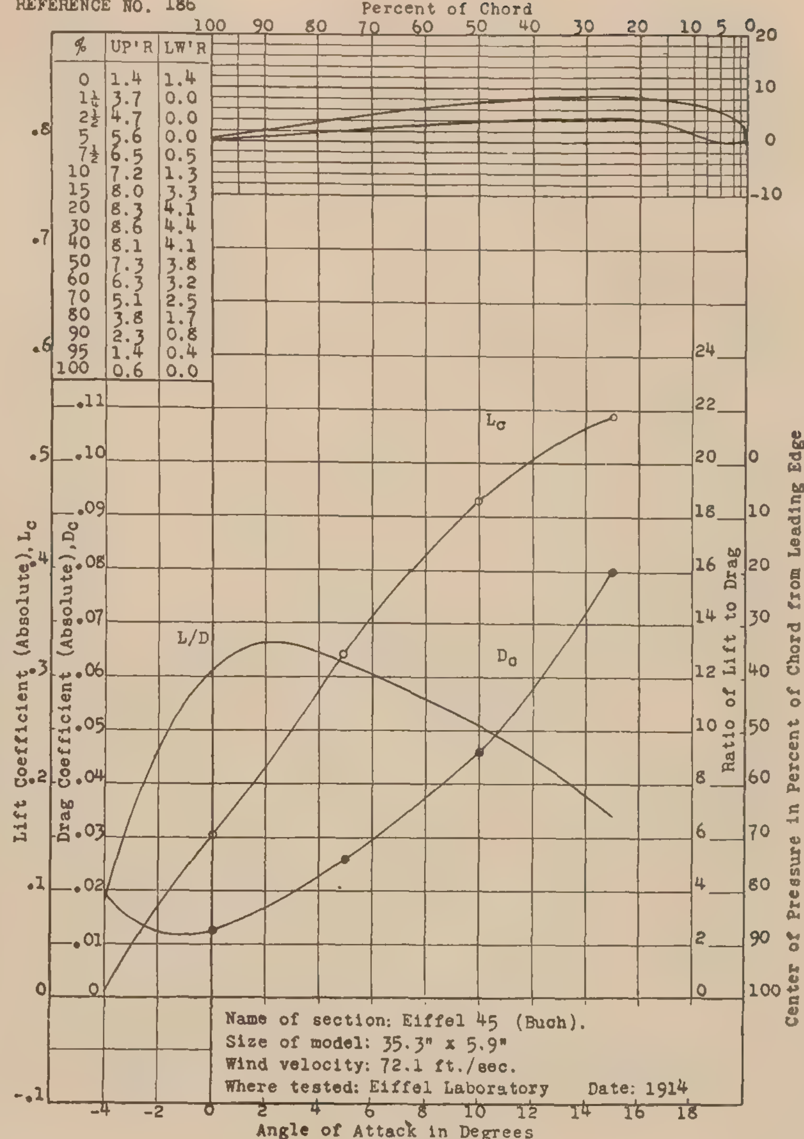
Name of section: Eiffel 43
Size of model: 35.4" x 5.9"
Wind velocity: 82 ft./sec.
Where tested: Eiffel Laboratory Date: 1914

REFERENCE NO. 185



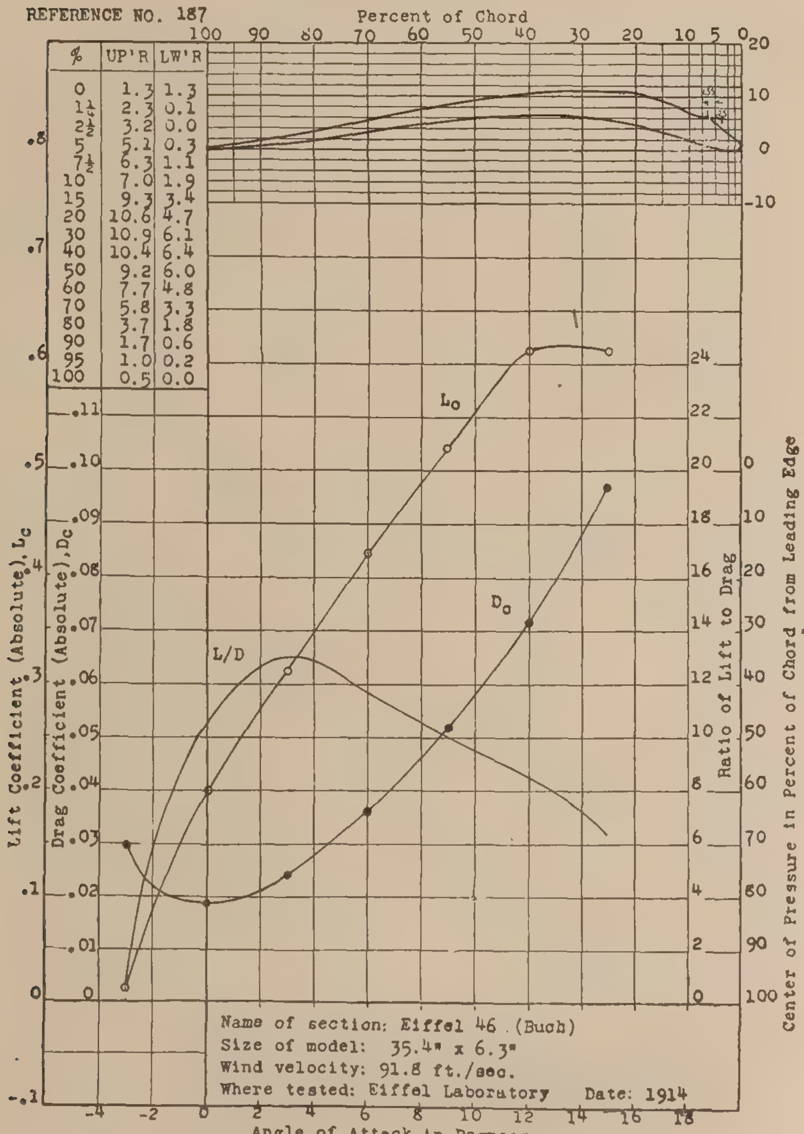
%	UP'R	LW'R
0	1.1	1.1
1½	2.1	0.0
2½	2.2	0.1
5	2.5	0.2
7½	2.7	0.5
10	3.1	0.8
15	4.2	1.8
20	6.3	2.9
30	9.1	4.0
40	8.7	3.9
50	7.8	3.4
60	6.2	2.5
70	4.2	1.5
80	2.5	0.7
90	1.1	0.1
95	0.6	0.0
100	0.3	0.0

REFERENCE NO. 186



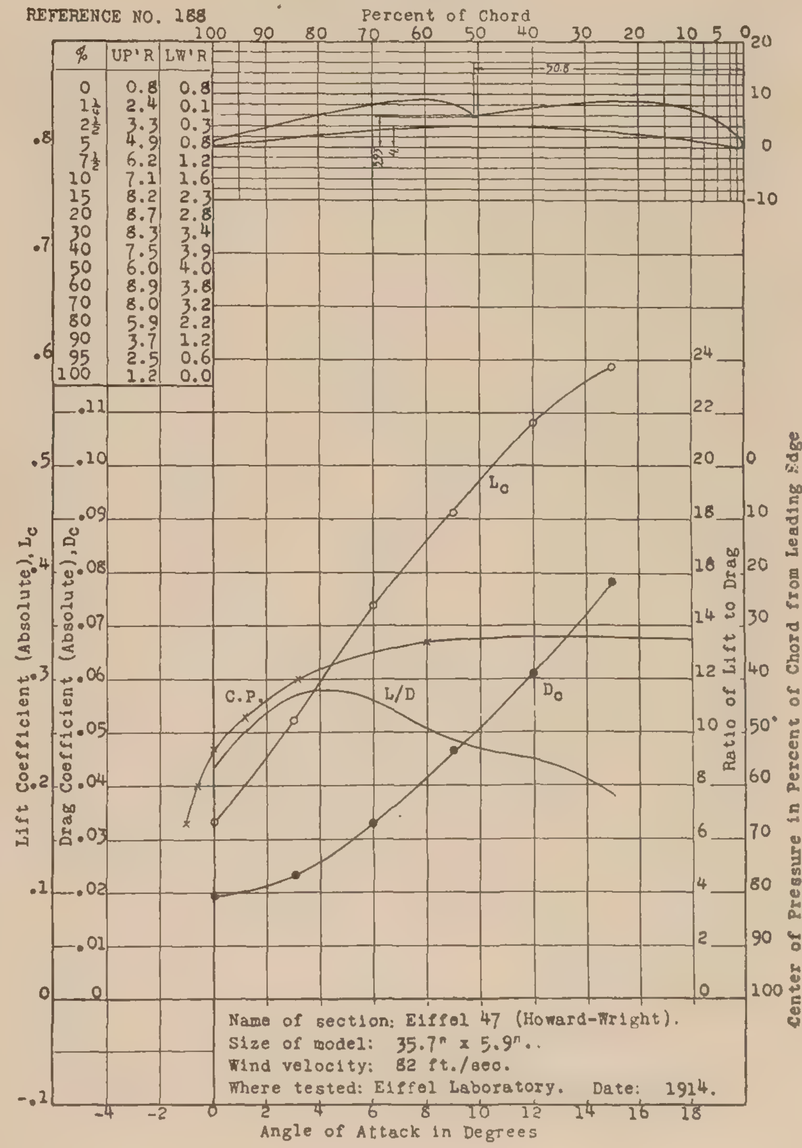
%	UP'R	LW'R
0	1.4	1.4
1½	3.7	0.0
2½	4.7	0.0
5	5.6	0.0
7½	6.5	0.5
10	7.2	1.3
15	8.0	3.3
20	8.3	4.1
30	8.6	4.4
40	8.1	4.1
50	7.3	3.8
60	6.3	3.2
70	5.1	2.5
80	3.8	1.7
90	2.3	0.8
95	1.4	0.4
100	0.6	0.0

REFERENCE NO. 187



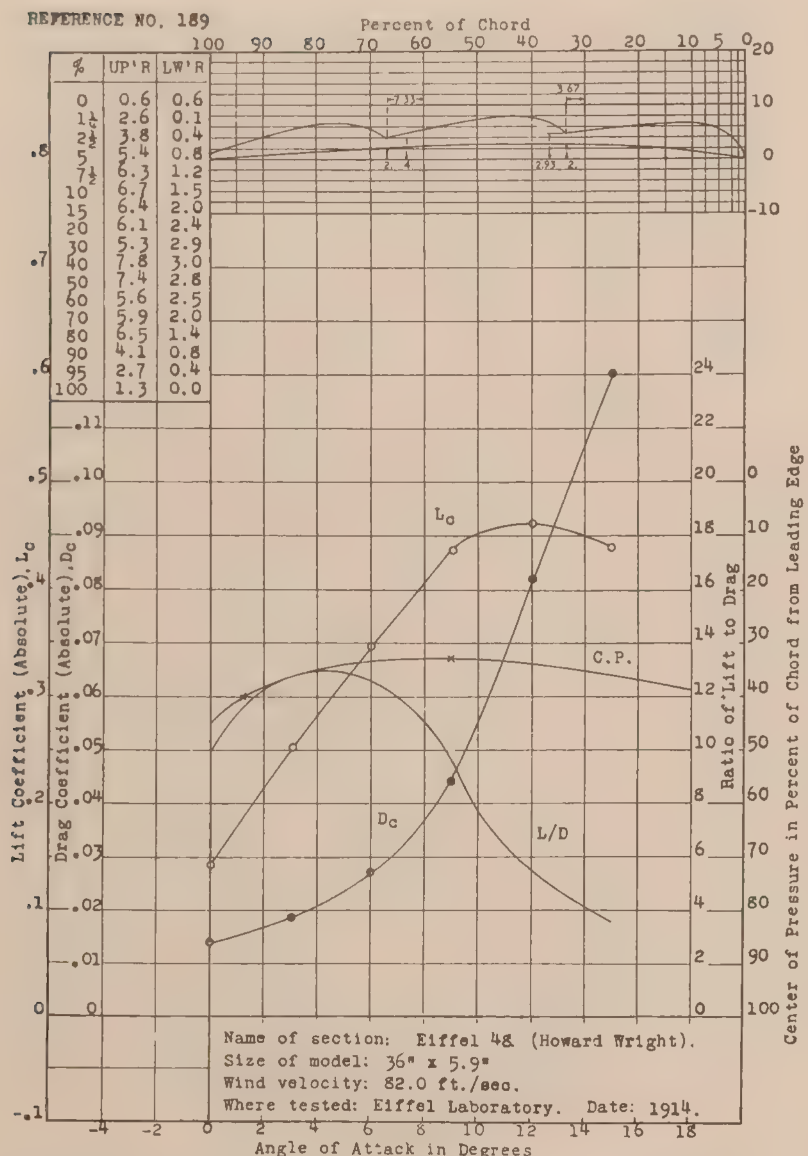
%	UP'R	LW'R
0	1.3	1.3
1½	2.3	0.1
2½	3.2	0.0
5	5.1	0.3
7½	6.3	1.1
10	7.0	1.9
15	9.3	3.4
20	10.6	4.7
30	10.9	6.1
40	10.4	6.4
50	9.2	6.0
60	7.7	4.8
70	5.8	3.3
80	3.7	1.8
90	1.7	0.6
95	1.0	0.2
100	0.5	0.0

REFERENCE NO. 188



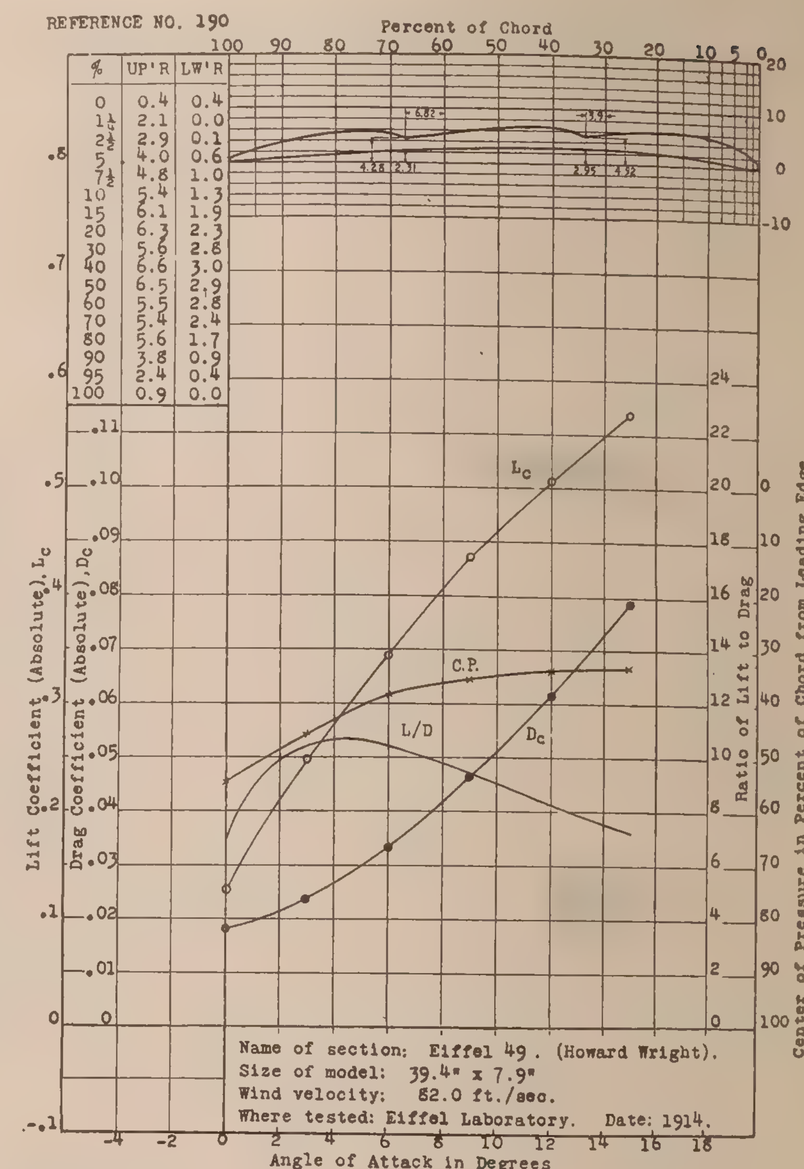
%	UP'R	LW'R
0	0.8	0.8
1½	2.4	0.1
2½	3.3	0.3
5	4.9	0.8
7½	6.2	1.2
10	7.1	1.6
15	8.2	2.3
20	8.7	2.8
30	8.3	3.4
40	7.5	3.9
50	6.0	4.0
60	8.9	3.8
70	8.0	3.2
80	5.9	2.2
90	3.7	1.2
95	2.5	0.6
100	1.2	0.0

REFERENCE NO. 189



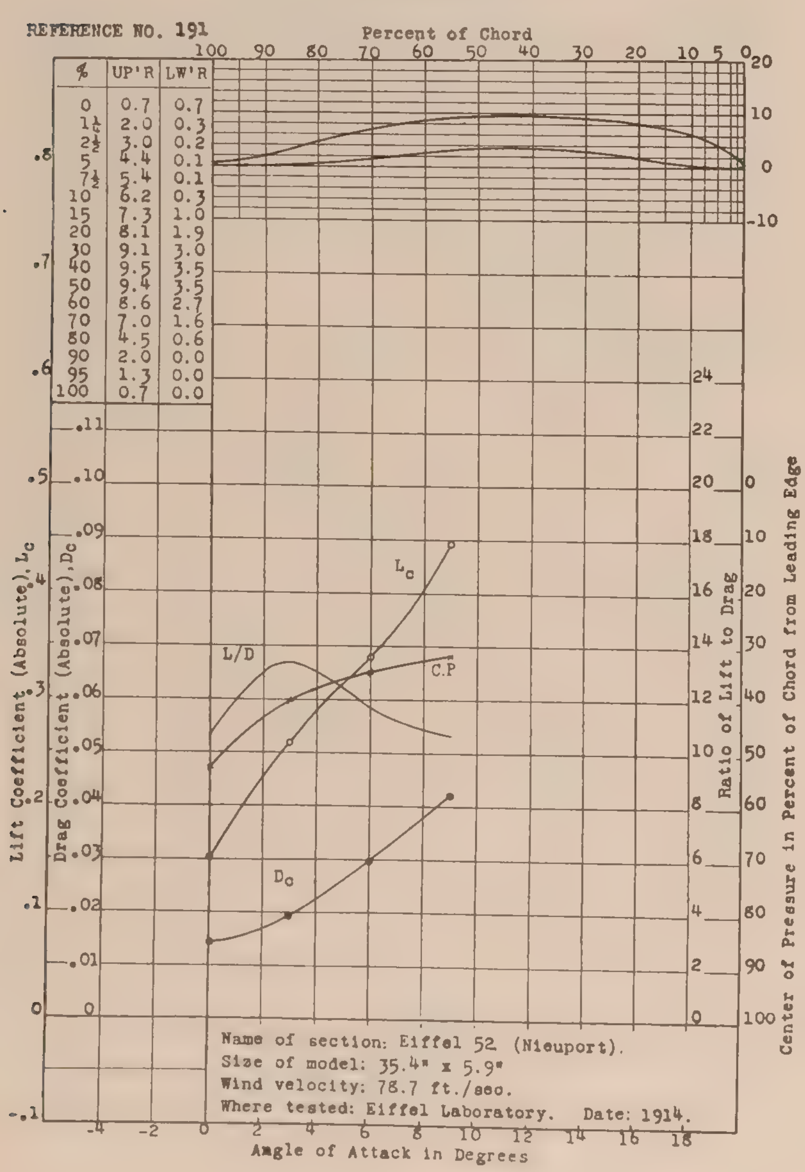
%	UP'R	LW'R
0	0.6	0.6
1¼	2.6	0.1
2½	3.8	0.4
5	5.4	0.8
7½	6.3	1.2
10	6.7	1.5
15	6.4	2.0
20	6.1	2.4
30	5.3	2.9
40	7.8	3.0
50	7.4	2.8
60	5.6	2.5
70	5.9	2.0
80	6.5	1.4
90	4.1	0.8
95	2.7	0.4
100	1.3	0.0

REFERENCE NO. 190



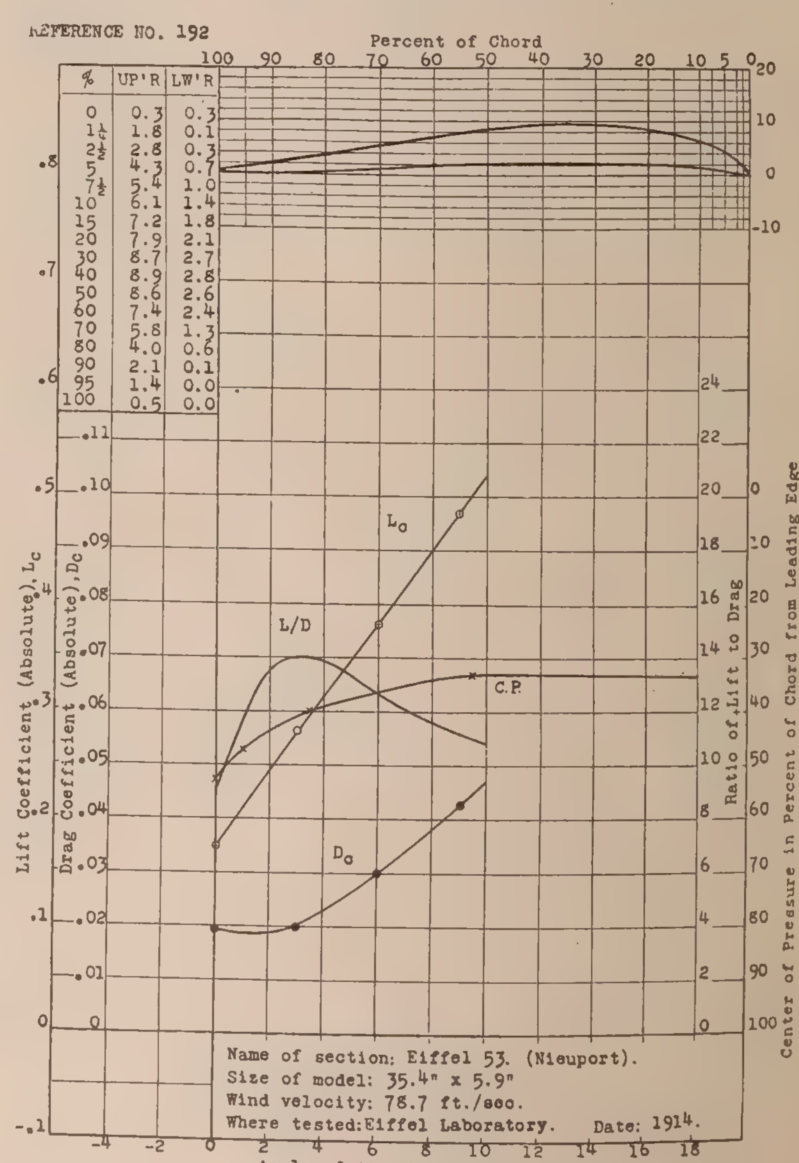
%	UP'R	LW'R
0	0.4	0.4
1¼	2.1	0.0
2½	2.9	0.1
5	4.0	0.6
7½	4.8	1.0
10	5.4	1.3
15	6.1	1.9
20	6.3	2.3
30	5.6	2.8
40	6.6	3.0
50	6.5	2.9
60	5.5	2.8
70	5.4	2.4
80	5.6	1.7
90	3.8	0.9
95	2.4	0.4
100	0.9	0.0

REFERENCE NO. 191



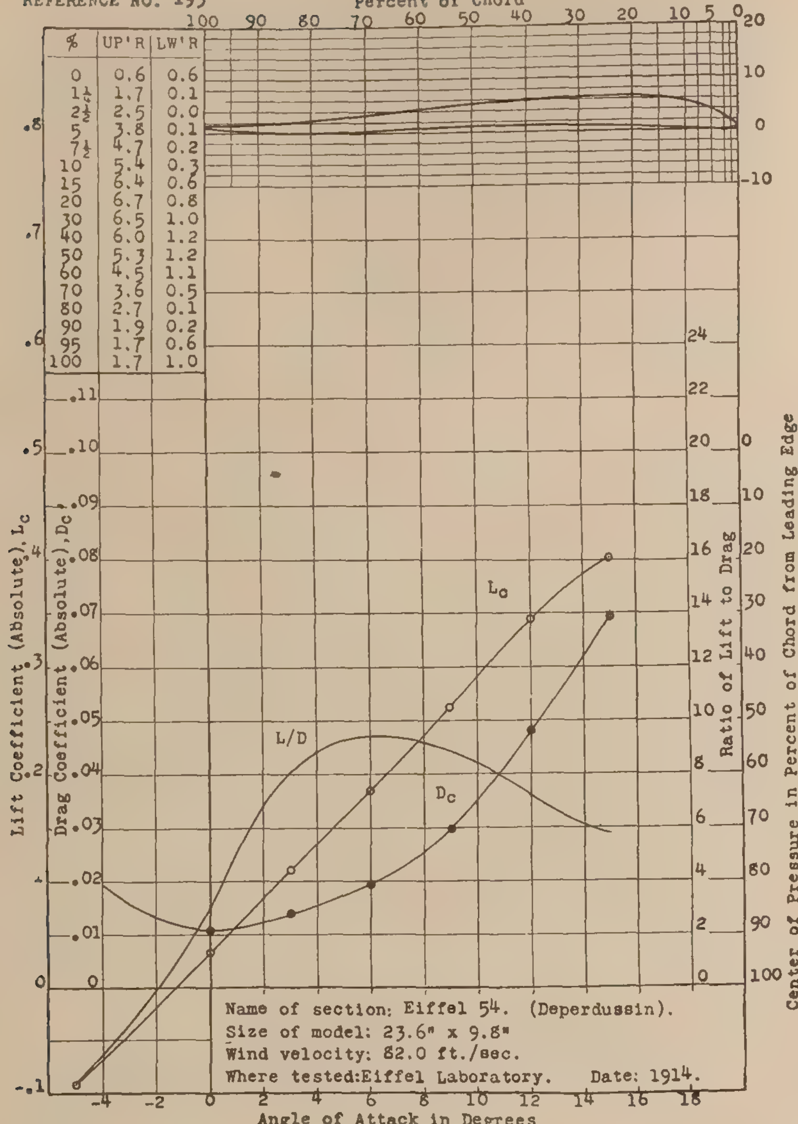
%	UP'R	LW'R
0	0.7	0.7
1¼	2.0	0.3
2½	3.0	0.2
5	4.4	0.1
7½	5.4	0.1
10	6.2	0.3
15	7.3	1.0
20	8.1	1.9
30	9.1	3.0
40	9.5	3.5
50	9.4	3.5
60	8.6	2.7
70	7.0	1.6
80	4.5	0.6
90	2.0	0.0
95	1.3	0.0
100	0.7	0.0

REFERENCE NO. 192



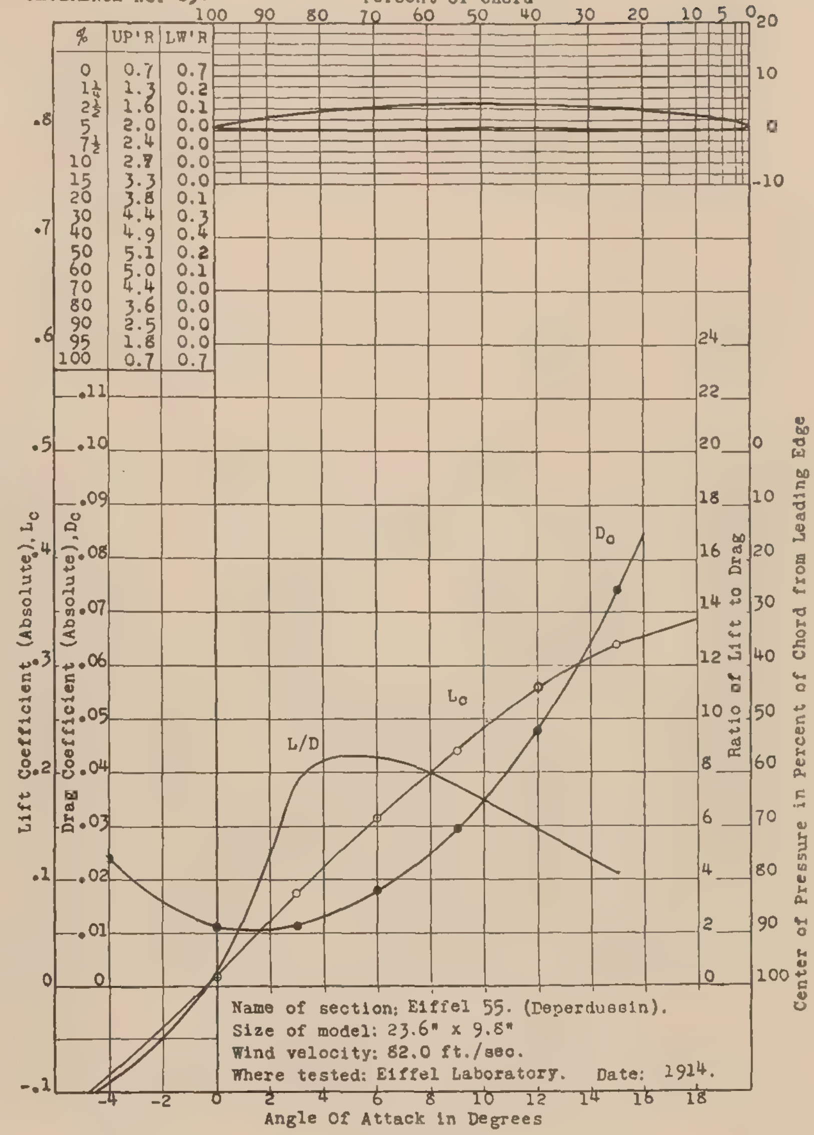
%	UP'R	LW'R
0	0.3	0.3
1¼	1.8	0.1
2½	2.8	0.3
5	4.3	0.7
7½	5.4	1.0
10	6.1	1.4
15	7.2	1.8
20	7.9	2.1
30	8.7	2.7
40	8.9	2.8
50	8.6	2.6
60	7.4	2.4
70	5.8	1.3
80	4.0	0.6
90	2.1	0.1
95	1.4	0.0
100	0.5	0.0

REFERENCE NO. 193



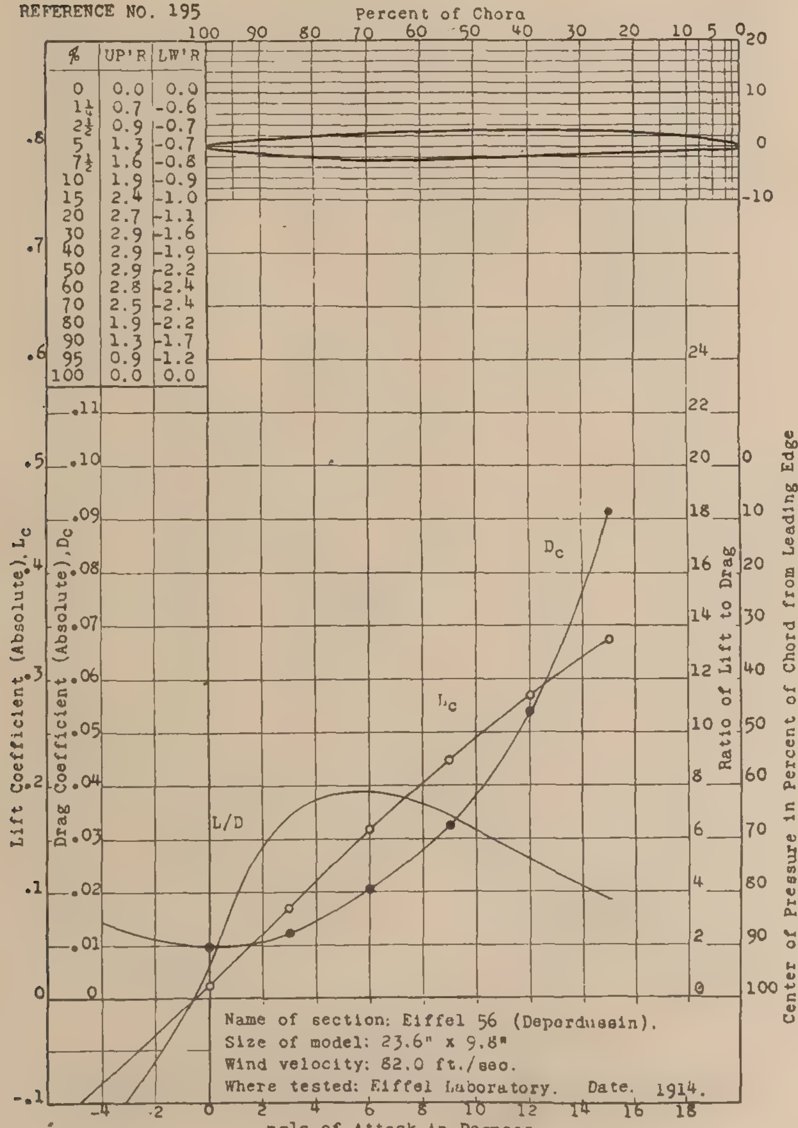
%	UP'R	LW'R
0	0.6	0.6
1¼	1.7	0.1
2½	2.5	0.0
5	3.8	0.1
7½	4.7	0.2
10	5.4	0.3
15	6.4	0.6
20	6.7	0.8
30	6.5	1.0
40	6.0	1.2
50	5.3	1.2
60	4.5	1.1
70	3.6	0.5
80	2.7	0.1
90	1.9	0.2
95	1.7	0.6
100	1.7	1.0

REFERENCE NO. 194



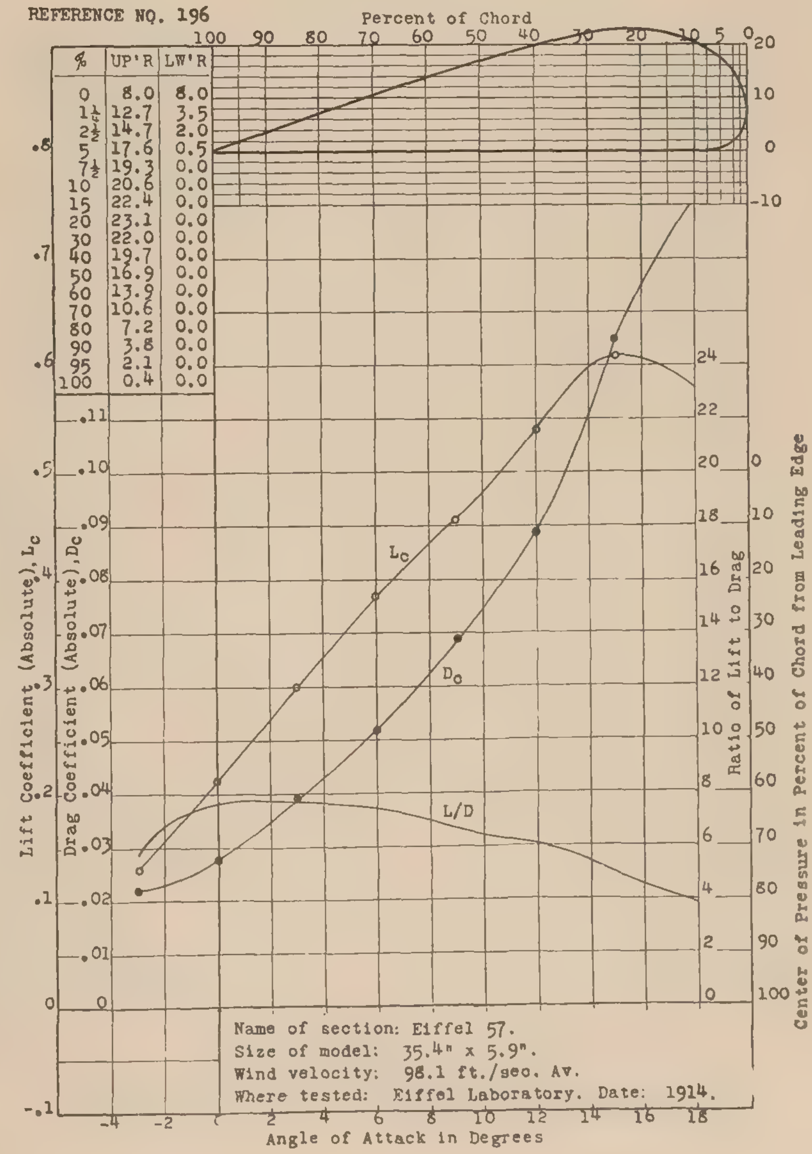
%	UP'R	LW'R
0	0.7	0.7
1¼	1.3	0.2
2½	1.6	0.1
5	2.0	0.0
7½	2.4	0.0
10	2.7	0.0
15	3.3	0.0
20	3.8	0.1
30	4.4	0.3
40	4.9	0.4
50	5.1	0.2
60	5.0	0.1
70	4.4	0.0
80	3.6	0.0
90	2.5	0.0
95	1.8	0.0
100	0.7	0.7

REFERENCE NO. 195



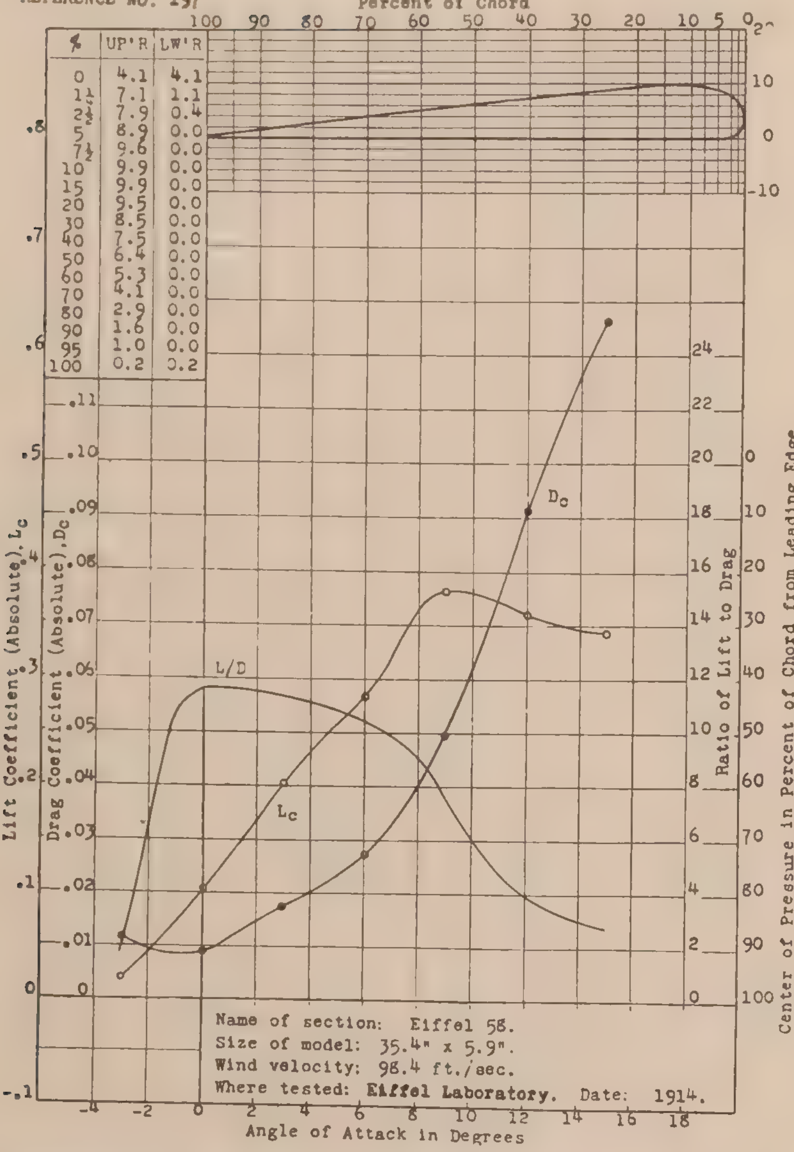
%	UP'R	LW'R
0	0.0	0.0
1¼	0.7	-0.6
2½	0.9	-0.7
5	1.3	-0.7
7½	1.6	-0.8
10	1.9	-0.9
15	2.4	-1.0
20	2.7	-1.1
30	2.9	-1.6
40	2.9	-1.9
50	2.9	-2.2
60	2.8	-2.4
70	2.5	-2.4
80	1.9	-2.2
90	1.3	-1.7
95	0.9	-1.2
100	0.0	0.0

REFERENCE NO. 196



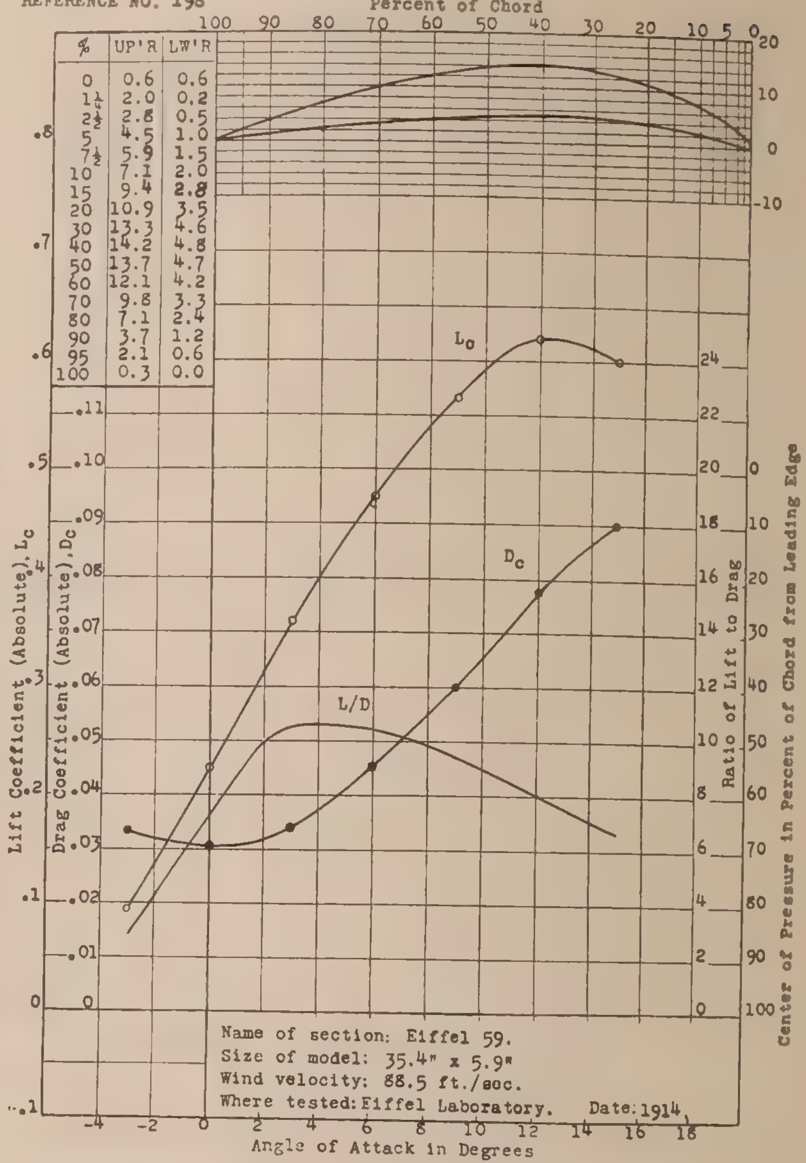
%	UP'R	LW'R
0	8.0	8.0
1¼	12.7	3.5
2½	14.7	2.0
5	17.6	0.5
7½	19.3	0.0
10	20.6	0.0
15	22.4	0.0
20	23.1	0.0
30	22.0	0.0
40	19.7	0.0
50	16.9	0.0
60	13.9	0.0
70	10.6	0.0
80	7.2	0.0
90	3.8	0.0
95	2.1	0.0
100	0.4	0.0

REFERENCE NO. 197



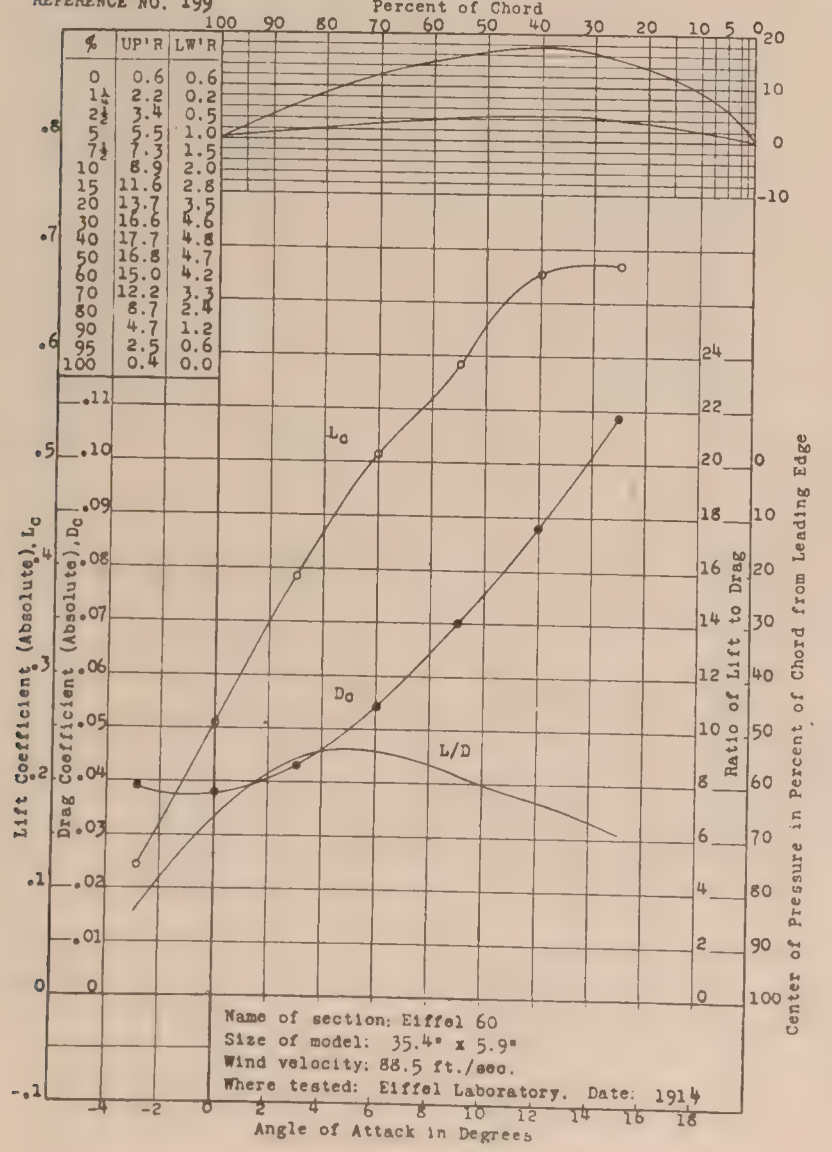
%	UP'R	LW'R
0	4.1	4.1
1¼	7.1	1.1
2½	7.9	0.4
5	8.9	0.0
7½	9.6	0.0
10	9.9	0.0
15	9.9	0.0
20	9.5	0.0
30	8.5	0.0
40	7.5	0.0
50	6.4	0.0
60	5.3	0.0
70	4.1	0.0
80	2.9	0.0
90	1.6	0.0
95	1.0	0.0
100	0.2	0.2

REFERENCE NO. 198



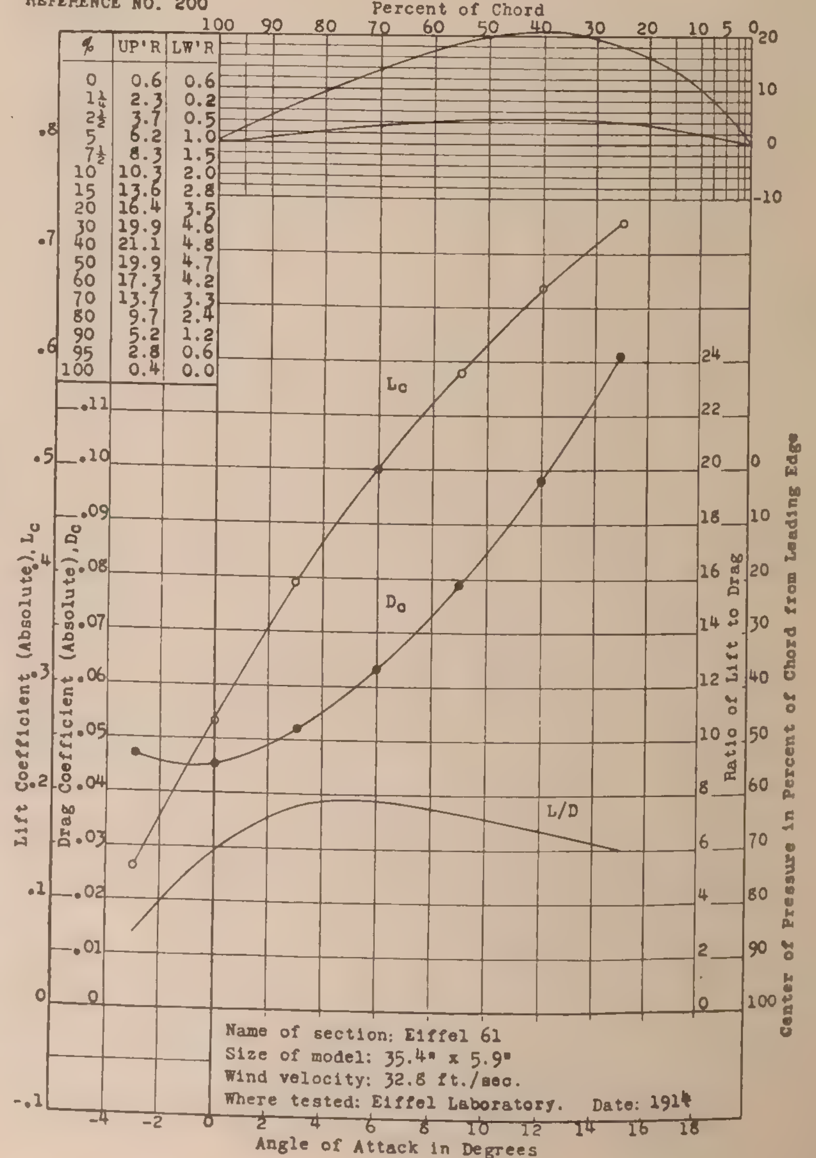
%	UP'R	LW'R
0	0.6	0.6
1¼	2.0	0.2
2½	2.8	0.5
5	4.5	1.0
7½	5.9	1.5
10	7.1	2.0
15	9.4	2.8
20	10.9	3.5
30	13.3	4.6
40	14.2	4.8
50	13.7	4.7
60	12.1	4.2
70	9.8	3.3
80	7.1	2.4
90	3.7	1.2
95	2.1	0.6
100	0.3	0.0

REFERENCE NO. 199



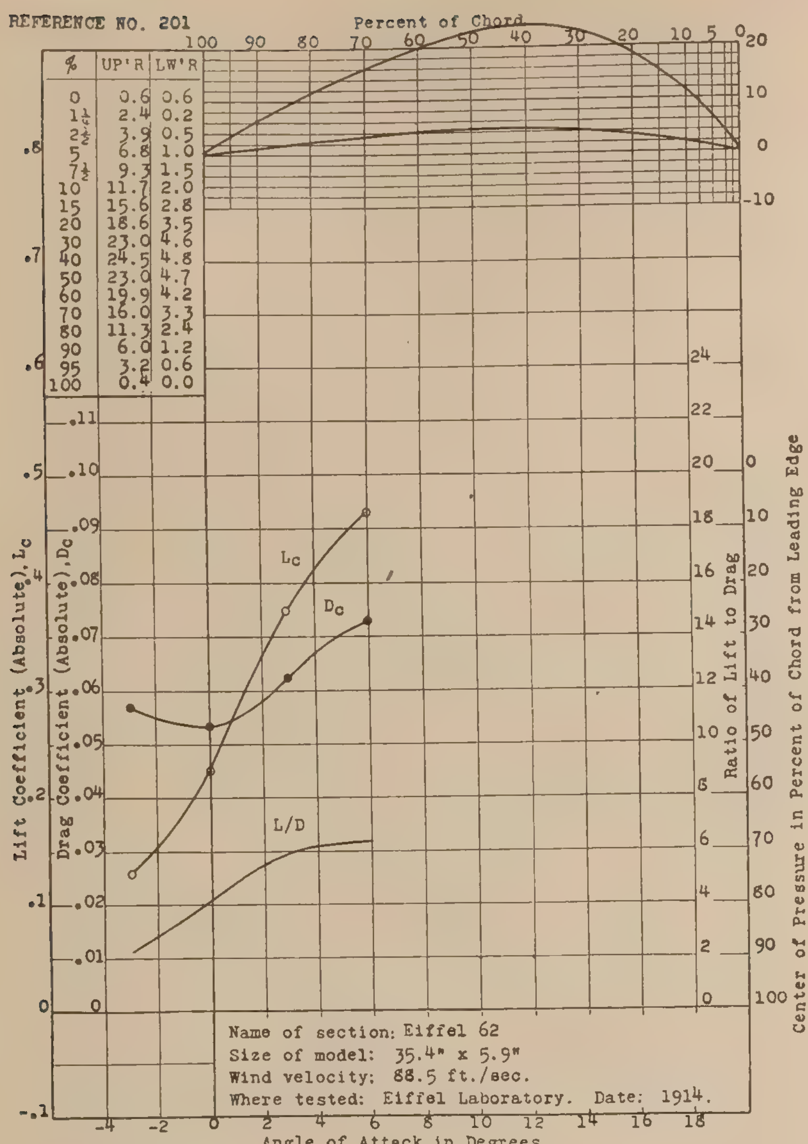
%	UP'R	LW'R
0	0.6	0.6
1¼	2.2	0.2
2½	3.4	0.5
5	5.5	1.0
7½	7.3	1.5
10	8.9	2.0
15	11.6	2.8
20	13.7	3.5
30	16.6	4.6
40	17.7	4.8
50	16.8	4.7
60	15.0	4.2
70	12.2	3.3
80	8.7	2.4
90	4.7	1.2
95	2.5	0.6
100	0.4	0.0

REFERENCE NO. 200



%	UP'R	LW'R
0	0.6	0.6
1¼	2.3	0.2
2½	3.7	0.5
5	6.2	1.0
7½	8.3	1.5
10	10.3	2.0
15	13.6	2.8
20	16.4	3.5
30	19.9	4.6
40	21.1	4.8
50	19.9	4.7
60	17.3	4.2
70	13.7	3.3
80	9.7	2.4
90	5.2	1.2
95	2.8	0.6
100	0.4	0.0

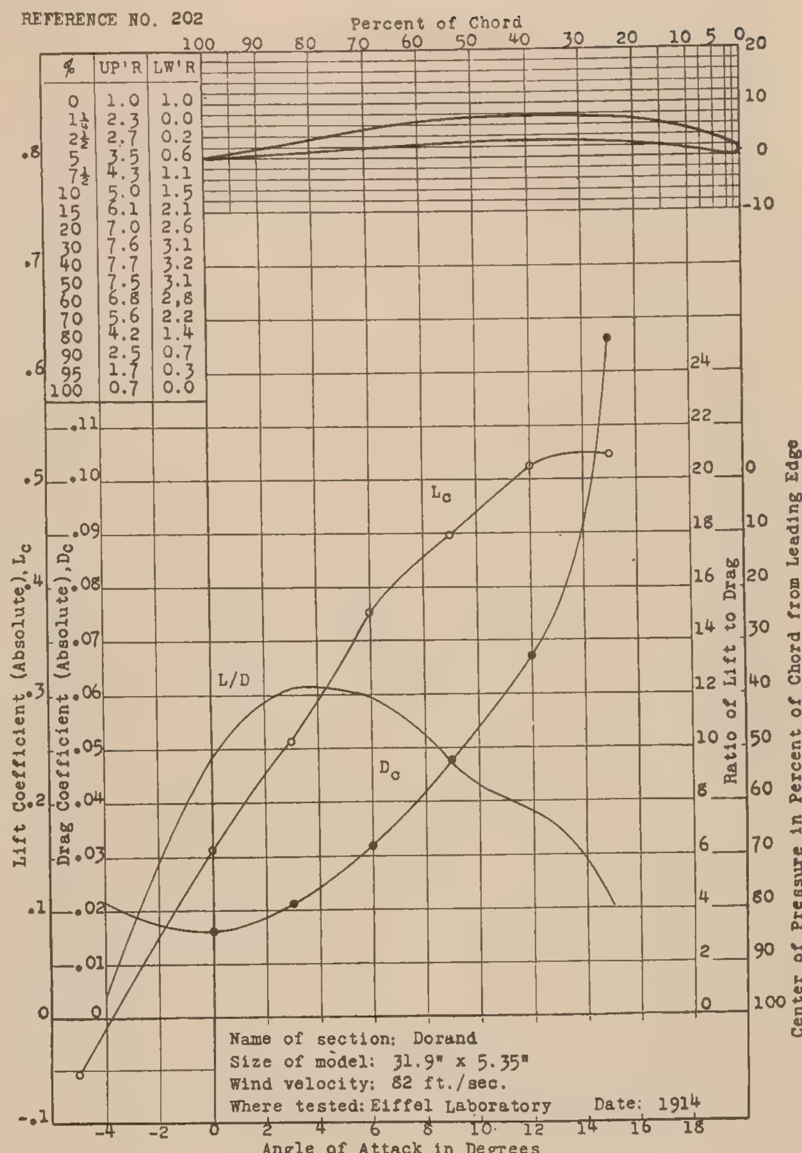
REFERENCE NO. 201



%	UP'R	LW'R
0	0.6	0.6
1½	2.4	0.2
2½	3.9	0.5
5	6.8	1.0
7½	9.3	1.5
10	11.7	2.0
15	15.6	2.8
20	18.6	3.5
30	23.0	4.6
40	24.5	4.8
50	23.0	4.7
60	19.9	4.2
70	16.0	3.3
80	11.3	2.4
90	6.0	1.2
95	3.2	0.6
100	0.4	0.0

Name of section: Eiffel 62
Size of model: 35.4" x 5.9"
Wind velocity: 88.5 ft./sec.
Where tested: Eiffel Laboratory. Date: 1914.

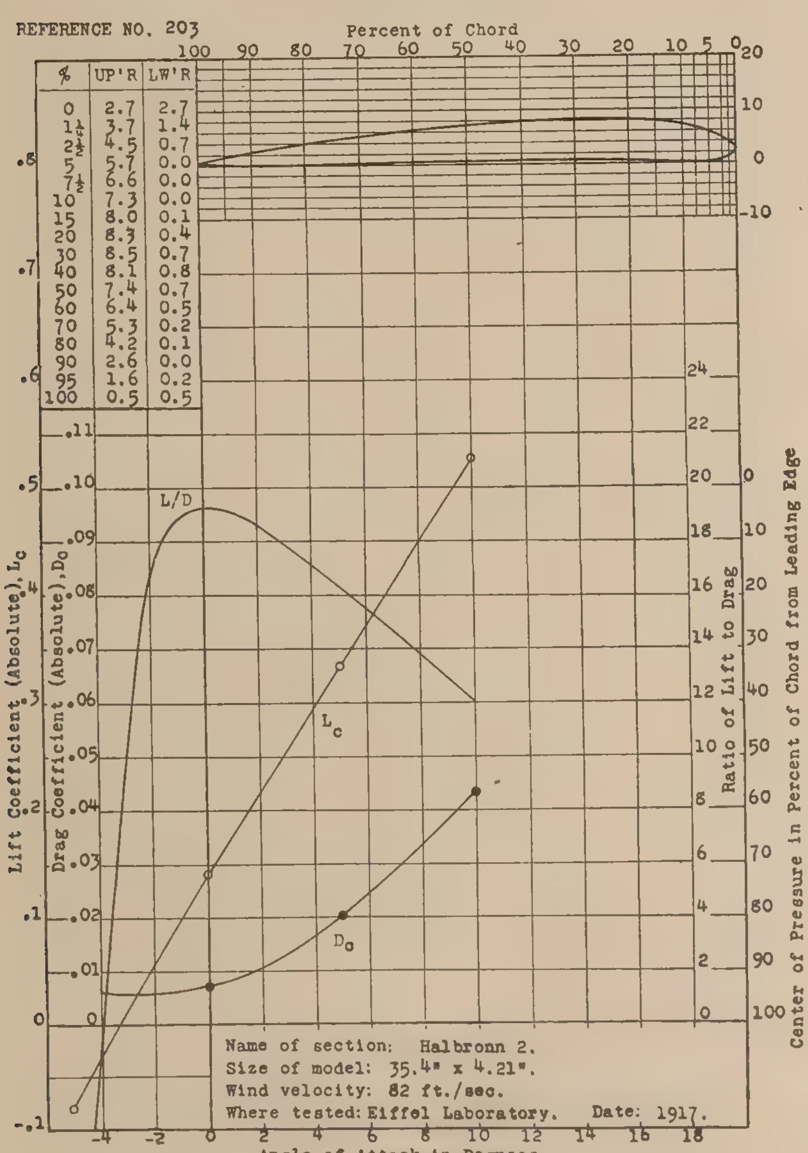
REFERENCE NO. 202



%	UP'R	LW'R
0	1.0	1.0
1½	2.3	0.0
2½	2.7	0.2
5	3.5	0.6
7½	4.3	1.1
10	5.0	1.5
15	6.1	2.1
20	7.0	2.6
30	7.6	3.1
40	7.7	3.2
50	7.5	3.1
60	6.8	2,8
70	5.6	2.2
80	4.2	1.4
90	2.5	0.7
95	1.7	0.3
100	0.7	0.0

Name of section: Dorand
Size of model: 31.9" x 5.35"
Wind velocity: 82 ft./sec.
Where tested: Eiffel Laboratory Date: 1914

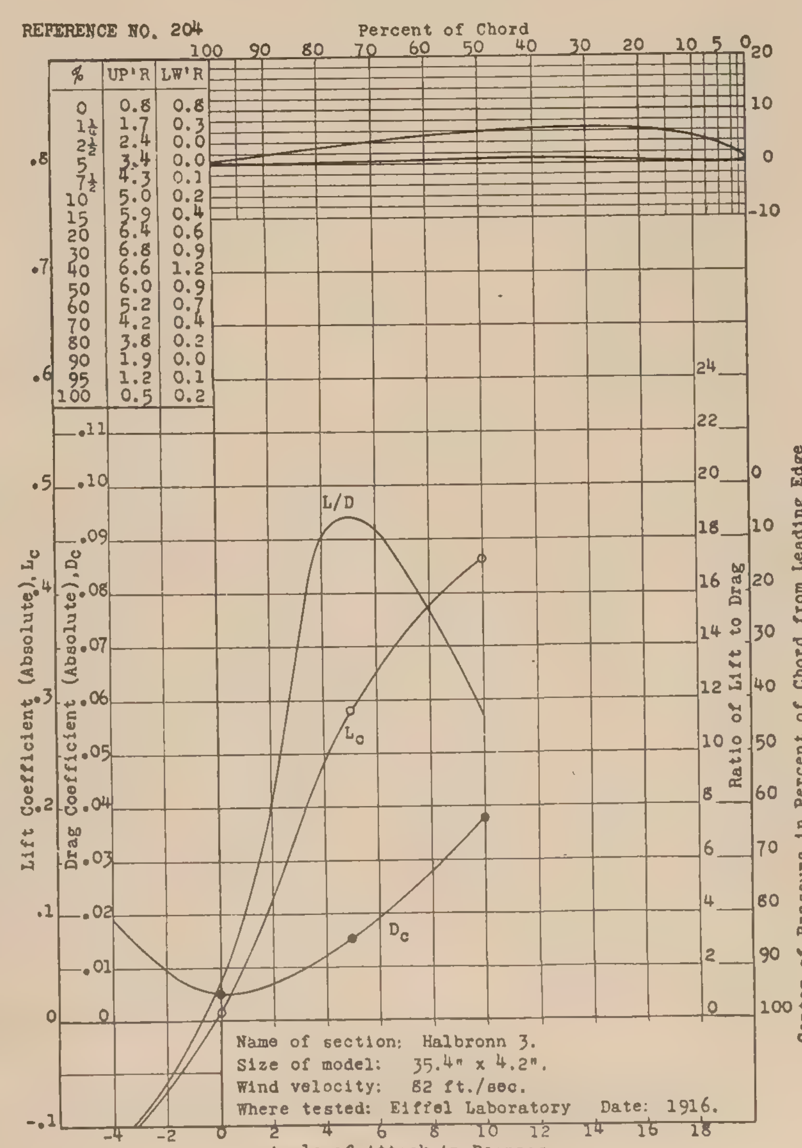
REFERENCE NO. 203



%	UP'R	LW'R
0	2.7	2.7
1½	3.7	1.4
2½	4.5	0.7
5	5.7	0.0
7½	6.6	0.0
10	7.3	0.0
15	8.0	0.1
20	8.3	0.4
30	8.5	0.7
40	8.1	0.8
50	7.4	0.7
60	6.4	0.5
70	5.3	0.2
80	4.2	0.1
90	2.6	0.0
95	1.6	0.2
100	0.5	0.5

Name of section: Halbronn 2.
Size of model: 35.4" x 4.21".
Wind velocity: 82 ft./sec.
Where tested: Eiffel Laboratory. Date: 1917.

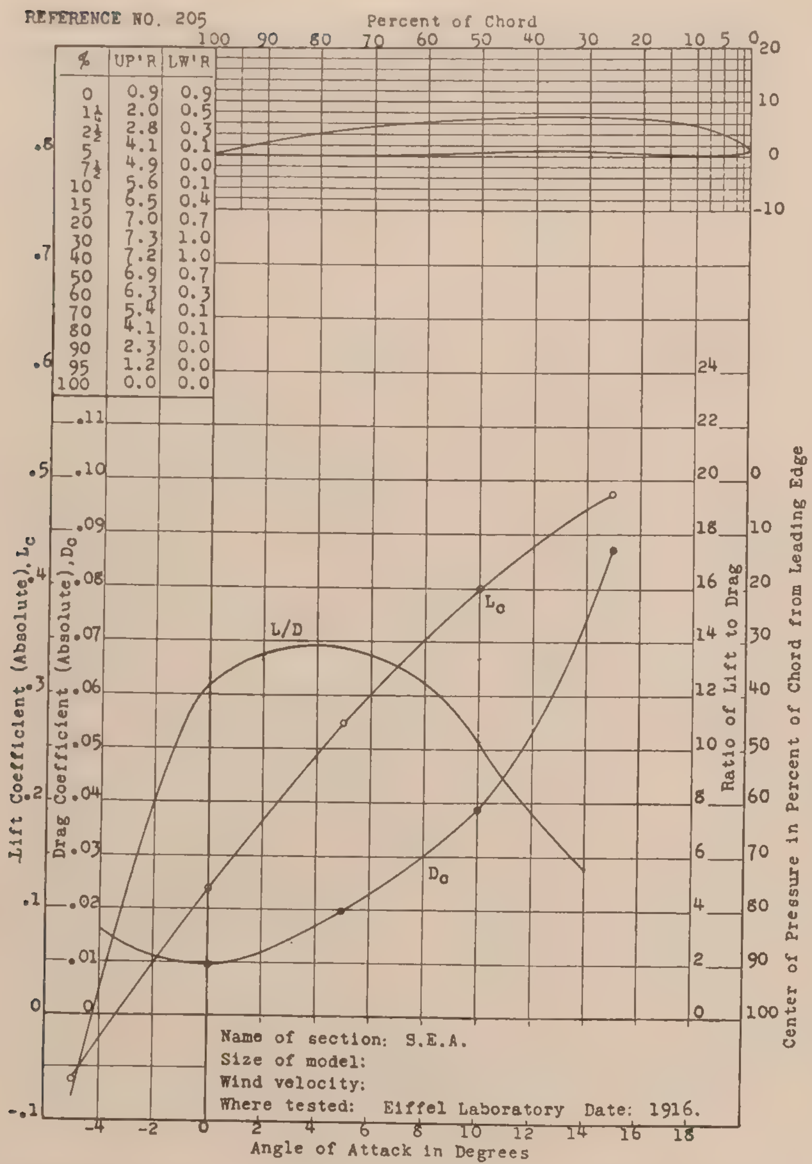
REFERENCE NO. 204



%	UP'R	LW'R
0	0.8	0.8
1½	1.7	0.3
2½	2.4	0.0
5	3.4	0.0
7½	4.3	0.1
10	5.0	0.2
15	5.9	0.4
20	6.4	0.6
30	6.8	0.9
40	6.6	1.2
50	6.0	0.9
60	5.2	0.7
70	4.2	0.4
80	3.8	0.2
90	1.9	0.0
95	1.2	0.1
100	0.5	0.2

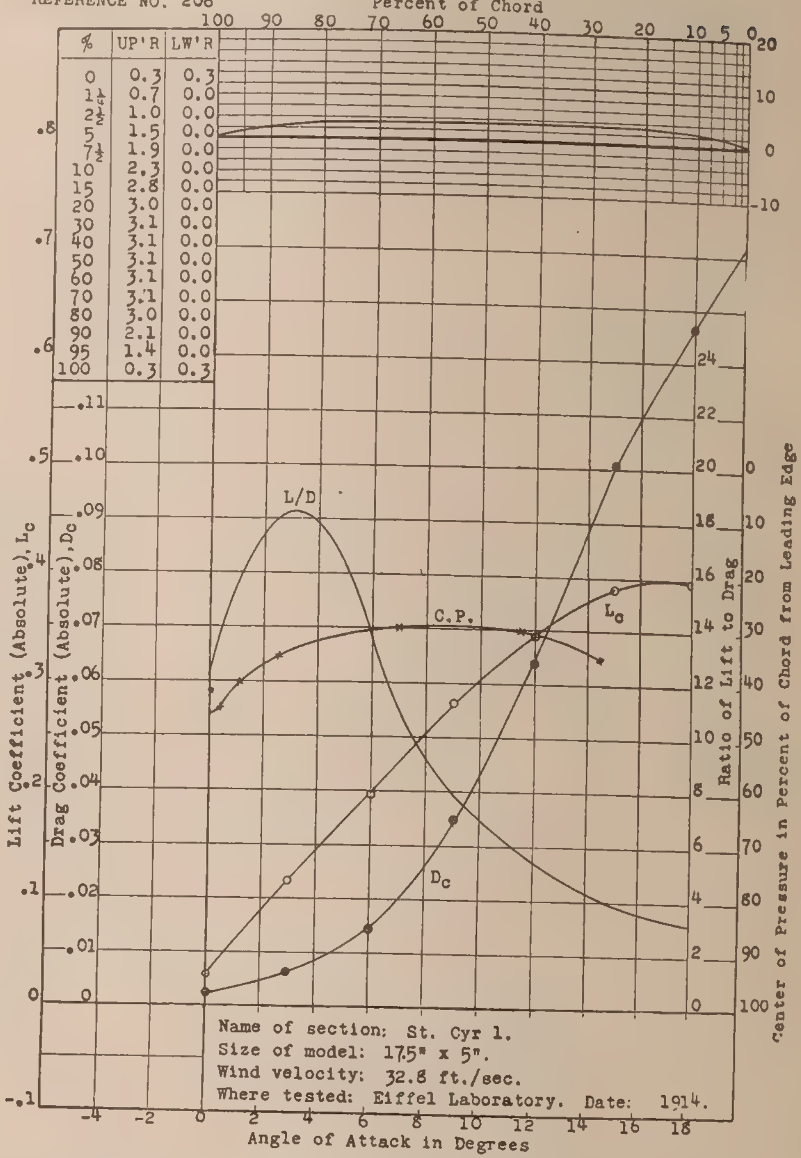
Name of section: Halbronn 3.
Size of model: 35.4" x 4.2".
Wind velocity: 82 ft./sec.
Where tested: Eiffel Laboratory Date: 1916.

REFERENCE NO. 205



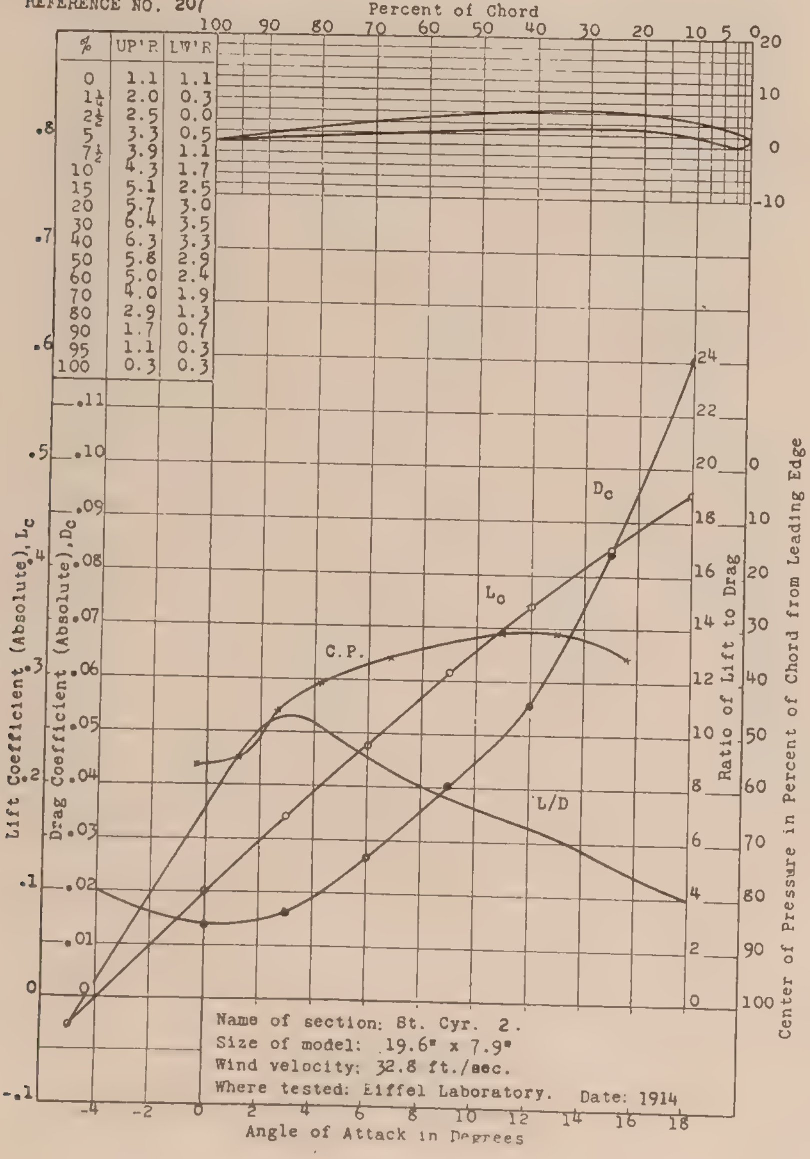
%	UP'R	LW'R
0	0.9	0.9
1¼	2.0	0.5
2½	2.8	0.3
5	4.1	0.1
7½	4.9	0.0
10	5.6	0.1
15	6.5	0.4
20	7.0	0.7
30	7.3	1.0
40	7.2	1.0
50	6.9	0.7
60	6.3	0.3
70	5.4	0.1
80	4.1	0.1
90	2.3	0.0
95	1.2	0.0
100	0.0	0.0

REFERENCE NO. 206



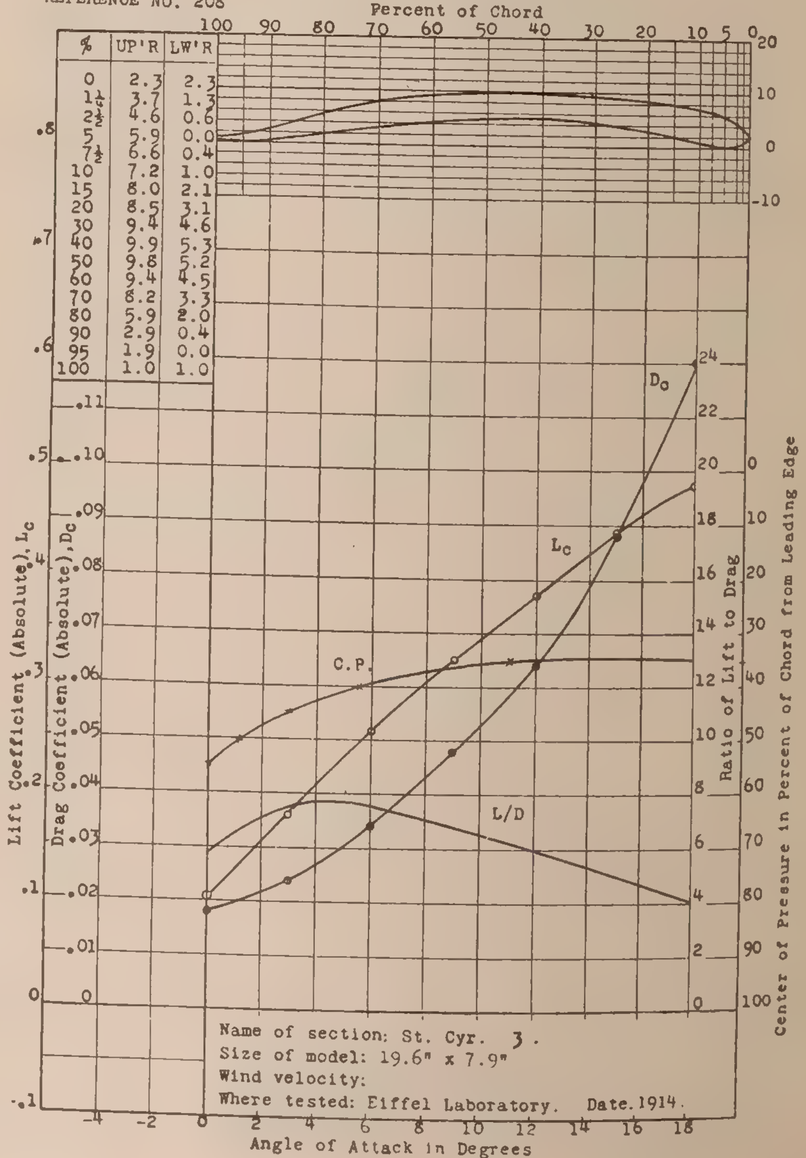
%	UP'R	LW'R
0	0.3	0.3
1¼	0.7	0.0
2½	1.0	0.0
5	1.5	0.0
7½	1.9	0.0
10	2.3	0.0
15	2.8	0.0
20	3.0	0.0
30	3.1	0.0
40	3.1	0.0
50	3.1	0.0
60	3.1	0.0
70	3.1	0.0
80	3.0	0.0
90	2.1	0.0
95	1.4	0.0
100	0.3	0.3

REFERENCE NO. 207



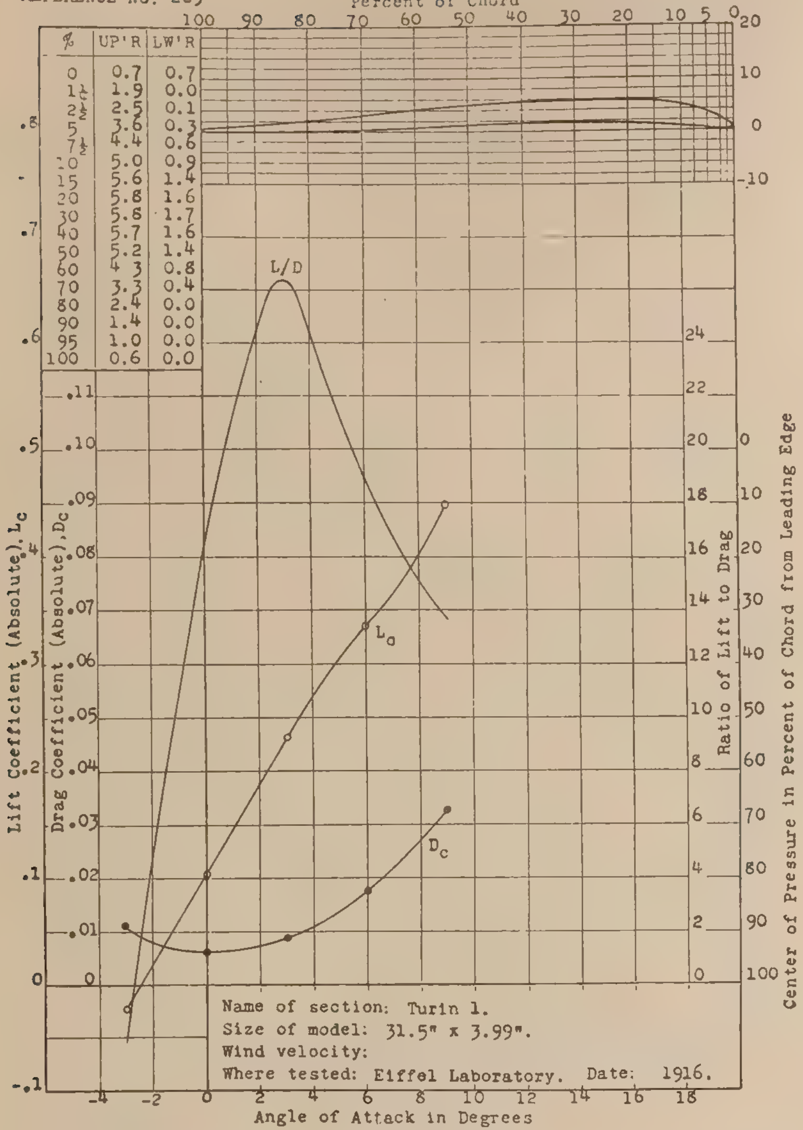
%	UP'R	LW'R
0	1.1	1.1
1¼	2.0	0.3
2½	2.5	0.0
5	3.3	0.5
7½	3.9	1.1
10	4.3	1.7
15	5.1	2.5
20	5.7	3.0
30	6.4	3.5
40	6.3	3.3
50	5.8	2.9
60	5.0	2.4
70	4.0	1.9
80	2.9	1.3
90	1.7	0.7
95	1.1	0.3
100	0.3	0.3

REFERENCE NO. 208



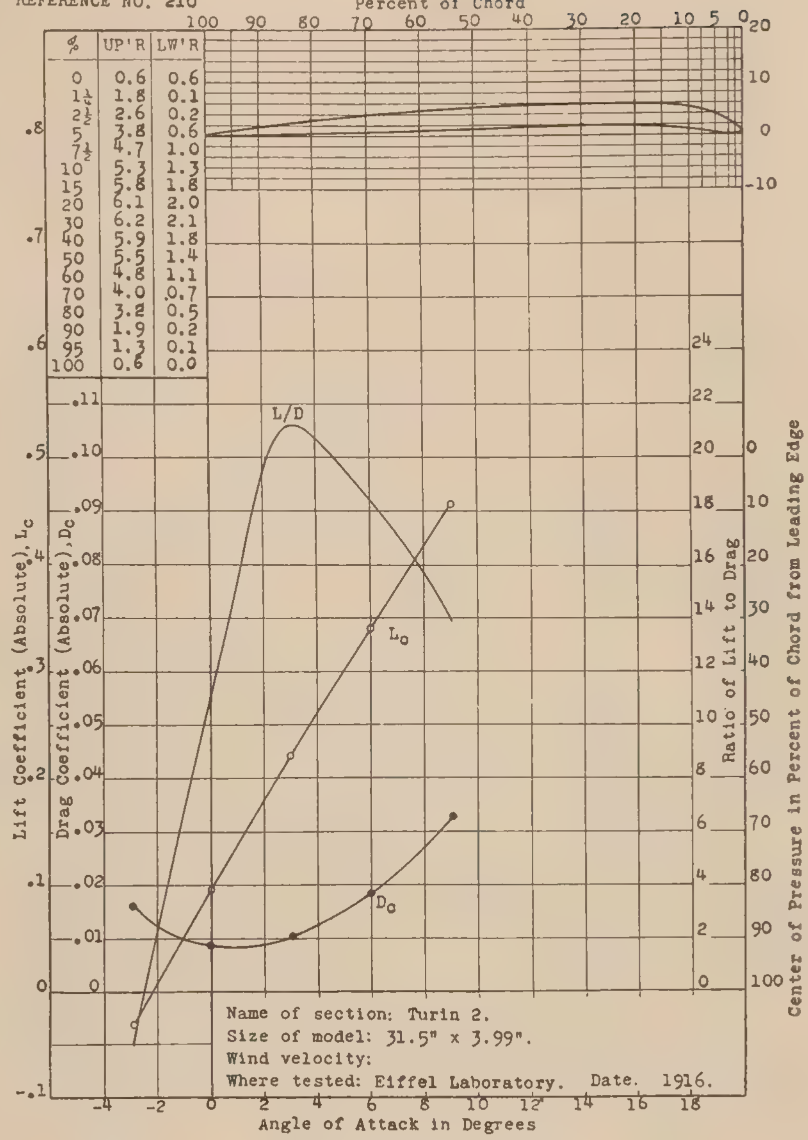
%	UP'R	LW'R
0	2.3	2.3
1¼	3.7	1.3
2½	4.6	0.6
5	5.9	0.0
7½	6.6	0.4
10	7.2	1.0
15	8.0	2.1
20	8.5	3.1
30	9.4	4.6
40	9.9	5.3
50	9.8	5.2
60	9.4	4.5
70	8.2	3.3
80	5.9	2.0
90	2.9	0.4
95	1.9	0.0
100	1.0	1.0

REFERENCE NO. 209



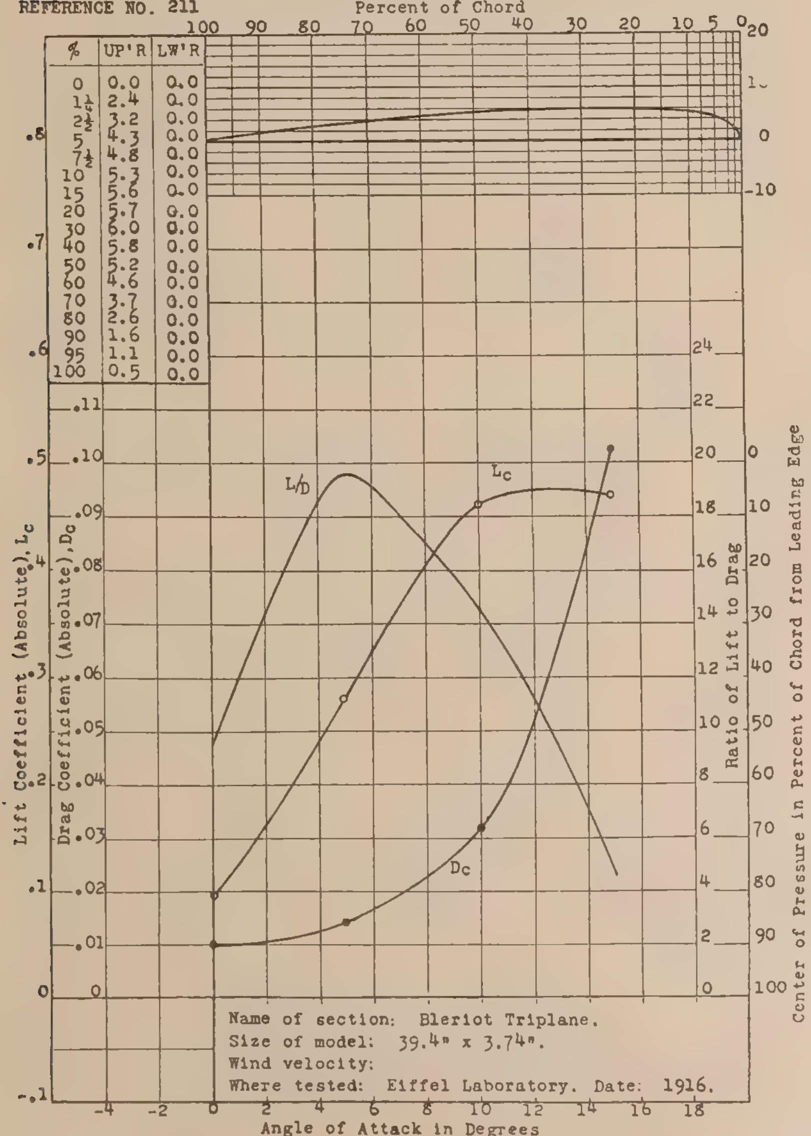
%	UP'R	LW'R
0	0.7	0.7
1¼	1.9	0.0
2½	2.5	0.1
5	3.6	0.3
7½	4.4	0.6
10	5.0	0.9
15	5.6	1.4
20	5.8	1.6
30	5.8	1.7
40	5.7	1.6
50	5.2	1.4
60	4 3	0.8
70	3.3	0.4
80	2.4	0.0
90	1.4	0.0
95	1.0	0.0
100	0.6	0.0

REFERENCE NO. 210



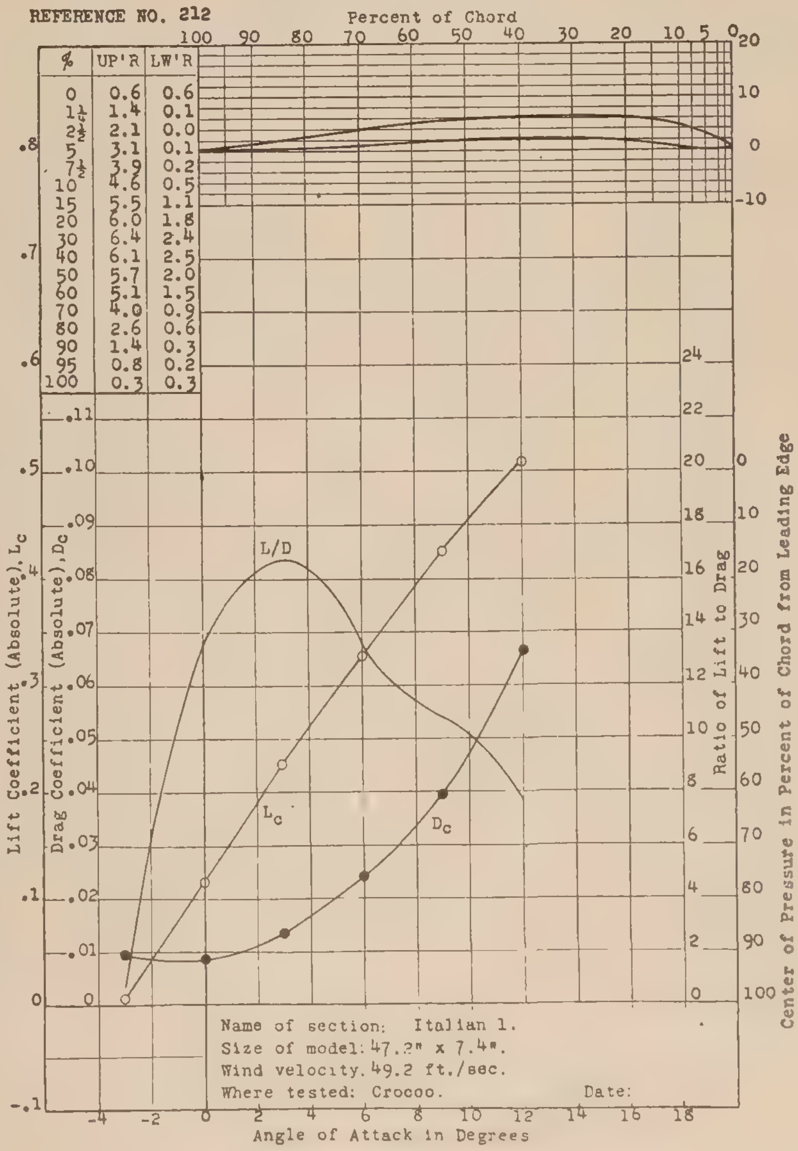
%	UP'R	LW'R
0	0.6	0.6
1¼	1.8	0.1
2½	2.6	0.2
5	3.8	0.6
7½	4.7	1.0
10	5.3	1.3
15	5.8	1.8
20	6.1	2.0
30	6.2	2.1
40	5.9	1.8
50	5.5	1.4
60	4.8	1.1
70	4.0	0.7
80	3.2	0.5
90	1.9	0.2
95	1.3	0.1
100	0.6	0.0

REFERENCE NO. 211

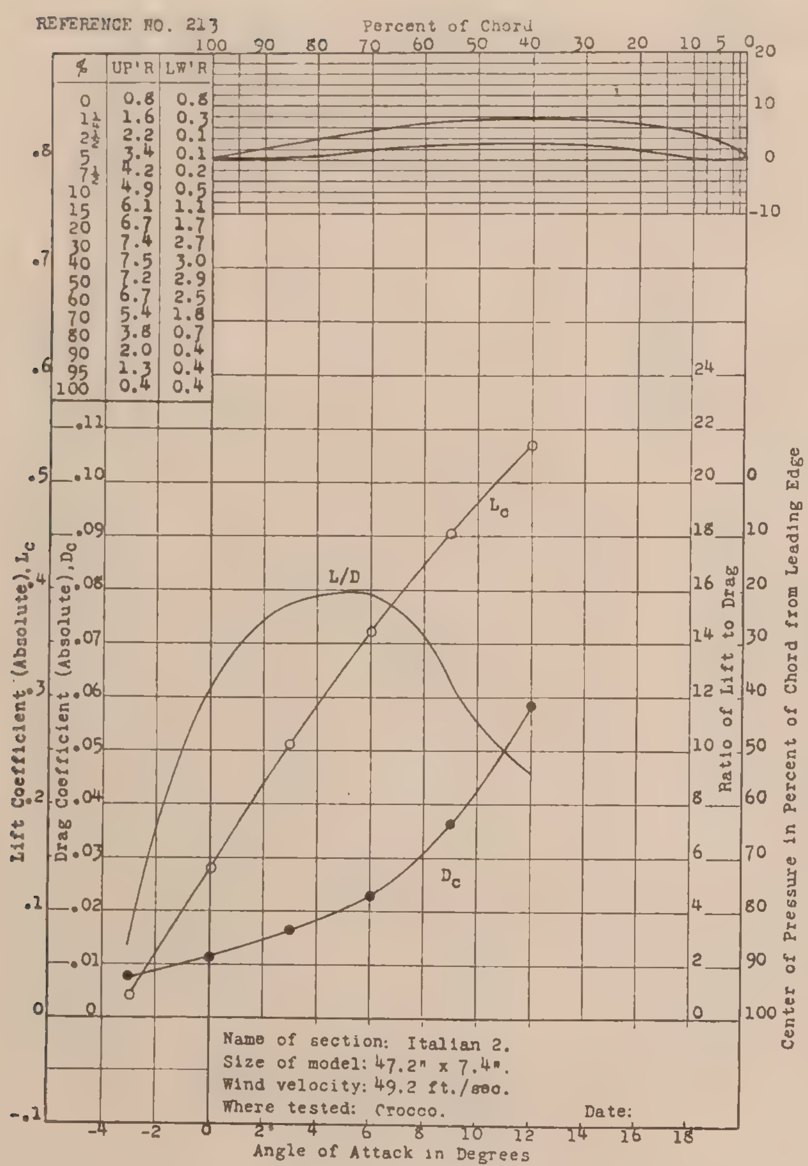


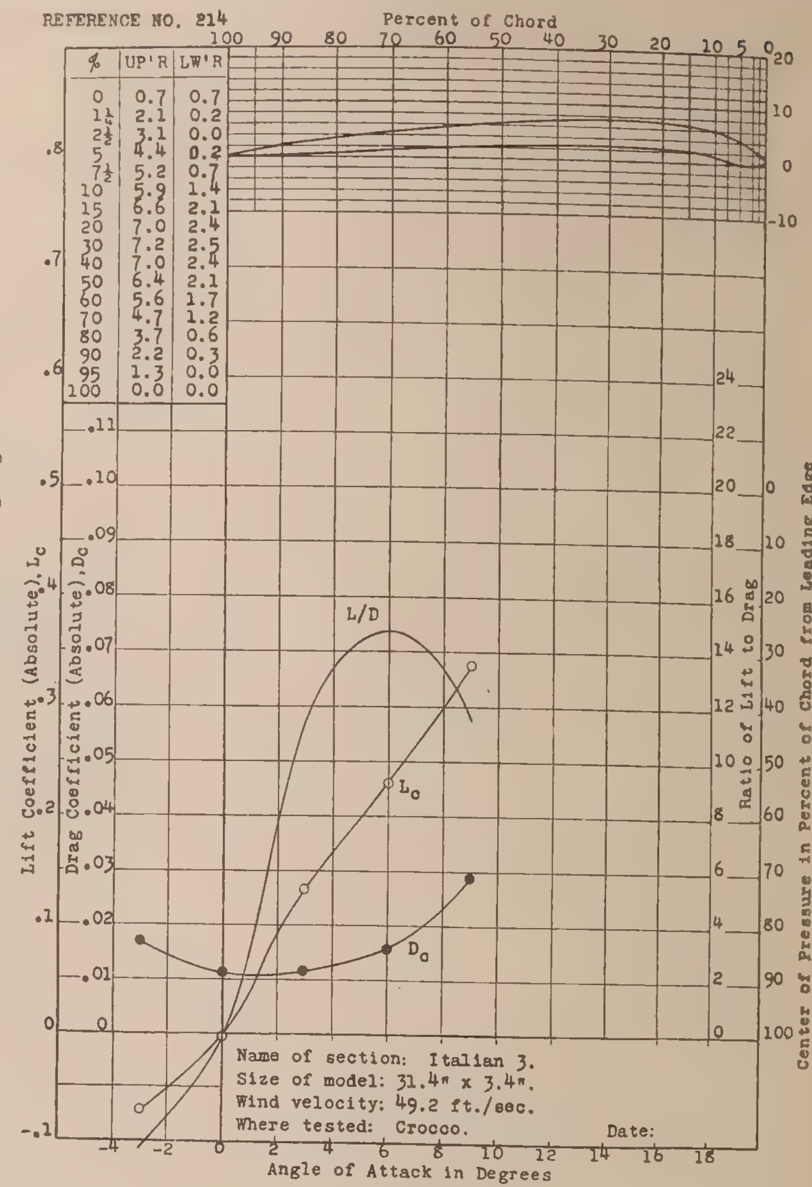
%	UP'R	LW'R
0	0.0	0.0
1¼	2.4	0.0
2½	3.2	0.0
5	4.3	0.0
7½	4.8	0.0
10	5.3	0.0
15	5.6	0.0
20	5.7	0.0
30	6.0	0.0
40	5.8	0.0
50	5.2	0.0
60	4.6	0.0
70	3.7	0.0
80	2.6	0.0
90	1.6	0.0
95	1.1	0.0
100	0.5	0.0

REFERENCE NO. 212



%	UP'R	LW'R
0	0.6	0.6
1¼	1.4	0.1
2½	2.1	0.0
5	3.1	0.1
7½	3.9	0.2
10	4.6	0.5
15	5.5	1.1
20	6.0	1.8
30	6.4	2.4
40	6.1	2.5
50	5.7	2.0
60	5.1	1.5
70	4.0	0.9
80	2.6	0.6
90	1.4	0.3
95	0.8	0.2
100	0.3	0.3


REFERENCE NO. 213
Percent of Chord
% | UP'R | LW'R
0 | 0.8 | 0.8
1½ | 1.6 | 0.3
2½ | 2.2 | 0.1
5 | 3.4 | 0.1
7½ | 4.2 | 0.2
10 | 4.9 | 0.5
15 | 6.1 | 1.1
20 | 6.7 | 1.7
30 | 7.4 | 2.7
40 | 7.5 | 3.0
50 | 7.2 | 2.9
60 | 6.7 | 2.5
70 | 5.4 | 1.8
80 | 3.8 | 0.7
90 | 2.0 | 0.4
95 | 1.3 | 0.4
100 | 0.4 | 0.4
L/D
L_c
D_c
Lift Coefficient (Absolute), L_c
Drag Coefficient (Absolute), D_c
Ratio of Lift to Drag
Center of Pressure in Percent of Chord from Leading Edge
Name of section: Italian 2.
Size of model: 47.2" x 7.4".
Wind velocity: 49.2 ft./sec.
Where tested: Crocco.
Date:
Angle of Attack in Degrees


REFERENCE NO. 214
Percent of Chord
% | UP'R | LW'R
0 | 0.7 | 0.7
1½ | 2.1 | 0.2
2½ | 3.1 | 0.0
5 | 4.4 | 0.2
7½ | 5.2 | 0.7
10 | 5.9 | 1.4
15 | 6.6 | 2.1
20 | 7.0 | 2.4
30 | 7.2 | 2.5
40 | 7.0 | 2.4
50 | 6.4 | 2.1
60 | 5.6 | 1.7
70 | 4.7 | 1.2
80 | 3.7 | 0.6
90 | 2.2 | 0.3
95 | 1.3 | 0.0
100 | 0.0 | 0.0
L/D
L_c
D_c
Lift Coefficient (Absolute), L_c
Drag Coefficient (Absolute), D_c
Ratio of Lift to Drag
Center of Pressure in Percent of Chord from Leading Edge
Name of section: Italian 3.
Size of model: 31.4" x 3.4".
Wind velocity: 49.2 ft./sec.
Where tested: Crocco.
Date:
Angle of Attack in Degrees

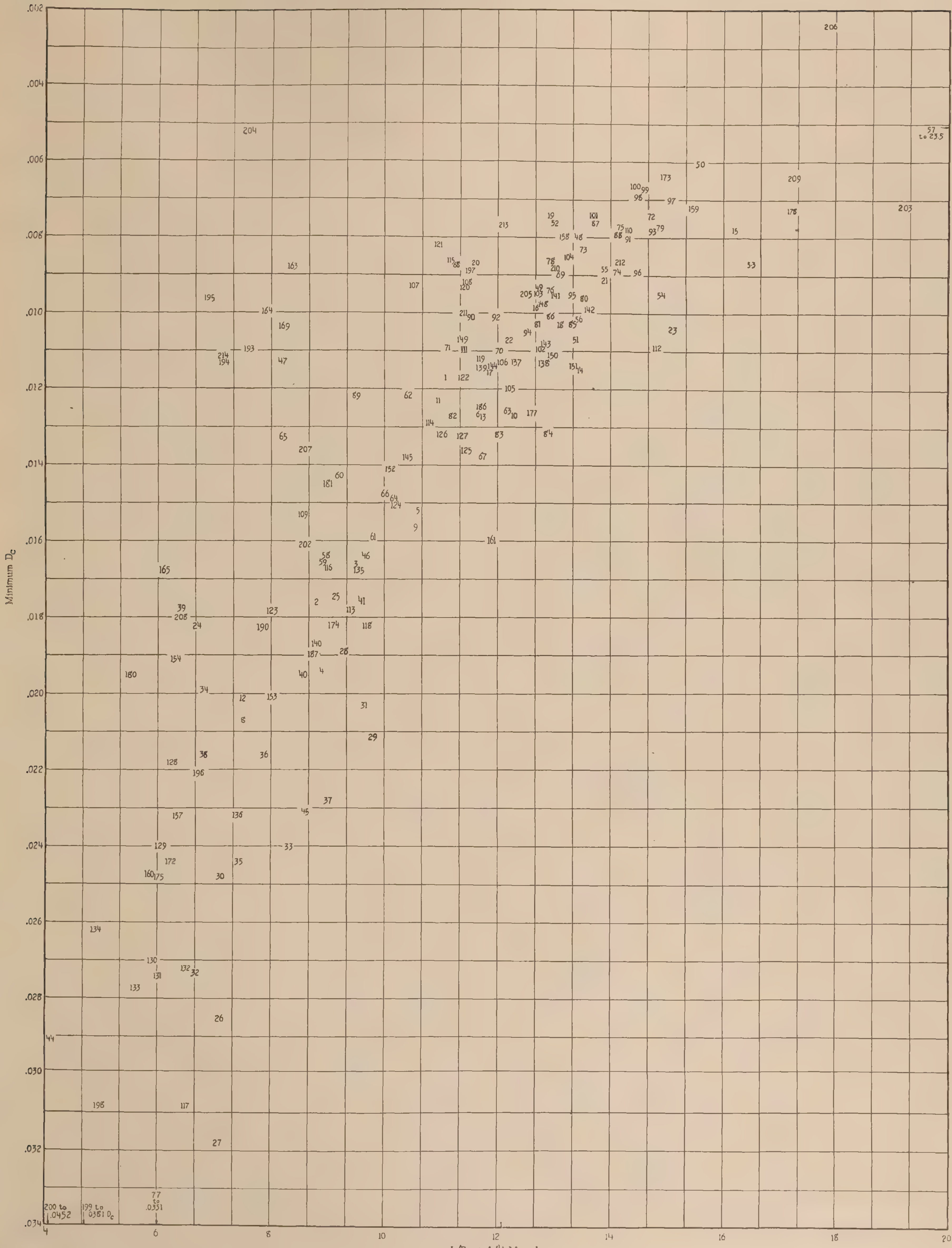


Chart No. 1

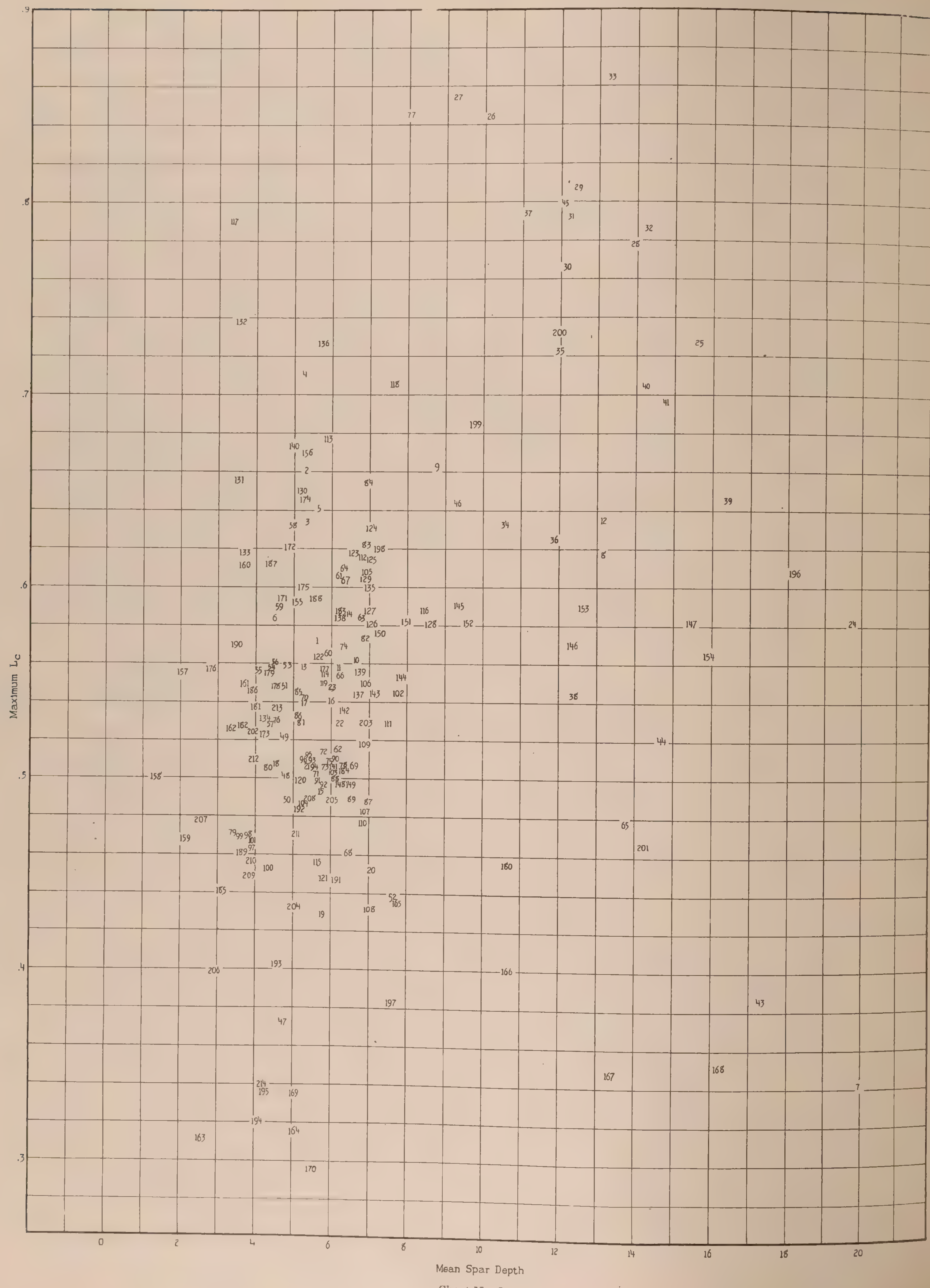


Chart No. 2.

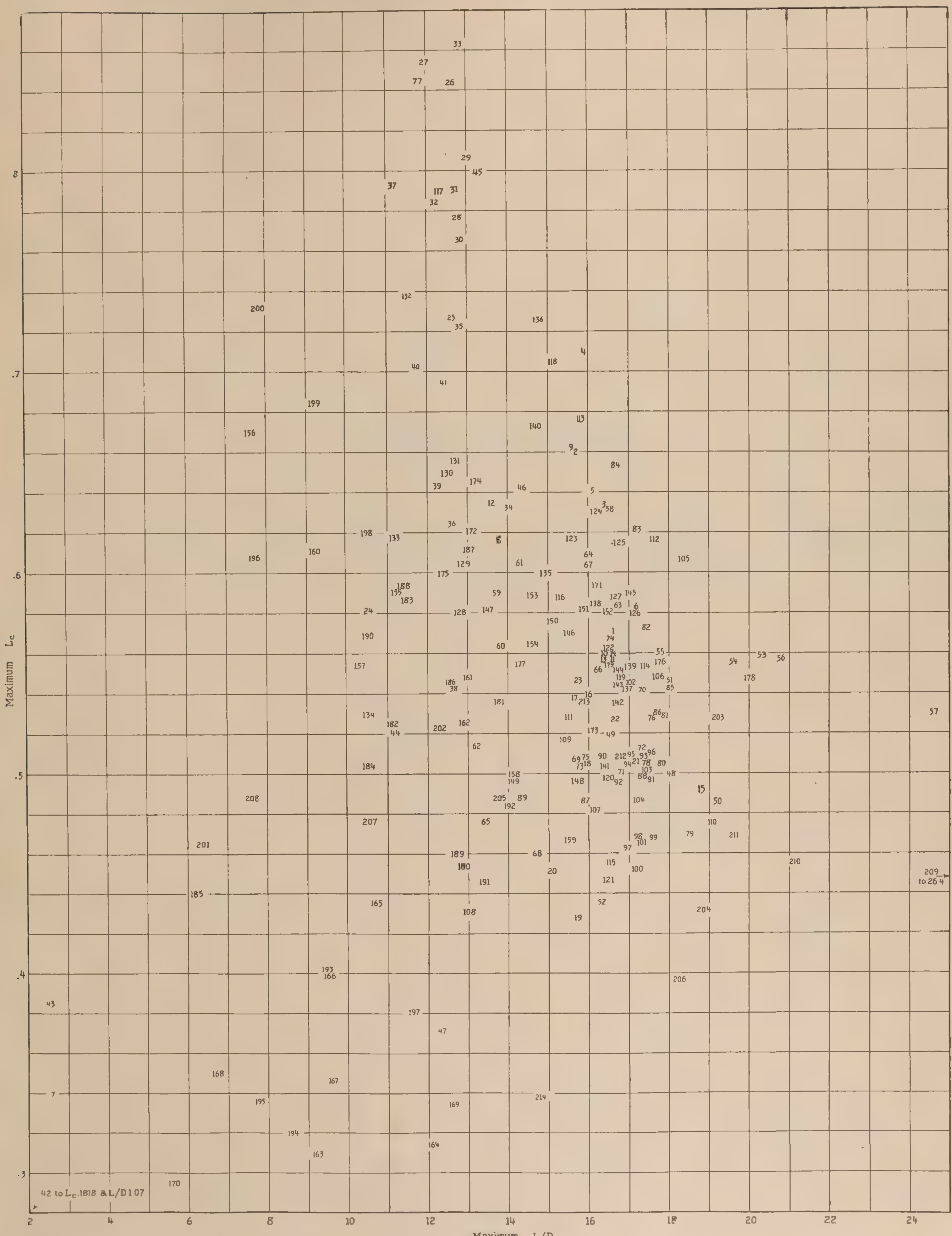


Chart No. 3.

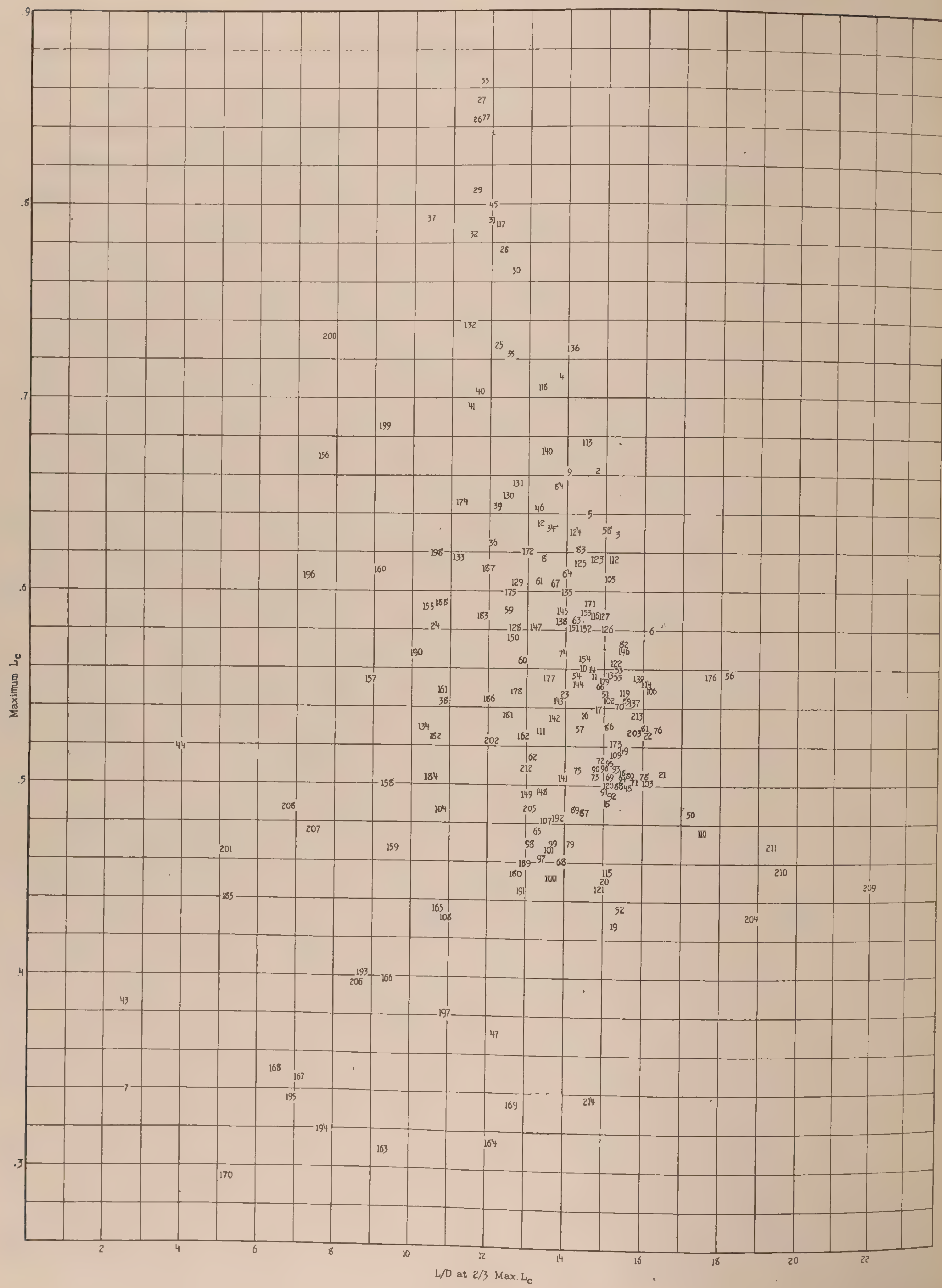


Chart No. 4.

REPORT No. 94

THE EFFICIENCY OF SMALL BEARINGS IN INSTRUMENTS OF THE TYPE USED IN AIRCRAFT

By F. H. NORTON
Aerodynamical Laboratory, National Advisory Committee for Aeronautics, Langley Field, Va.

REPORT No. 94.

THE EFFICIENCY OF SMALL BEARINGS IN INSTRUMENTS OF THE TYPE USED IN AIRCRAFT.

By F. H. Norton,

Aerodynamical Laboratory, N. A. C. A., Langley Field, Va.

SUMMARY.

This investigation was undertaken by F. H. Norton, physicist at the research laboratory of the National Advisory Committee for Aeronautics, Langley Field, Va., to supplement the rather meager data available on the construction and mechanical properties of small bearings and pivots suitable for use in aeronautical instruments. The static and running friction, for thrust and radial loads, was determined for several conical pivots and for plain cylindrical journals and ball bearings. Also the static rocking friction was measured for several conical and ball bearings under a heavy load, especially to determine their suitability for use in an N. P. L. type wind tunnel balance. It is found that for a given small load the conical pivots give less friction than any other type, and their wearing qualities, when hardened, are excellent. When the load exceeds about 1,000 gms., ball bearings give less friction than pivots, and, of course, stand shocks and wear better. Very small ball bearings are unsatisfactory because the proportional accuracy of the balls and races is not as high as in the larger sizes. For rocking pivots under heavy loads it was found that a ball and socket bearing was superior to a pivot resting in a socket. Vibration greatly reduces the static friction of a pivot.

RUNNING CONICAL PIVOTS.

The pivots oil best if mounted on the revolving part, and on very small pivots it is best to cut a small groove above the point for an oil stop, as on the balance staffs of watches. Tool steel, such as used for making taps and reamers, is most suitable for pivots. The pivots are

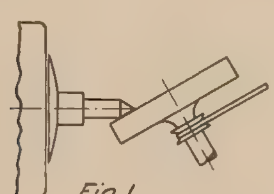


Fig. 1

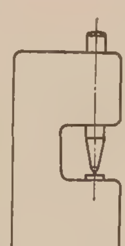


Fig. 2.

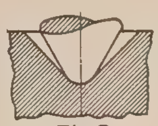


Fig. 3

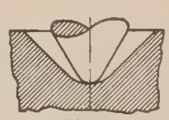


Fig. 4.

Fig. 5.

turned to size, hardened, and drawn very slightly. The point of the pivot can now be polished on a revolving lap, as shown in figure 1. Fine emery is used first, and then rouge, the hardness of the lap determining the radius of the point. A metal lap will give a small radius and a cloth lap a large one, but it is usually necessary to examine the point under a microscope and get the radius correct by hand lapping. It is not possible to grind a satisfactory pivot on a cylindrical grinder with an abrasive wheel, as the size of the grit, in even a fine wheel, approaches the diameter of the pivot at the point, and a very irregular surface is obtained, even though it may look smooth to the unaided eye. As only the extreme point of the pivot bears, this portion must be polished carefully; the surface on the remainder of the pivot is of little importance.

The sockets may be made by turning, turning and lapping, countersinking a small hole, or by punching, the last being by far the most satisfactory method. It is necessary to make a punch of the correct angle and radius of point in the same manner as the pivots, and it should be hardened and polished with the same care. It is convenient to hold this punch in a guide over the center of the blank socket, as shown in figure 2. It is very important that the punch be struck only one blow, which gives a socket as highly polished as the punch. If more than one blow is struck, the polish is lost. The socket is now hardened to the same degree as the pivot. It is important that the radius at the bottom of the socket be equal to that on the end of the pivot (fig. 3), for if it is larger the pivot will slide around (fig. 4), and if it is smaller (fig. 5) the friction is considerably increased. Before assembly both pivot and socket must be carefully cleaned to remove any chips or grit, and when together should be oiled with a light oil, such as watch oil, more for protection than for lubrication.

The running friction was determined by mounting a flywheel on the pivots to be tested, as shown in figure 6. The wheel was driven so that its speed was always the same at the start and then allowed to come to rest by its own friction, the time being taken, thus giving the relative running friction. The weight of the moving parts was 160 gms. and their moment of inertia 425 cm.^{2}, so that the radius of gyration was 1.63 cm. In order to obtain some idea of the actual value of the running friction, one set of pivots was tested with the air friction eliminated in the following manner: A hollow brass case was fitted around the wheel and belted, so that it could be driven at any speed. A small slit in the rim of the case allowed the wheel to be started by the friction of a small rubber disk mounted on a motor. This slit also allowed a black spot on the rim of the wheel to be observed. The wheel was first driven by the motor to a given speed, and the case was revolved in the same direction and speed as the wheel. It was possible to keep the box within a few revolutions a minute of the wheel at high speeds by the stroboscopic effect of the slit in the box and the spot on the wheel. At very low speeds it was more difficult to keep the speeds equal, but the air friction is almost negligible at this time.

As the only friction is pivot friction and the moment of inertia and initial speed of the wheel are known, the running friction may be found. Using the same wheel and pivot without the case, we have the same initial kinetic energy and pivot friction, but also air friction, which varies as the square of the rotational speed.

If T = average frictional moment of the pivot,

and A = average frictional moment of the air,

$$\text{K. E.} = \tfrac{1}{2} I\omega^2 = \tfrac{1}{2} T\omega t + \tfrac{1}{3} A\omega t$$

When $A = o$, t, the time to come to rest, is known for one case, and knowing I, the moment of inertia of the wheel and ω, its initial velocity, T, can be solved for. Now, substituting this value of T in the equation and calling t the time to come to rest in the open air with the same pivots, we can solve for A, the value of which is independent of the nature of the bearings used. Assuming that the friction is proportional to the time required to come to rest, it is possible by substituting in the equation, this time for any pivot, to obtain the approximate running friction for that pivot. The result obtained in this way can only be an approximation, as it involves the assumption that the frictional moment varies with velocity in accordance with the same law for all pivots, and also that the air frictional moment varies in the same manner. If it be assumed, for example, that the running friction varies with speed less rapidly than does the air friction (as is generally the case) the effect of the air friction will be less than that computed when the pivot friction is smaller than in the case used as a basis for the computations, and larger when the pivot friction exceeds that value.

The value of A was computed in one case and found to be 0.203 gm. cm. Since this moment acting alone would bring the wheel to rest in 13 minutes, and since the best pivot ran for over 20 minutes in free air it is evident that, as predicted in the preceding paragraph, the computed air friction is too high to be used in correcting the results with exceptionally good pivots. Tests were made with the shaft horizontal, and also with the shaft vertical, and the lower pivot acting as a step bearing.

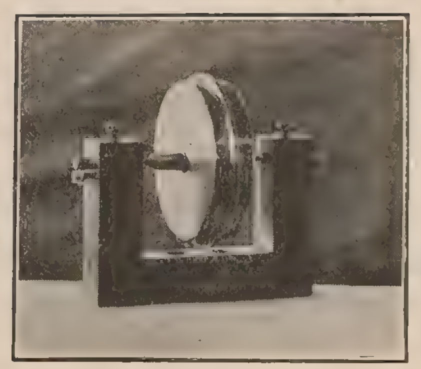


FIG. 6.

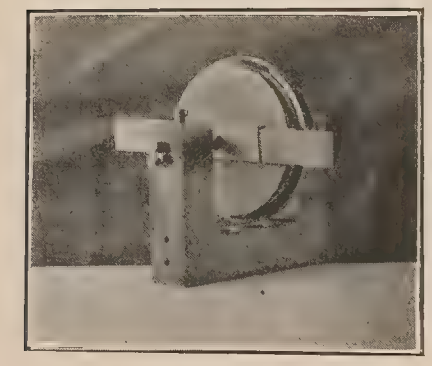


FIG. 7.

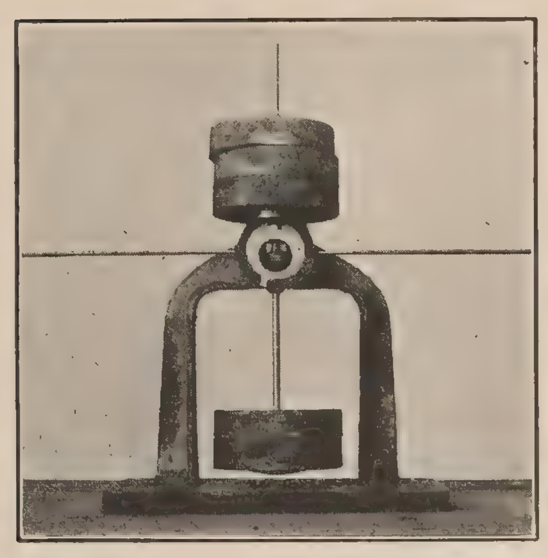


FIG. 13.

The static friction was determined by placing a graduated beam on the pivot axle and sliding a rider of known weight out from the center until the axle started to revolve. This point was found on each side of the axle and the mean of the two readings taken as the moment arm. It was found that the static friction varied for different positions of the axle, presumably because of minute irregularities in the pivots and sockets, so several readings were taken and averaged. The static friction was also determined for the shaft alone. The apparatus for doing this is shown in figure 7.

In the following table are given the properties of the conical pivots tested:

CONICAL PIVOTS.

Pivot.	Socket.	Time to come to rest from 3,000 r. p. m.		Starting moment.		Centering.	Wear.	Remarks.
		Vertical.	Horizontal.	160 gms. weight.	25 gms. weight.			
		′ ″	′ ″	cm. gm.	cm. gm.			
60°, fair polish, hardened, 0.001-inch radius.	75°, punched, 0.001-inch radius.	8 47	7 27	0.170	0.023	Very good		
Do	90°, punched, 0.001-inch radius.	11 45	9 00	.151	.011	Good		
Do	120°, punched, 0.001-inch radius.	12 22	9 22	.132	.011	Poor		
60°, rather rough soft steel, 0.002-inch radius.	90°, punched, 0.001-inch radius.	10 37	7 12	.280	.066	Good		
Do	90°, turned, 0.005-inch diameter flat spot at bottom.	8 21	8 12	.292	.033	Very poor		
Do	90°, turned, 0.013-inch hole in center.	2 50	4 10	.530	.132	Very good	Groove worn on spindle.	
Do	Coradi socket from planimeter, 90°, 0.004-inch hole.	9 55	6 50			Good	Groove worn on Coradi spindle.	Pivot broke during test.
60°, highly polished, hardened, 0.0005-inch radius.	90°, punched, hardened, high polish.	20 30	15 15	.080	.008	do		Polished on revolving lap; no wear; static friction inaccurate.
60°, highly polished soft steel, 0.0005-inch radius.	90°, punched, soft steel, high polish.	8 00	7 00			Poor		Wore badly.

Results are for two pivots, one on each side of wheel.

In figure 8 are plotted curves showing the effect of the socket angle on the friction. A 75° socket gives the best centering, and the 120° the least friction, but as a general thing a 90° socket will be found most satisfactory and is almost always used in instruments. Sockets

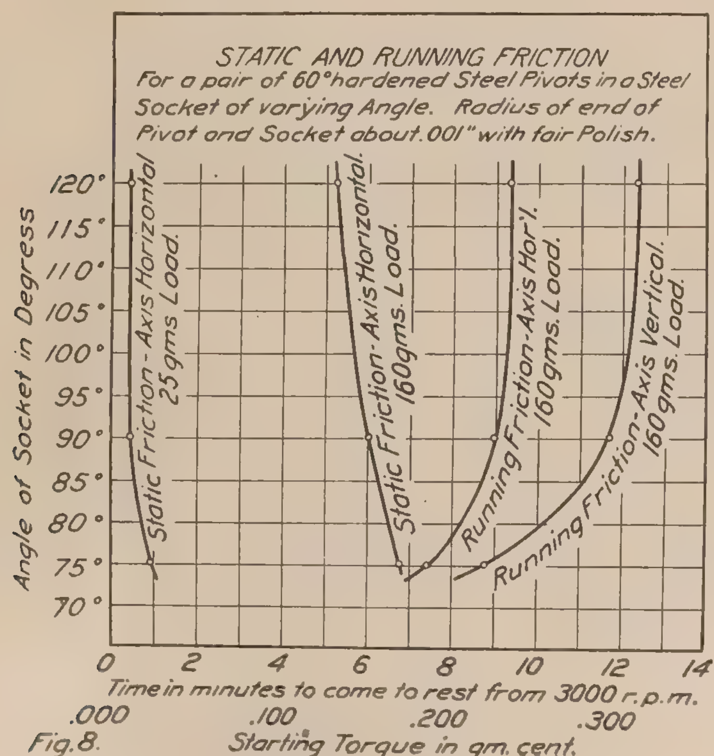


Fig. 8.

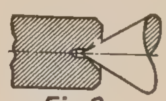


Fig. 9.

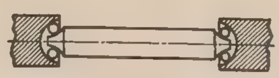


Fig. 12

with holes in the bottom give good centering but have considerable friction. An enlarged section of this type of socket, which is used on some planimeters and similar instruments, is shown in figure 9.

As shown in the preceding table, two similar sets of pivots and sockets were tested, one hardened and the other soft. The hard pivot and socket showed no signs of wear, while the soft ones wore so badly they rattled around loosely in the sockets and gave more than twice the friction of the hardened ones. Pivots were tried with and without oil and showed no appreciable difference in either static or running friction.

The radius at the point of the pivot can be made as small as 0.0005 inch when it is desired to reduce the friction to a minimum, giving a starting torque of only 0.0005 gm. cm. per gm. of weight. A pivot of this sharpness can not be used for loads much greater than 150 gms., and it is advisable for continuous running to make the radius twice as large as this. If the pivots and sockets are hardened and highly polished they make excellent bearings for light loads and give less friction than any other type.

METHODS OF HOLDING AND ADJUSTING PIVOTS AND SOCKETS.

Sockets must be capable of a fine adjustment and when adjusted be solidly supported. Several satisfactory methods of doing this are illustrated below. In figure 19 is shown the method used by Coradi. This is an excellent arrangement, as the socket is not rotated while

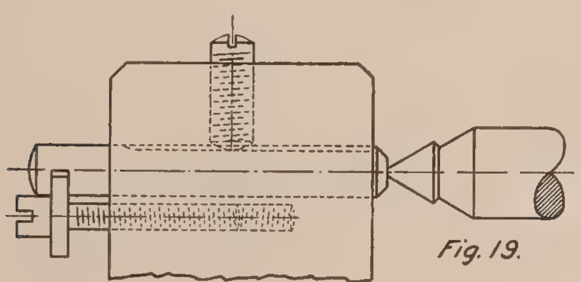


Fig. 19.

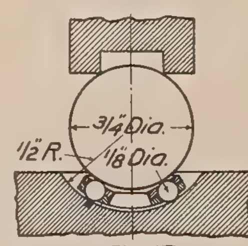


Fig. 17.

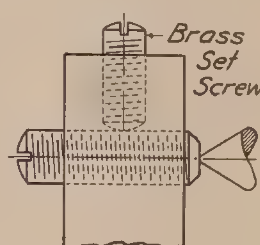


Fig. 20.

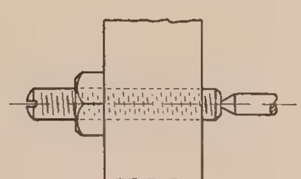


Fig. 21.

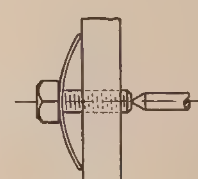


Fig. 22.

being adjusted and can be firmly clamped in place. It takes up considerable space, however, and can not be used in some locations for this reason. Another method is shown in figure 20 that is quite satisfactory and simple to construct. A screw and lock nut is sometimes used (fig. 21) but is not suited to fine adjustment. Figure 22 shows the method of locking the balance sockets in clocks by means of a cupped spring washer. This method gives only a small adjustment and rather insecure locking.

RUNNING CYLINDRICAL BEARINGS.

The cylindrical pivots were turned as smoothly as possible on a light lathe and were given a fair polish with crocus cloth, but were not hardened. The sockets were drilled with an ordinary twist drill and were not lapped out. The pivots were an easy fit in the sockets, but no looseness could be felt when they were oiled. The friction did not, however, seem to be altered by any reasonable amount of looseness. The static and running friction was determined in the same way as before, but the bearings were run in until the friction was constant.

All bearings were oiled with porpoise-jaw oil. The results obtained are given in the following table:

CYLINDRICAL BEARINGS.

Pivot.	Socket.	Time to come to rest from 3,000 r. p. m., shaft horizontal.	Starting moment— 160 gms. weight.	Starting moment— 25 gms. weight.	Wear.	Mean running friction.	Remarks.
		′ ″				gm. cm.	
1/64 inch long, average diameter 0.0964 inch	Brass	0 44	4.70	0.60	None	2.9	
1/64 inch long, average diameter 0.0520 inch	do	1 38	1.92	.24	do	1.27	
1/64 inch long, average diameter 0.0335 inch	do	2 10	.89	.27	do	.85	
1/64 inch long, average diameter 0.0215 inch	do	4 03	.53	.19	do	.37	
1/64 inch long, average diameter 0.0157 inch	do	4 32	.50	.11	Considerable.	.34	
1/64 inch long, average diameter 0.0124 inch	do	4 32	.39	.03	do	.34	Pivot broke fourth run.
1/64 inch long, average diameter 0.0520 inch	Steel	1 28					
1/64 inch long, average diameter 0.0124 inch	do	4 35					Pivot broke third run.
1/64 inch long, average diameter 0.0215 inch	Brass		.08				Vibrating under same conditions as in airplane.
1/64 inch long, average diameter 0.0964 inch	do		.21				Do.

Results are for two bearings, one on each side of wheel.

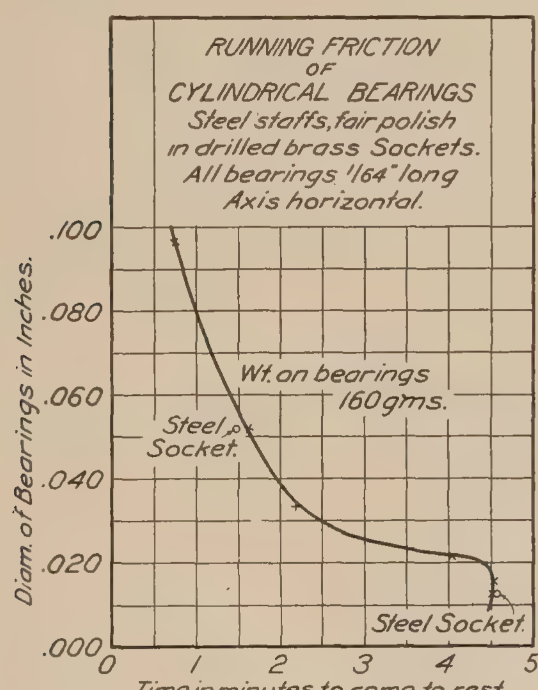


Fig. 10.

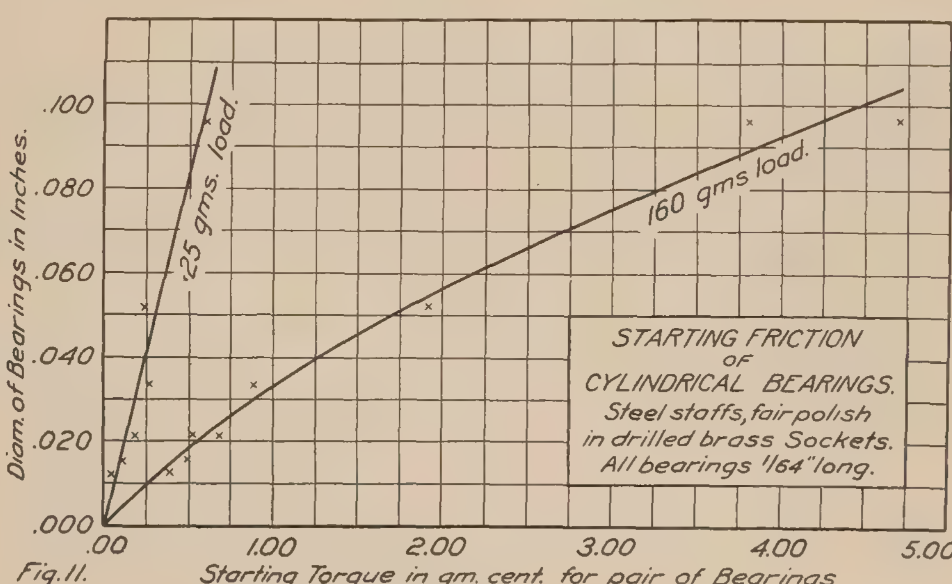


Fig. 11.

A curve of bearing diameter against running time is plotted in figure 10. The friction decreased rapidly with the diameter, until about 0.035 inch is reached, then the slope of the curve becomes much less down to a diameter of 0.020 inch. At this point the curve turns down sharply, so that no decrease in friction is obtained by reducing the diameter further. It is probable that the oil film breaks down at this point, as evidenced by the rapid wear below this diameter. For diameters in excess of this critical value the curve is approximately a rectangular hyperbola, indicating that the frictional moment is directly proportional to the diameter, and, therefore, that the frictional force acting at the periphery depends only on the magnitude of the total load and not on the intensity of pressure. It is evident that a small bearing should not carry over 500 pounds per square inch of projected area and 300 pounds would be safer. Two sizes of socket were tried of steel but the same result was obtained.

In figure 11 the static friction is plotted against bearing diameter for two loads. The values are rather irregular because of inequalities in the bearing surfaces, but it is evident that the curves are nearly straight lines starting at the origin and also that the ratio of the slopes of the two lines is the same as the ratio of the two weights. It may be concluded that the static friction increases as the weight on the bearing and as the bearing diameter, but is independent of the bearing length. The static friction of any small bearing (brass or steel, lubricated) is given by the formula following.

$$T = K D L,$$

where T = the starting torque in gm. cm.,
L = load in gms.,
K = 0.23 when D is in inches,
K = 0.0091 when D is in mm.,
D = diameter of bearing.

In order to determine the effect of lubrication two sizes of bearing were run with and without oil. It was impossible to detect any difference between oiled and dry bearings by the starting torque, but the oiled bearings had about half as much running friction as the dry ones. It is probable that the heavily loaded shaft cuts through the oil film when it is at rest, so that the same conditions of starting torque prevail whether oil is present or not.

The values of "mean running friction" given in the table were computed with due allowance for air friction in the manner already described. It will be noted that the mean lubricated running friction was only very little less than the static friction, and the mean dry running friction was therefore distinctly greater than the static. Since running friction at low speeds is always less than static friction, this excess must be attributable to variation of running friction with speed. It may be caused by heating, expansion, and partial seizure of the shaft, with a resultant great increase in friction when running at high speeds.

As the bearings of airplane instruments are used under conditions of vibration such conditions were simulated by placing the testing apparatus on a 2-horsepower electric motor frame and running the motor slightly out of balance at 1,800 r. p. m. In the case of a $\frac{3}{32}$ inch diameter bearing the static friction was reduced to less than one-twentieth of its steady value by vibration. The starting friction of a smaller pivot, 0.0215 inch in friction, is very marked, especially with large bearings, and should be taken into account when designing instruments for these conditions. The gain, however, is not quite as great as might at first appear, for the pivots must be made larger to stand the strains of vibrating conditions. Advantage is taken of this means of reducing friction in some sensitive wireless relays where a clock taps the frame of the instrument at short intervals.

In order to give a better idea of the size of bearings tested, the following table gives the diameters used in clocks and watches:

Type of staff.	Diameter.
	Inch.
Alarm clock escape and second hand	0.021
Alarm clock wheel up to minute hand	.030
Alarm clock main spring	.050
Watch escape and second hand	0.010-.013
Watch balance wheel, not jeweled	.007
Watch escape, jeweled	.007
Watch balance wheel, jeweled	.004

The cylindrical bearings have much more friction than conical pivots, but need not be hardened and do not require the delicate adjustment necessary with the pivots.

RUNNING BALL BEARINGS.

Three small ball bearings were tested for static and running friction. The first two were stock radial bearings, and the third was a cone bearing made with $\frac{1}{16}$-inch balls, as shown in figure 12. The description of the bearings and their friction is given in the following table:

Bearing.	Static friction—		Time to come to rest from 3,000 r. p. m.		Remarks.
	160 gms.	60 gms.	Horizontal.	Vertical.	
			′ ″	′ ″	
High-grade bearing: $\frac{5}{8}$ inch outside diameter, $\frac{5}{32}$ inch inside diameter, $8\frac{1}{8}$-inch balls; no retainer.	0.75	....	1 20	....	Ran smoothly.
Cheap pressed races: $\frac{11}{16}$ inch outside diameter, $\frac{3}{16}$ inch inside diameter, $15\frac{3}{32}$ inch balls; no retainer.	2.25	....	23	....	Races rough.
Cone and cup bearing: $\frac{7}{16}$ inch outside diameter, $\frac{1}{8}$ inch inside diameter, $10\frac{1}{16}$-inch balls; no retainer.	.27	....	3 15	3 30	The balls were not closely enough of the same size, and did not run smoothly. Very difficult to assemble.

Friction is for a pair of bearings. All runs with dry bearings.

The first bearing was the most satisfactory and has about the same friction as a $\frac{1}{32}$-inch diameter cylindrical bearing and, of course, will carry enormously greater loads. It is evident that small ball bearings can not be made at present with enough accuracy to compete with pivots, under light loads (less than 1,000 gms.). However, for continuous running, and where it is necessary to pass the axle through the bearing, ball bearings can be used to advantage. The static and running friction is considerably increased by oil, but as this increase is independent of the load, oiling makes little difference on a heavily loaded bearing.

ROCKING PIVOTS.

The apparatus used to measure the static friction is shown in figure 13. It is simply a pendulum with horizontal arms to carry the riders. The weights are so placed that the center of gravity of the moving parts is at the pivot point, and one of the riders is pulled out until the pendulum starts to move. This is done for each rider in turn, and the mean reading is taken as the moment arm. In order to be sure that the pivot is unstrained at the beginning of the test, the end of one of the arms is held and the pendulum rocked slightly in a plane at right angles to the plane in which the test is carried out. As it was rather difficult to adjust the center of gravity exactly at the pivot point, the moment arm for the sharper pivots could not be accurately determined, and the values given may be in error as much as 50 per cent. In the first test the sockets were sharp at the bottom and had an angle of 120°, and the pivots were 90° with various radii at the points. In figure 14 is plotted the starting torque in gm. cm., against diameter of the point. With the lighter load the torque increases uniformly with the diameter of the point and the curve passes through the origin. As the load is increased the curves still start for the origin at large diameters, but as the diameters are decreased the curves fall below the straight line, due probably to flattening of the point after the unit load exceeds a certain value.

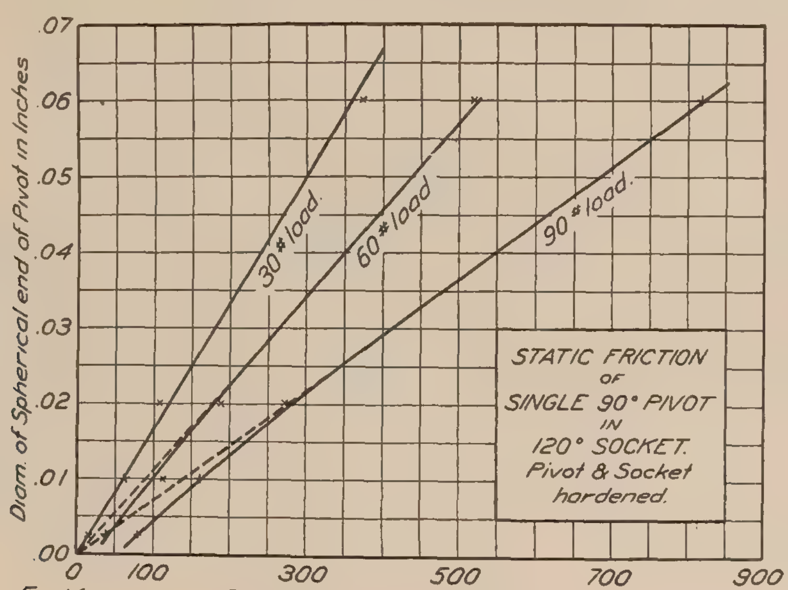


Fig. 14.

Pivots that were loaded above this value showed wear and distortion. It seems evident that with a 120° socket and 90° pivot that the greatest permissible load will be given by:

$$L = 3{,}000\ d$$

where d is the diameter in inches of the pivot point

and L = total load in pounds.

The friction of these same pivots when resting on a flat plate, instead of on a socket, is about halved, but of course would stand very litte tangential load. In all cases when the loading was high the point left an indentation on the plate. One pivot and socket was inclined 15° in order to introduce a tangential component similar to that found in wind tunnel balances. The friction was the same as when the load line passed through the pivot axis.

As the same socket was used for all sizes of pivot, the rounded pivots rested in the socket as shown in figure 15, bearing on an annular area. If a socket had been used in each case to

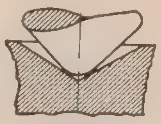


Fig. 15.

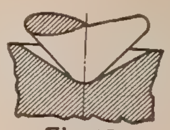


Fig. 16.

fit the pivot point the bearing would be as shown in figure 16, obtaining more area without increasing the diameter, thus carrying more load with the same friction, but with some danger of imperfect centering.

ROCKING BALL BEARING.

A hemispherical socket with a 1-inch radius was turned out, hardened, and lapped down with a steel ball as a lap to a good polish. A ring of $\frac{1}{8}$-inch balls held in a retainer was placed in the socket, and on these a $\frac{3}{4}$-inch steel ball rested. A section of the bearing is shown in figure

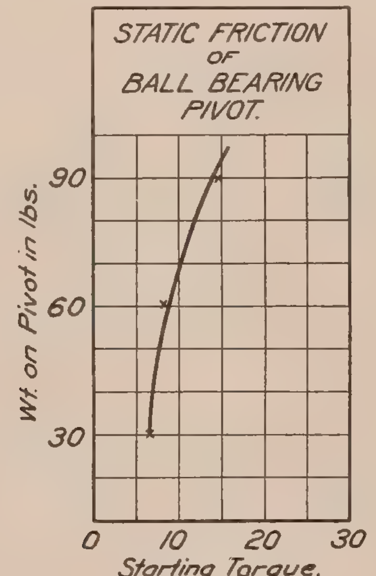


Fig. 18.

17. This bearing was tested in the same way as the pivots, and with all loads had less friction than the sharpest pivot, while it is evident that it could support many times the load (fig. 18). The small balls showed no tendency to crawl out of the socket, and the centering was very good. This type of bearing seems to be better than a pivot for wind tunnel balances.

REPORT No. 95

DIAGRAMS OF AIRPLANE STABILITY

By H. BATEMAN
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REPORT No. 95.

DIAGRAMS OF AIRPLANE STABILITY.

By H. Bateman,

California Institute of Technology.

INTRODUCTION.

§1. This report was prepared by Dr. H. Bateman for publication by the National Advisory Committee for Aeronautics. The theory of small oscillations about a state of steady motion which was developed many years ago by E. J. Routh[1] has been applied with marked success in aerodynamics, the desired simplicity of the equations being secured by the introduction of the resistance coefficients by G. H. Bryan.[2] This simplification of the equations is based on the assumption that in a slight departure from a state of steady motion the increments in the component aerodynamical forces and couples can be expressed in terms of the increments of the component velocities of translation and rotation alone without any additional terms depending, for instance, on the increments of the accelerations. This assumption seems to give a good approximation to the truth in the case of an airplane, but in the case of a balloon the additional terms are required. When a flying machine is treated as a rigid body the general type of steady motion is one in which the center of gravity describes a helix and the algebraic equation which determines the temporal characteristics of the oscillations is of the eighth degree, but this equation can be simplified in certain cases. In the case of an airplane having a plane of symmetry, the oscillations about a state of steady rectilinear flight can be regarded as built up from longitudinal and lateral oscillations which are practically independent of one another. When certain resistance coefficients are assumed to be zero each set of oscillations is associated with an algebraic equation of the fourth degree.[2]

A notable simplification also occurs in the case of a body like a parachute which has an axis of symmetry, when the steady motion is rectilinear and in the direction of the axis of symmetry.

In a recent report on the dynamical analysis,[3] Messrs. Klemin, Warner and Denkinger have studied the effect on the period and rate of subsidence of the pitching oscillations of an airplane of a change in one of the resistance derivatives when all the others are kept constant. It occurred to the author that it might be worth while to continue this work by considering simultaneous variations in two or three of the derivatives and to pay special attention to cases in which a slight change in some of the derivatives has (1) no effect, and (2) a marked effect in the characteristics of the long oscillation.

The results which have been obtained are exhibited by means of diagrams in which two of the resistance coefficients are used as coordinates, and curves are drawn along which the modulus of decay of a long oscillation has a constant value, numbers being given to indicate the value of the period at various points of each curve. At Mr. Warner's suggestion numbers have been given in some cases to indicate the ratio of the period to the time of damping to half the initial amplitude, as this quantity is adopted as a measure of stability in the report to which we have just referred.

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[1] The Stability of a Given State of Motion (Adams Prize Essay). London, 1877. Advanced Rigid Dynamics. Chaps. III and VI.

[2] Stability in Aviation. Macmillan & Co. (1911).

[3] Third Annual Report of the National Advisory Committee for Aeronautics. No. 17, p. 330. (1917.) This report will be referred to subsequently as 17.

The method has also been adapted to the lateral oscillations of a symmetrical airplane and to the oscillations of a parachute. The graphical method used here is inferior in many respects to the beautiful one devised by L. Bairstow and J. L. Nayler, British Advisory Committee's Report No. 116, 1915, but the work was completed before this report came to the writer's notice.

§2. *Pitching oscillations.*—When the pitching oscillations are regarded as independent of those in roll and yaw, the biquadratic equation which determines the temporal characteristics of the oscillations may be written in the form

$$\lambda\,(A\lambda+y)(\lambda+z)(\lambda+w)+x\left[\lambda(\lambda+z)+\lambda\frac{g}{V}\sin\theta_0\right]+\xi\eta\lambda(A\lambda+y)$$

$$(1)\qquad +(\xi x+\zeta w)(\lambda\delta+\cos\theta_0)+(\eta\zeta-zx)\frac{g}{V}(\lambda\delta-\sin\theta_0)+\lambda\zeta(\cos\theta_0-\eta)=o$$

where

$$x=-VM_w,\ y=-M_q,\ z=-X_u,\ w=-Z_w,$$

$$\xi=\frac{g}{V}Z_u,\ \eta=-\frac{V}{g}X_w,\ \zeta=-gM_u,\ \delta=\frac{1}{g}X_q,$$

$$(2)\qquad V=U+Z_q+X_q$$

and the notation is the same as in the reports of Hunsaker, Klemin, Denkinger, and Warner.

Let $\lambda^2-2\alpha\lambda+\gamma$ be one factor of the expression on the left hand side of equation (1). Replacing λ^2 by $2\alpha\lambda-\gamma$ we can reduce the above expression to one which is linear in λ and equate to zero the terms with and without λ. This gives us the two equations

$$(3)\quad \gamma[y+A(w+z)+4A\alpha]=(y+2A\alpha)(w+2\alpha)(z+2\alpha)+\xi\eta(y+2A\alpha)$$
$$+x\left[2\alpha+z+\frac{g}{V}\sin\theta_0+\delta(\xi-\frac{g}{V}z)\right]+\zeta\left[\delta(w+\frac{g}{V}\eta)+\cos\theta_0-\eta\right],$$

$$(4)\quad A\gamma^2-\gamma[(y+2A\alpha)(w+z+2\alpha)+A(wz+\xi\eta)+x]+\cos\theta_0(\xi x+\zeta w)-\frac{g}{V}\sin\theta_0(\eta\zeta-zx)=o,$$

which generally determine x and y uniquely [4] when α and γ are given, that is, when the period $p=\frac{2\pi}{\sqrt{\gamma-\alpha^2}}$ and the co-efficient of subsidence $=\alpha$ of the oscillation are given. Instead of the latter quantity it is convenient to use the time

$$t=\frac{\log_e 2}{-\alpha}=\frac{.69}{-\alpha},$$

which represents the time which it takes for the amplitude of a simple oscillation to fall to half value.

With the aid of equations (3) and (4) the curves $t=$constant ($\alpha=$constant) have been drawn in the (x, y) plane for various values of $z, w, \xi, \eta, \zeta, \delta$, and θ_0, the value $A=100$ being adopted in each case. We can use the same diagrams for any other value of A by simply altering the scales for x, y, and ζ. It should be noticed in fact that equation (1) is still satisfied if we replace A, x, y, ζ by $\kappa A, \kappa x, \kappa y, \kappa\zeta$, respectively, keeping the other quantities the same.

In diagrams I–V there are two sets of curves corresponding respectively to the values $\eta=1$ and $\eta=2$. Each set of curves is made up of three pairs, the two curves of each pair correspond

[4] They may fail to do this for certain particular values of α and γ when the two linear equations in x and y are the same. It should be noticed however, that when $\theta_0=o$, $\delta=o$, $\zeta=o$, the equations give

$$(2\alpha+w+z)\,[w^2+wz(\eta-1)+\xi(\eta-1)^2]=o$$

When $2\alpha+w+z=o$ we have $\gamma=\xi\eta+wz$ and the equation of the line along which both α and γ are constant is $x=o$. This is one of the boundaries of the region of stability in the (x, y) plane. The second factor $w^2+wz(\eta-1)+\xi(\eta-1)^2$ is generally positive with the values of the resistance coefficient usually found for an airplane. If it could vanish there would be an infinite number of straight lines along which α and γ are constant and connected by the relation

$$\gamma=\xi+\frac{w}{\eta-1}(2\alpha+z).$$

to the same value of t, but the upper curve corresponds to the value $\zeta=1$ and the lower curve to the value $\zeta=o$.

The values of z, w, ξ, in diagram I are roughly those found for the Curtiss J. N. 2 in Report 17. A few slight changes have been made to facilitate the calculations, but these do not affect the general conclusions. The values of η in diagram I are greater than the number derived from the value of X_w given in Report 17. This number is about 0.6 and it is remarked on page 332 that X_w decreases as the angle of incidence increases and the speed decreases. Since X_w may actually become negative as the critical speed is approached, curves have also been drawn for the values $\eta=o$ and $\eta=-1$.

In studying these diagrams it should be noticed in the first place that the lines $t=\infty$ and $x=0$ limit the region of stability. The first of these lines is curved. When η is decreased this line rises and the region of stability becomes more restricted. On this account alone we can expect a decrease in η to be unfavorable when the other resistance derivatives remain constant. This agrees with the result found in Report 17 but it is worth while to consider the matter more fully.

If we take any value of η and begin to draw the curves $t=$constant, starting from $t=\infty$ and gradually decreasing t, we find that a certain minimum value t_o is reached below which the curve no longer lies in the part of the region of stability shown on the diagram. It is clear from the diagrams that t_o increases when η decreases. This increase in the time of damping is partly offset by an increase in period; but, if the other resistance derivatives remain constant, the ratio of the period to the time of damping apparently decreases with η. This is seen more easily by looking at diagrams Ic and Ib.

If some of the other resistance coefficients alter at the same time as η the unfavorable effect of a decrease in the value of η may be partially or completely offset. Thus if, when we decrease η from 2 to 1, we increase y so as to keep t constant, we increase the period p and so improve the stability as far as the long oscillation is concerned. With small values of η a considerable increase in y may be needed to keep t constant when η is decreased, but a much smaller increase in y may be sufficient to offset the unfavorable effect of the decrease of η. The effect of decreasing η may also be offset by decreasing x; for, if we decrease x so as to keep t constant, the period p is seen to be increased. This may be seen very clearly in diagram I if we start with the point $t=13.8$, $p=20.1$, $\eta=2$, and pass first to the point with the same coordinates in the part of the diagram corresponding to $\eta=1$ and then proceed along a line $y=$constant until the curve $t=13.8$ is reached. A comparison of the different diagrams indicates that the general form and arrangement of the curves is roughly the same for the different sets of values of z, w, and ξ. An exception occurs in the case of the curves for $\eta=-1$ in diagram Ia. It will be seen that in this case the curves $\zeta=0$ cross the curves $\zeta=1$ while in the other diagrams a curve $\zeta=0$ remains below the corresponding curve $\zeta=1$.

It is generally assumed that $\zeta=0$, since the moment about the center of gravity of the airplane due to air forces is zero in horizontal flight and therefore will not be affected by a change in speed. It is easy, however, to imagine some arrangement which will make M_u and therefore ζ different from zero. The inclination of a flap held by a spring and exposed to the air will vary with the speed of the airplane, consequently the force on it will not be proportional to the square of the speed, and so ζ will be either positive or negative.

It will be seen from the diagrams that a positive value of ζ is generally unfavorable to stability. When η is negative, a machine with a moderately large value of x may be an exception to this rule; but, when X_w is positive, the stability can apparently be improved by making ζ negative.

If this is done with the aid of a flap held by a spring,[5] changes may also be produced in the other resistance derivatives, particularly X_u and Z_u. We must therefore also study the effect on stability of changes in ξ and z. This may be done with the aid of diagrams I–V.

[5] The stability of an aeroplane which has springs in the control connections has been discussed in a more rigorous manner by L. Bairstow and R. Jones. British Advisory Committee for Aeronautics. (R—M No. 210.)

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It appears that an increase in ξ from 2 to 3 has practically no effect on the position of the curves t=constant, but it does lower the period; hence an increase in ξ seems to be unfavorable.

A comparison of diagrams IV and V indicates that an increase of z from 0.1 to 0.2 lowers the curves t=constant and is on the whole favorable to stability.

The general conclusion then is that, if some kind of a flap held by a spring is to be used to improve the stability, as far as the long oscillation is concerned, it should be chosen so as to decrease Z_u, increase $-X_u$, and make M_u positive.

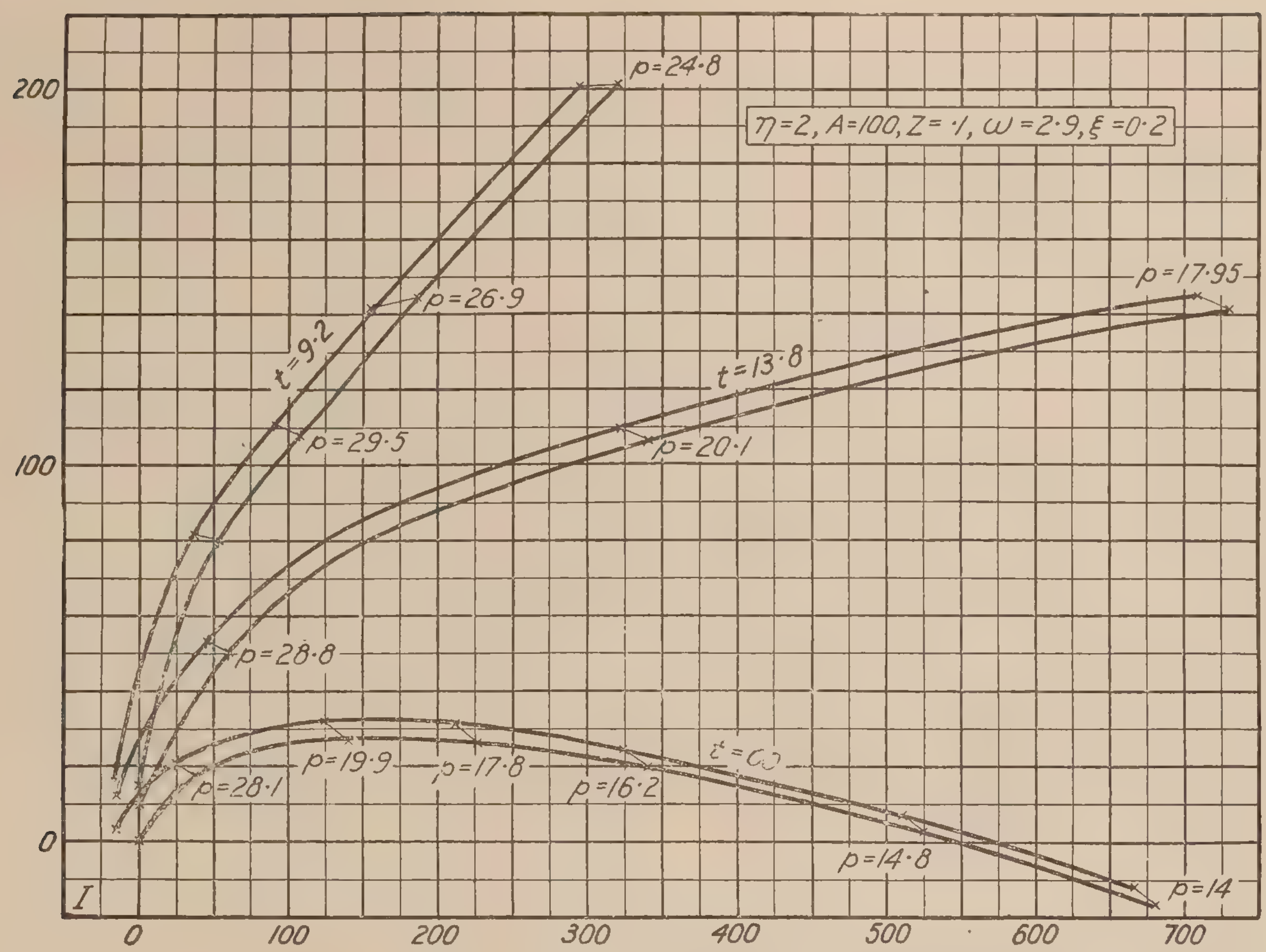
The diagrams may also be used for other purposes; in particular, they may be used to confirm some of the conclusions in Report 17. It will be noticed that as a point moves toward the origin along a curve t=constant, the period p increases, and so the stability is improved.[6] It may then be advantageous to decrease both VM_w and $-M_q$, but $-M_q$ must not be decreased too rapidly. There is a limit to the ratio of $-dy$ to $-dx$ if we wish the change to be favorable to stability, and this limiting value may be shown clearly on the diagram by drawing the curves for which $\frac{p}{t}$ is constant, in accordance with a suggestion made by Mr. Warner. These curves are shown in diagrams Ib and Ic. It will be noticed that the curves $\frac{p}{t}$ constant are steeper than the curves t=constant, but are not quite as steep as the curves p=constant.

A comparison of diagrams I and II indicates that an increase of w from 2.9 to 3.9 increases the period p and has more effect on the time of damping t when η is negative than when η is positive. When $\eta=2$, the time t apparently increases with w; but when $\eta=1$, t increases for small values of t, e. g., those less than 13.8, and decreases for large values of t. When $\eta=0$ and $\eta=-1$, t decreases as w increases, the effect being quite marked when $\eta=-1$.

In diagram VI the effect is shown of making δ different from zero. It appears that by making δ positive we decrease the time of damping and produce very little change in the period. Other things being equal, an increase in δ seems to be favorable to stability. It should be noticed, however, that X_q occurs in the expression for V, consequently unless an increase in X_q is balanced by a decrease in Z_q the numerical value of V will be lowered and the values of the quantities depending on V altered. It appears then that the effect of an increase in X_q may be offset by the change in V. It should be observed, however, that, when X_q is about 3, the percentage change in the value of V is not large, and so the effect of the increase in δ should predominate over the effect of the change in V. In diagram VI σ is written in place of δ.

In diagram VII the effect is shown of a change in θ_o, other things remaining the same. It appears that the stability is improved by making θ_o positive, for the curves t=constant are lowered. By making θ_o negative, the time of damping is increased. In particular, if the speed were not increased, the time of damping would be greater in a vertical dive than in horizontal flight. This may be seen when the stability of the parachute and helicopter are studied. (See Report No. 80.)

[6] It should be noticed that the nearer the representative point is to the origin of coordinates the more marked is the effect on p and t of small changes in x and y.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p=26·9
t=9·2
p=29·5
p=17·95
t=13·8
p=20·1
p=28·8
p=19·9
p=17·8
p=16·2
p=28·1
p=14·8
p=14
200
100
0
I
0
100
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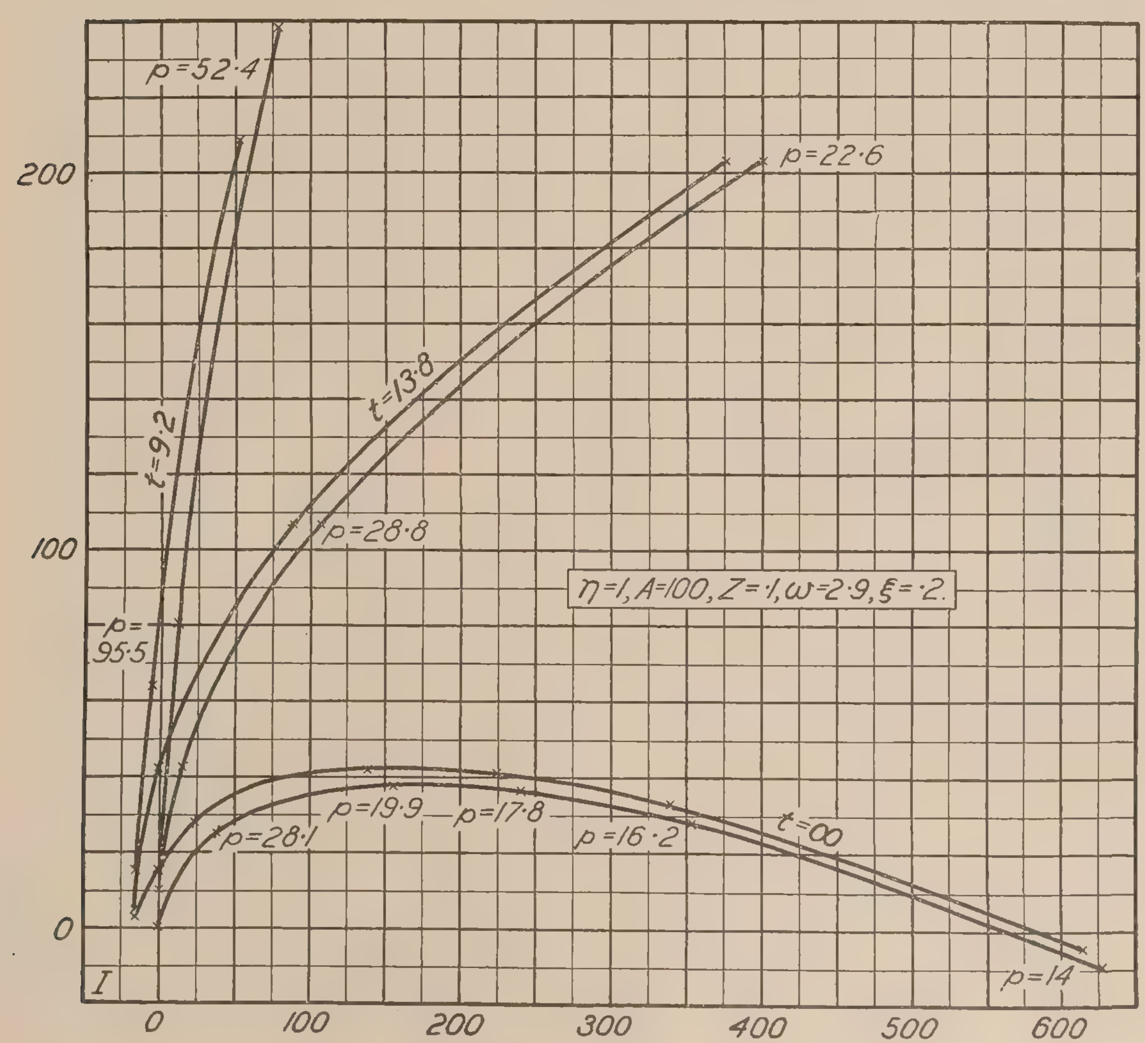
300
400
500
600
700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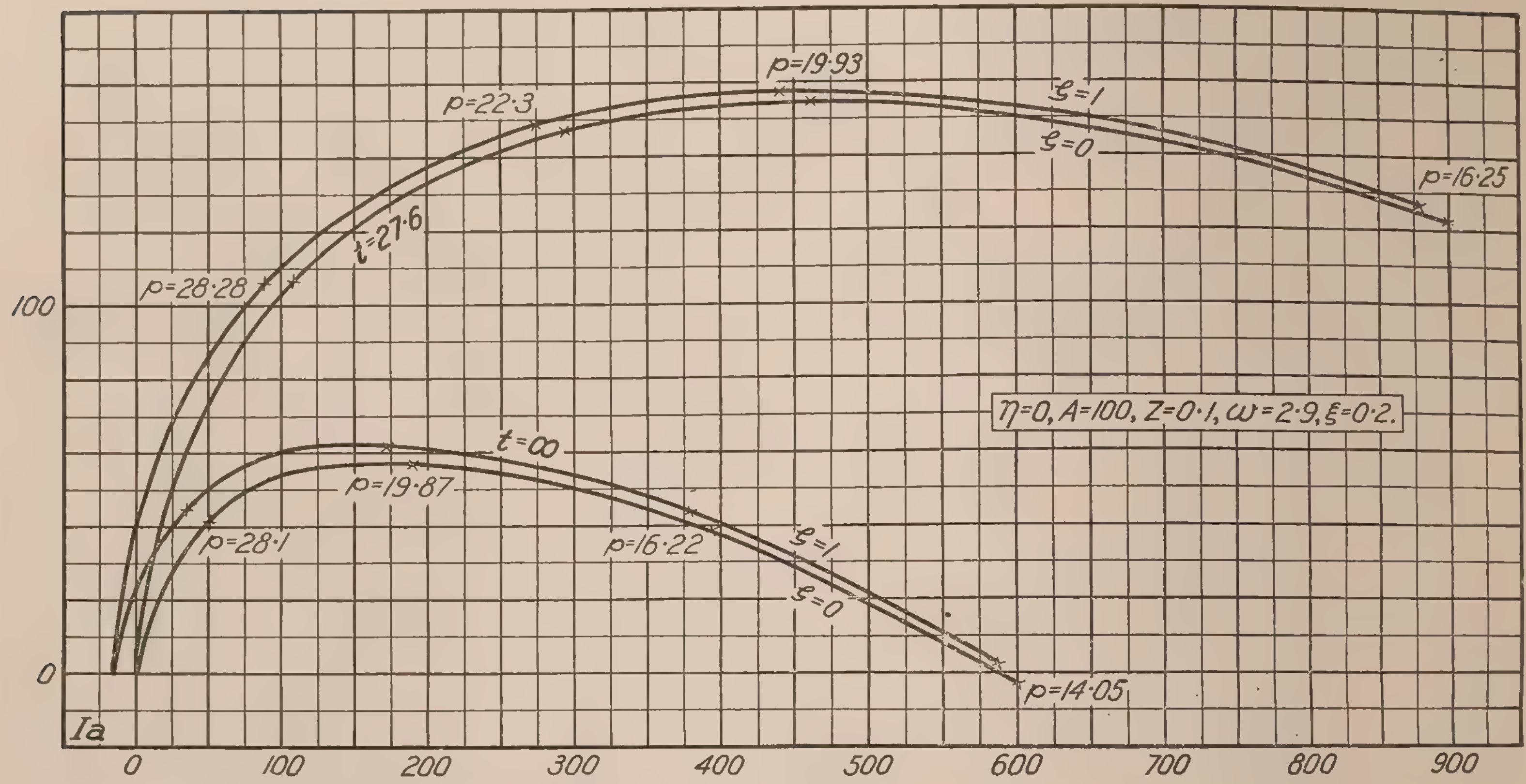
p=22·6
t=13·8
t=9·2
p=28·8
η=1, A=100, Z=·1, ω=2·9, ξ=·2.
p=
95·5
p=19·9
p=17·8
p=16·2
p=28·1
t=00
p=14
200
100
0
I
0
100
200
300
400
500
600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p=22·3
ς=1
ς=0
p=16·25
t=27·6
p=28·28
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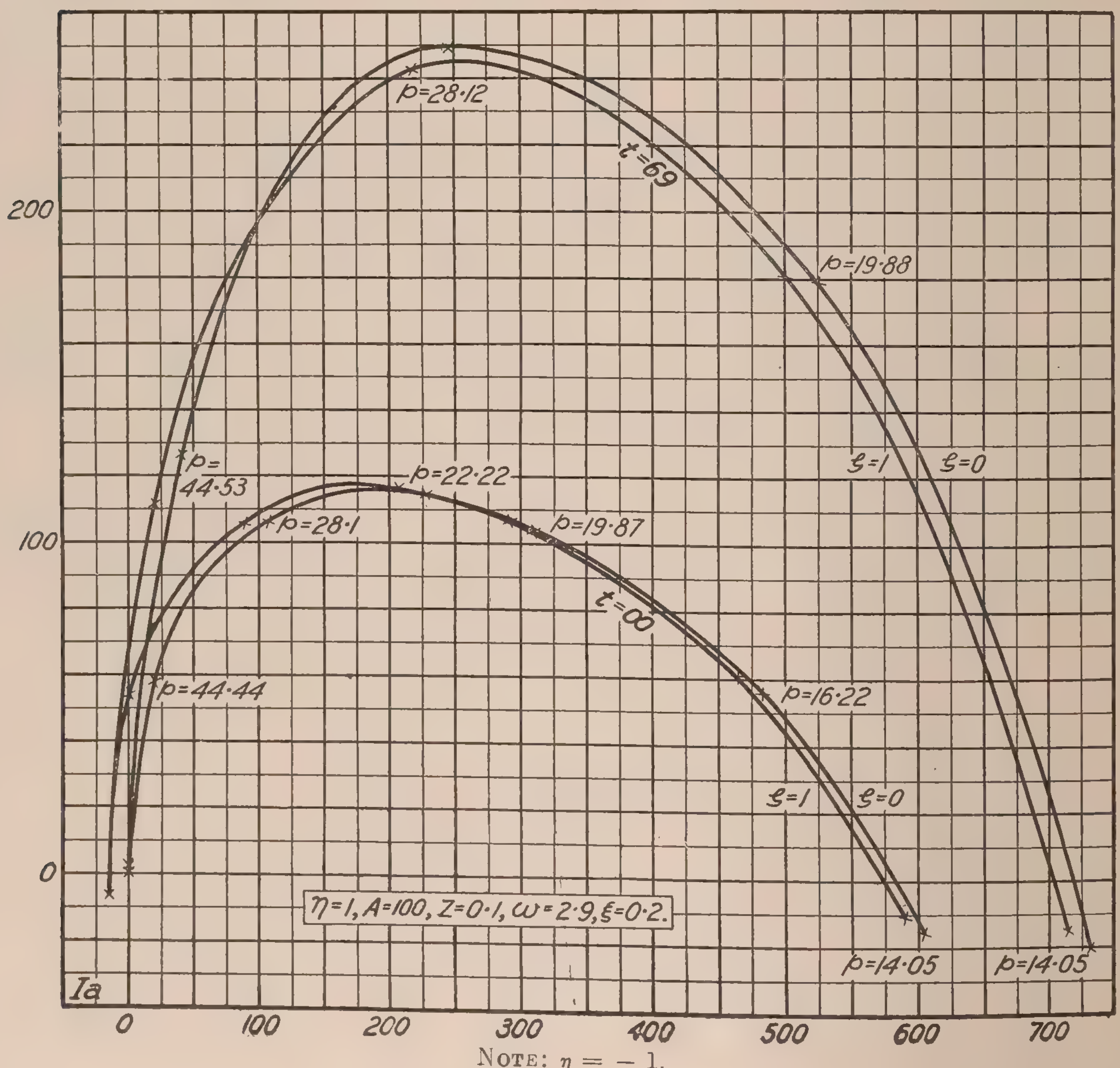
t=∞
p=19·87
p=28·1
p=16·22
p=14·05
Ia
100
0
0
100
200
300
400
500
600
700
800
900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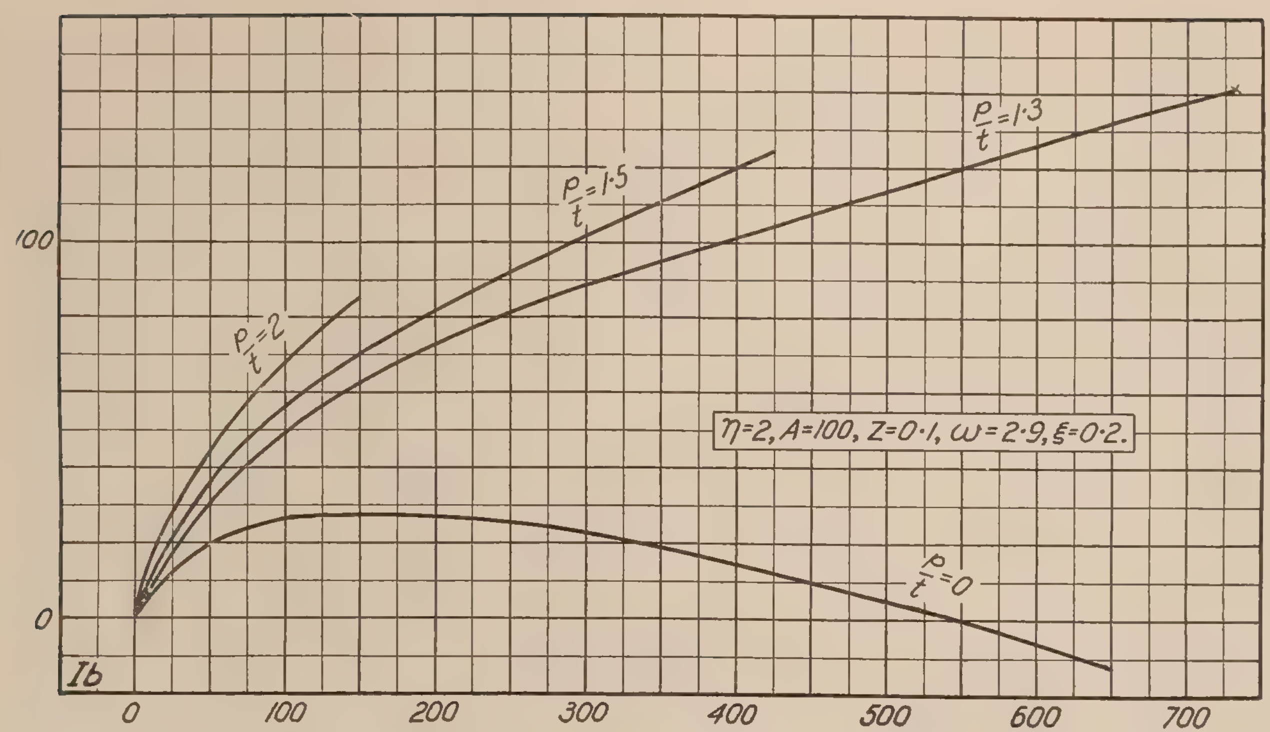
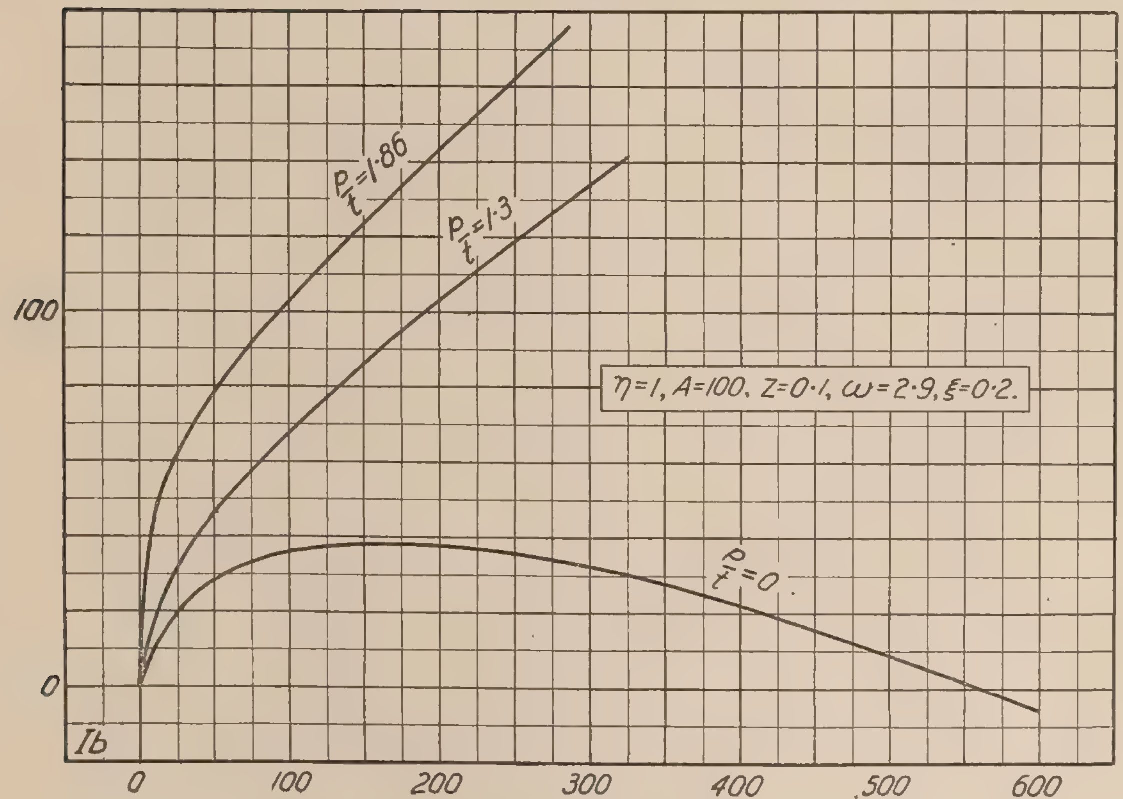
t=69
p=19·88
p=
44·53
ς=1
ς=0
p=22·22
p=28·1
p=19·87
t=∞
p=44·44
p=16·22
η=1, A=100, Z=0·1, ω=2·9, ξ=0·2.
p=14·05
p=14·05
Ia
200
100
0
0
100
200
300
400
500
600
700

NOTE: $\eta = -1$.


P/t = 1·3
P/t = 1·5
P/t = 2
η=2, A=100, Z=0·1, ω=2·9, ξ=0·2.
P/t = 0
100
0
Ib
0
100
200
300
400
500
600
700


P/t = 1·86
P/t = 1·3
η=1, A=100. Z=0·1, ω=2·9, ξ=0·2.
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100
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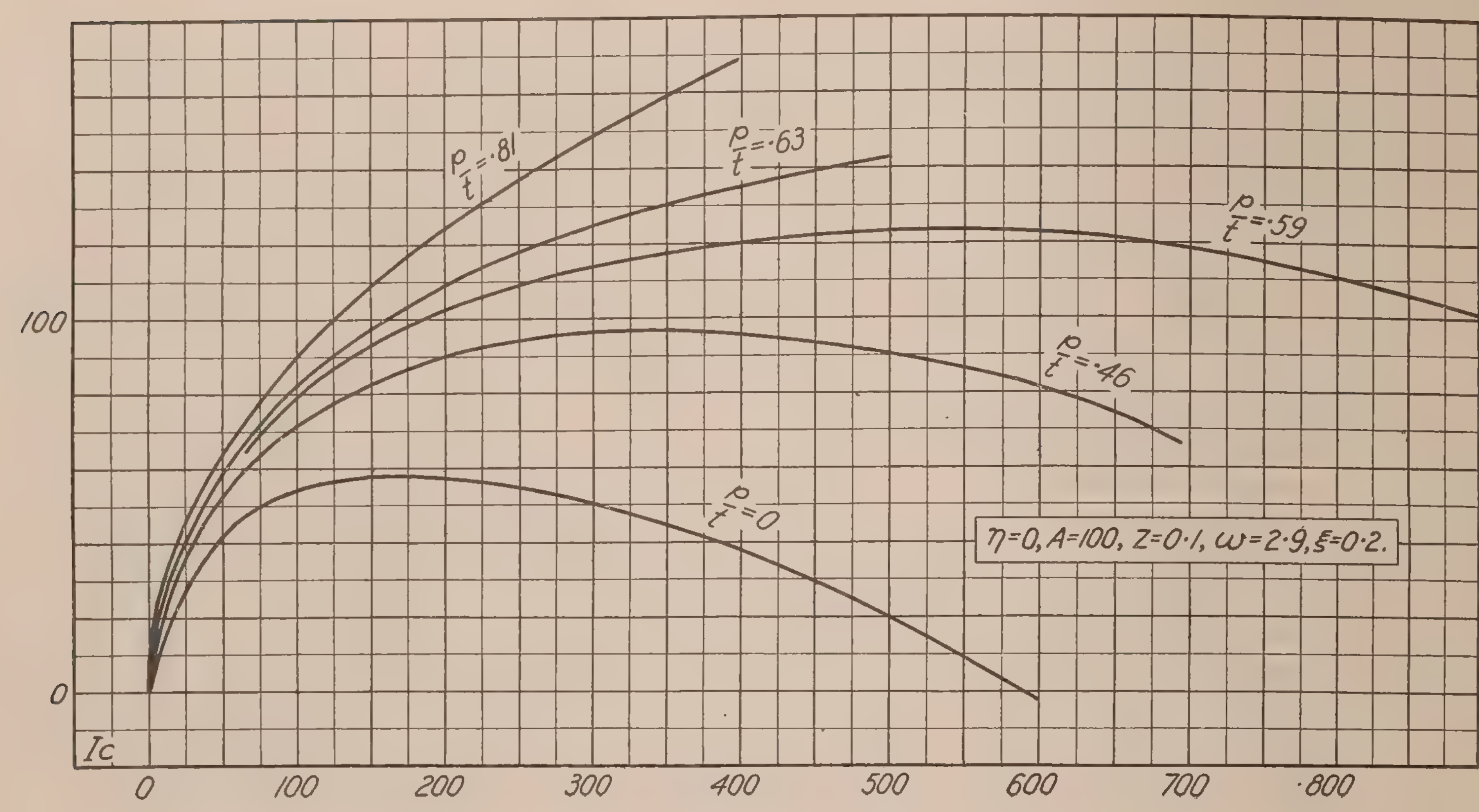
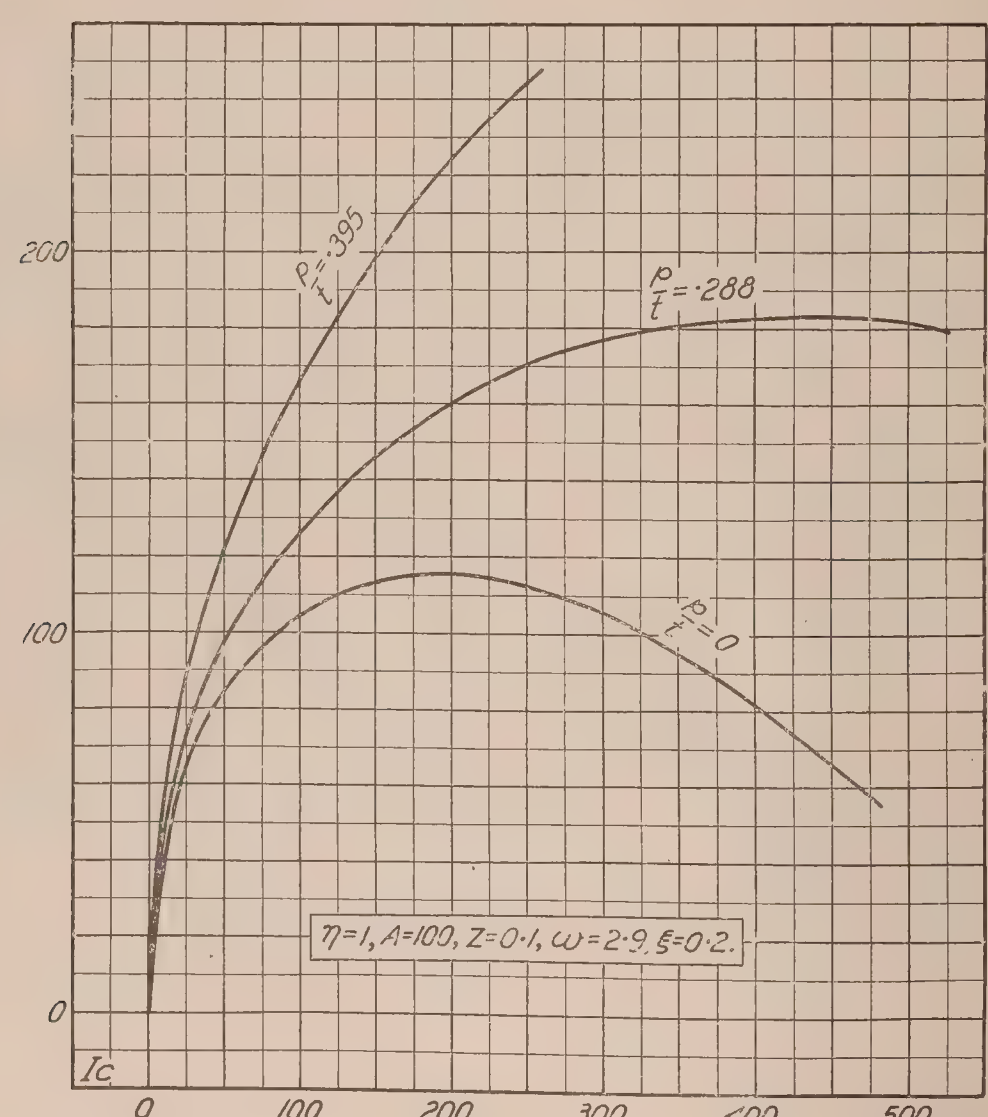
200
300
400
500
600



p/t = ·81
p/t = ·63
p/t = ·59
p/t = ·46
p/t = 0
η=0, A=100, Z=0·1, ω=2·9, ξ=0·2.
100
0
Ic
0
100
200
300
400
500
600
700
·800



p/t = ·395
p/t = ·288
p/t = 0
η=1, A=100, Z=0·1, ω=2·9, ξ=0·2.
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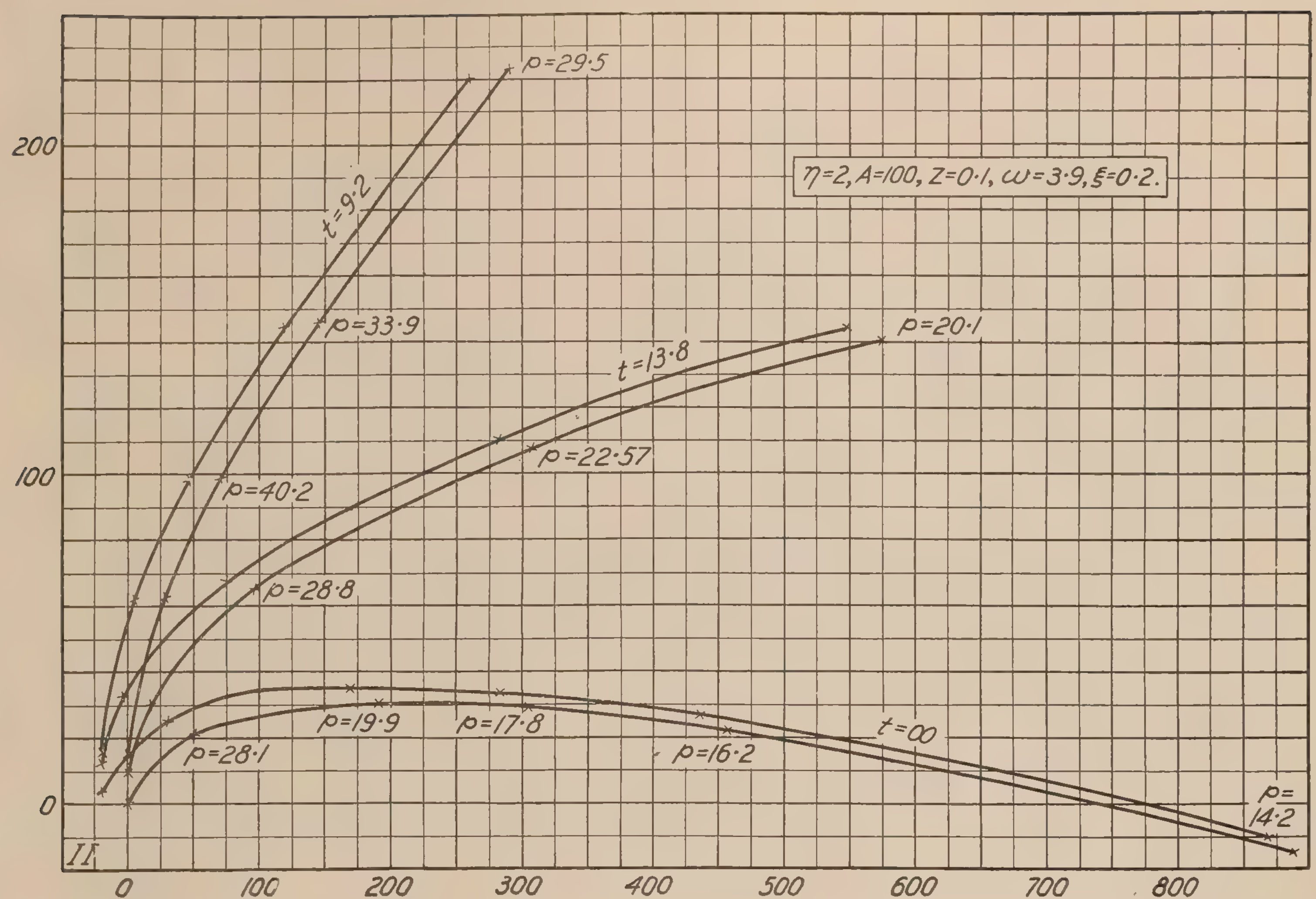
100
0
Ic
0
100
200
300
400
500

NOTE: $\eta = -1$.



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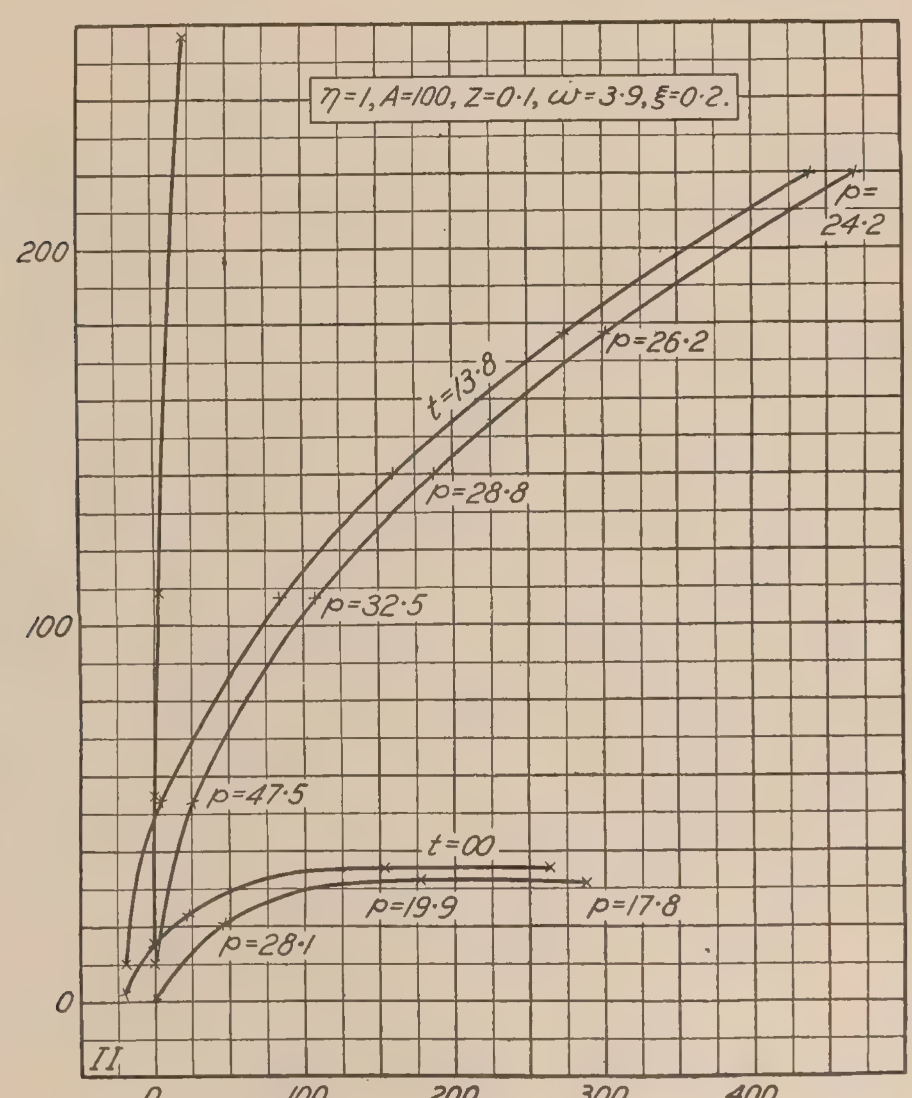
p=29·5
t=9·2
p=33·9
p=40·2
p=20·1
t=13·8
p=22·57
p=28·8
p=19·9
p=17·8
p=28·1
p=16·2
t=∞
p=
14·2
II
200
100
0
0
100
200
300
400
500
600
700
800



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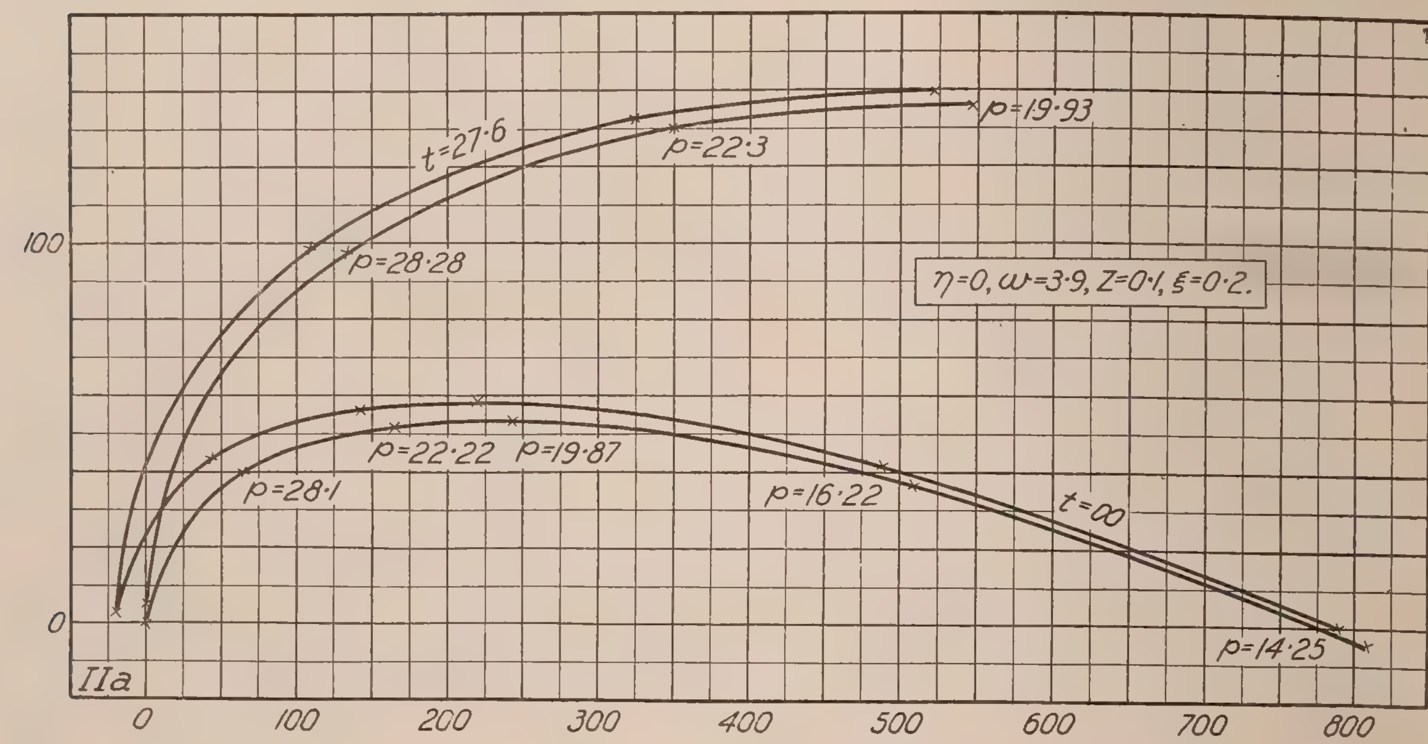
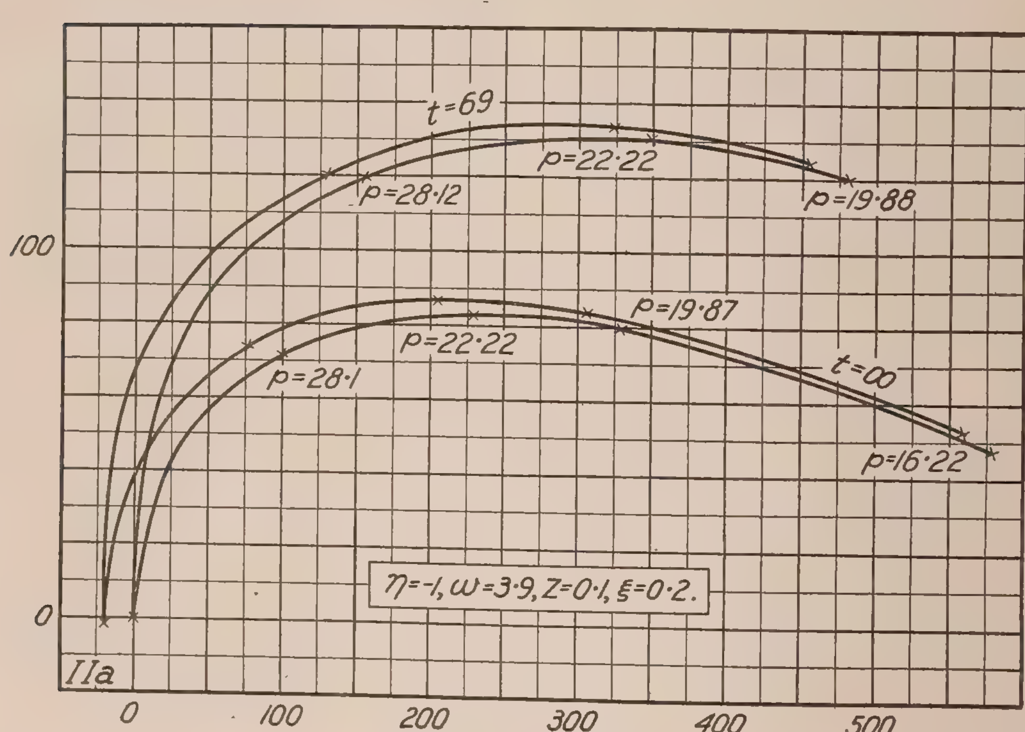
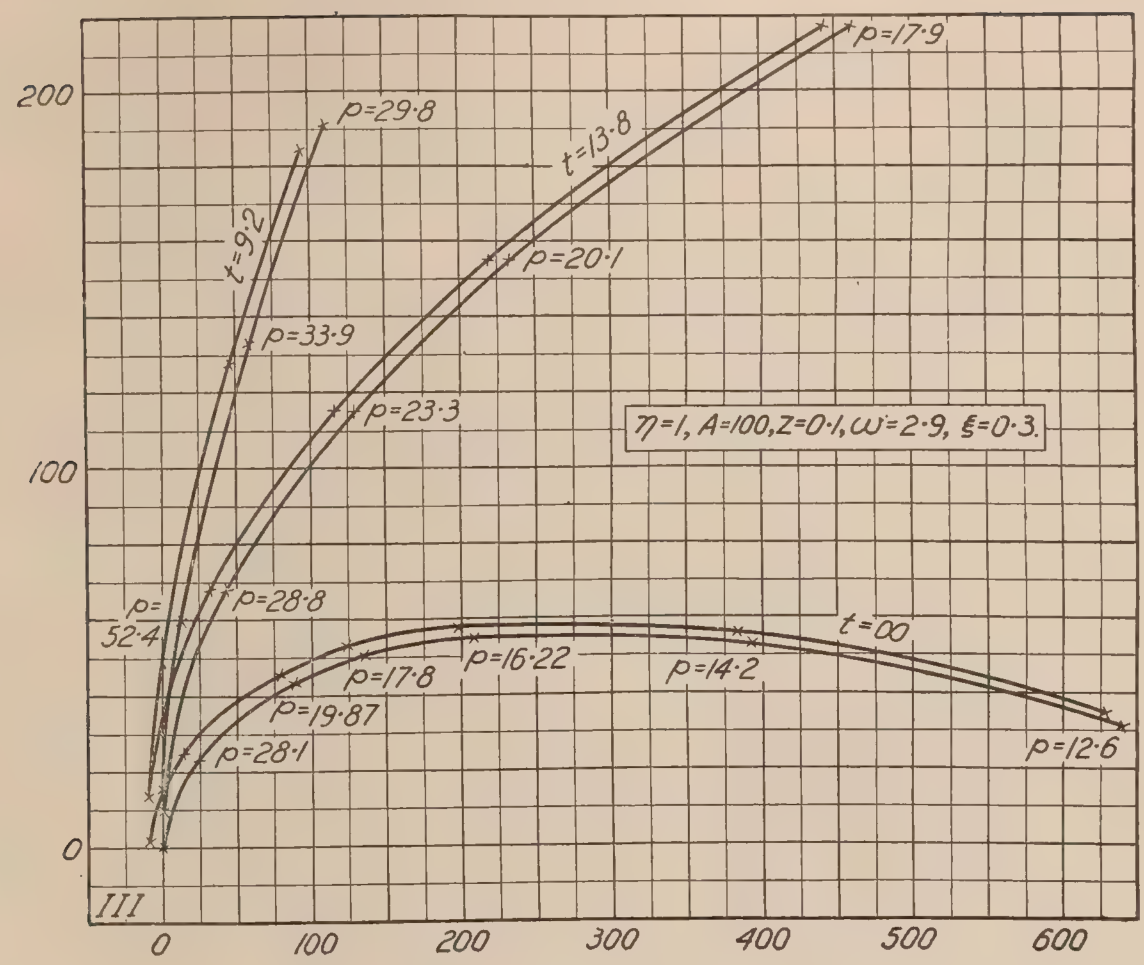
24·2
p=26·2
t=13·8
p=28·8
p=32·5
p=47·5
t=∞
p=19·9
p=17·8
p=28·1
II
200
100
0
0
100
200
300
400


t=27·6
p=19·93
p=22·3
p=28·28
η=0, ω=3·9, Z=0·1, ξ=0·2.
p=22·22
p=19·87
p=28·1
p=16·22
t=∞
p=14·25
IIa
100
0
0
100
200
300
400
500
600
700
800


t=69
p=22·22
p=28·12
p=19·88
p=19·87
p=22·22
p=28·1
t=∞
p=16·22
η=-1, ω=3·9, Z=0·1, ξ=0·2.
IIa
100
0
0
100
200
300
400
500


200
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III
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100
200
300
400
500
600
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p=17·9
t=13·8
t=9·2
p=20·1
p=33·9
p=23·3
η=1, A=100, Z=0·1, ω=2·9, ξ=0·3.
p=
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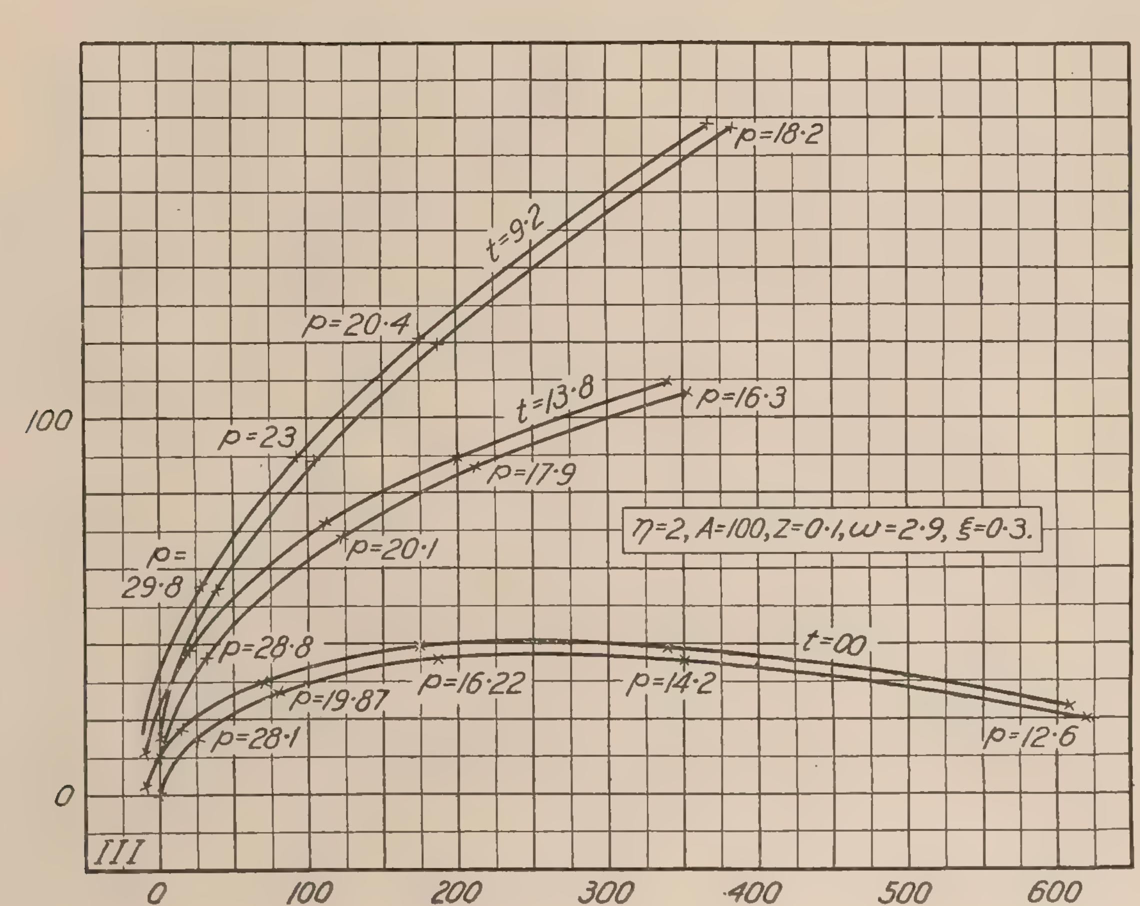
p=28·8
t=∞
p=16·22
p=17·8
p=14·2
p=19·87
p=28·1
p=12·6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III
0
100
200
300
400
500
600
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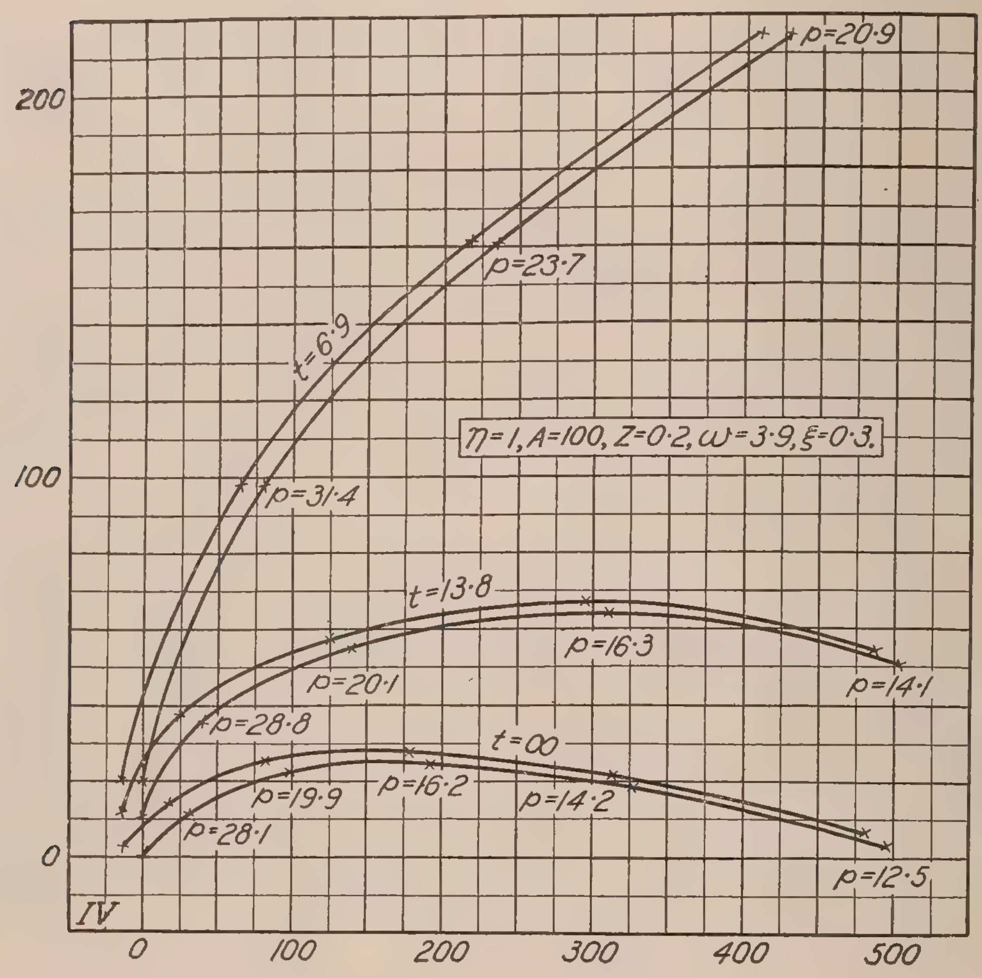
t=9·2
p=20·4
t=13·8
p=16·3
p=23
p=17·9
η=2, A=100, Z=0·1, ω=2·9, ξ=0·3.
p=
29·8
p=20·1
p=28·8
t=∞
p=16·22
p=14·2
p=19·87
p=28·1
p=12·6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t=00
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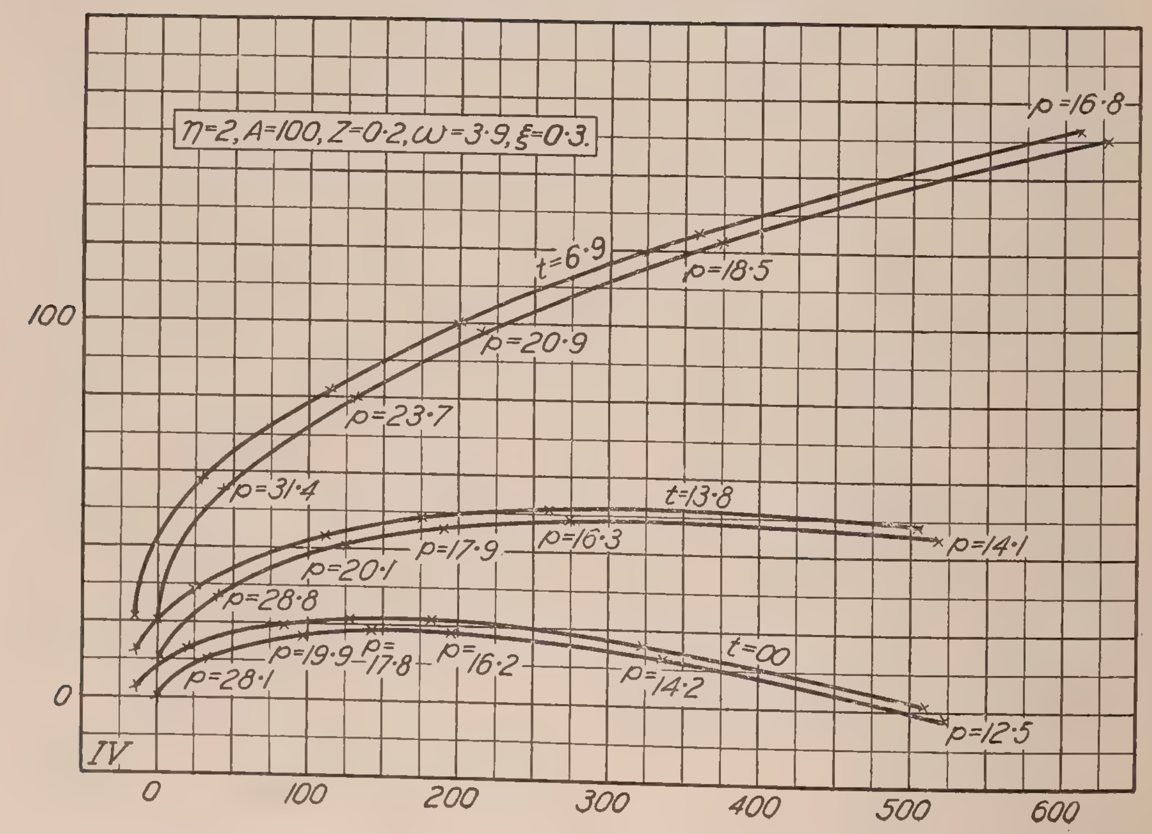
p=31·4
p=16·3
p=20·1
p=14·1
p=28·8
p=19·9
p=16·2
p=14·2
p=28·1
p=12·5
IV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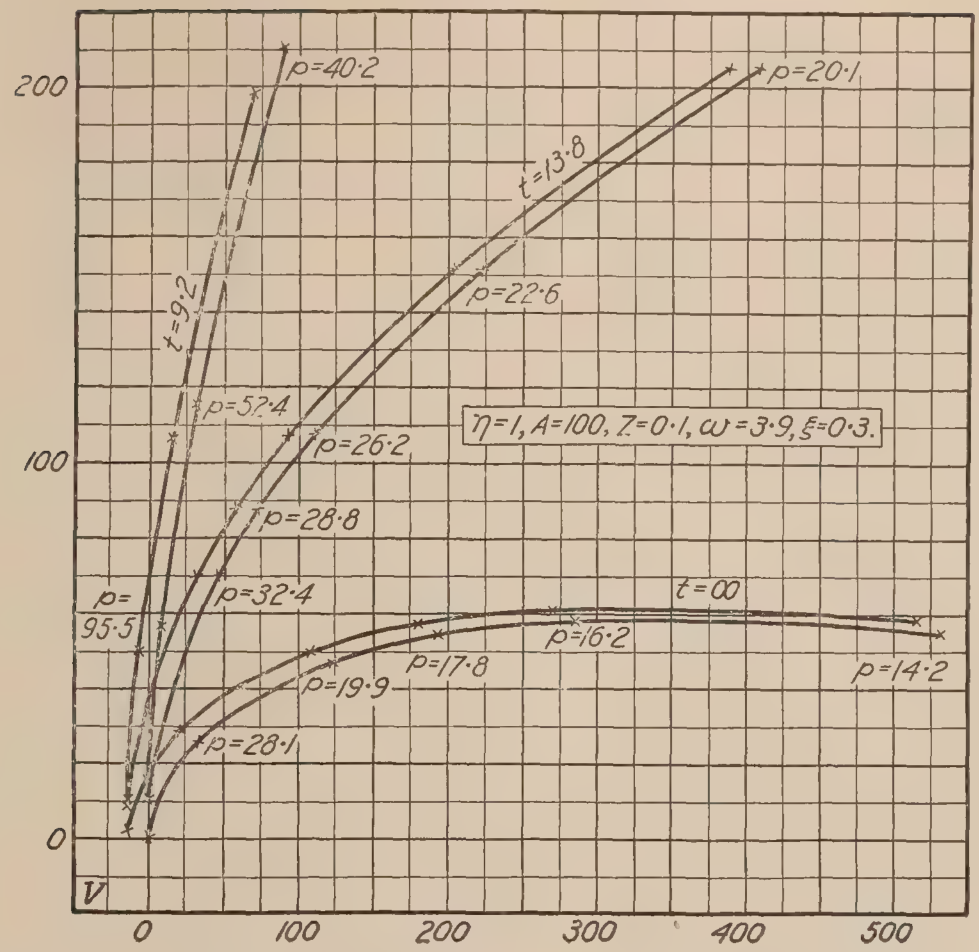
p=23·7
p=31·4
t=13·8
p=17·9
p=16·3
p=14·1
p=20·1
p=28·8
p=19·9
p=17·8
p=16·2
t=00
p=28·1
p=14·2
p=12·5
IV



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p=20·1
t=13·8
t=9·2
p=22·6
p=52·4
p=26·2
η=1, A=100, Z=0·1, ω=3·9, ξ=0·3.
p=28·8
p=32·4
p=95·5
t=∞
p=16·2
p=17·8
p=19·9
p=14·2
p=28·1
200
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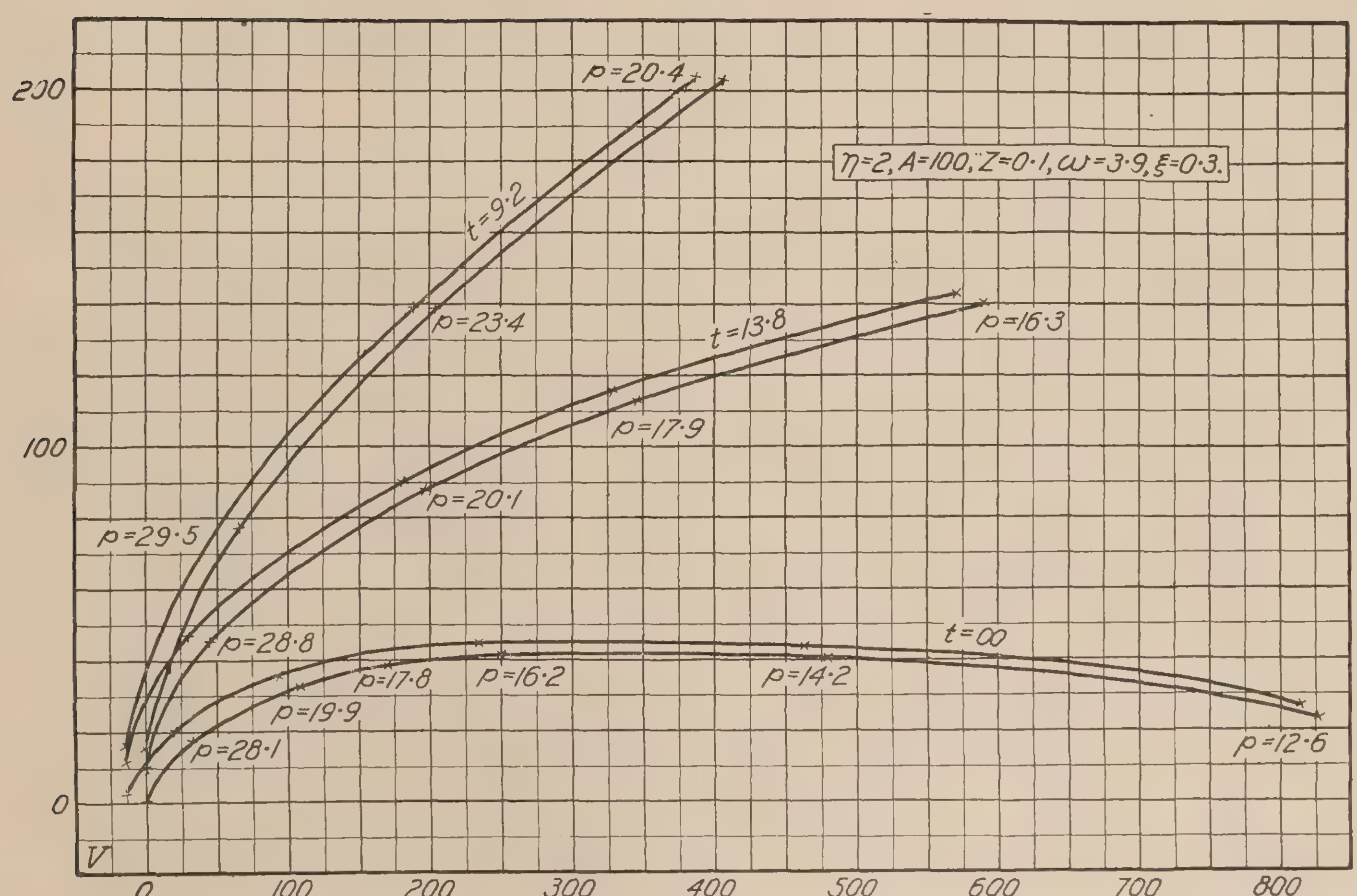
100
200
300
400
500



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η=2, A=100, Z=0·1, ω=3·9, ξ=0·3.
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t=13·8
p=16·3
p=17·9
p=20·1
p=29·5
p=28·8
t=∞
p=17·8
p=16·2
p=14·2
p=19·9
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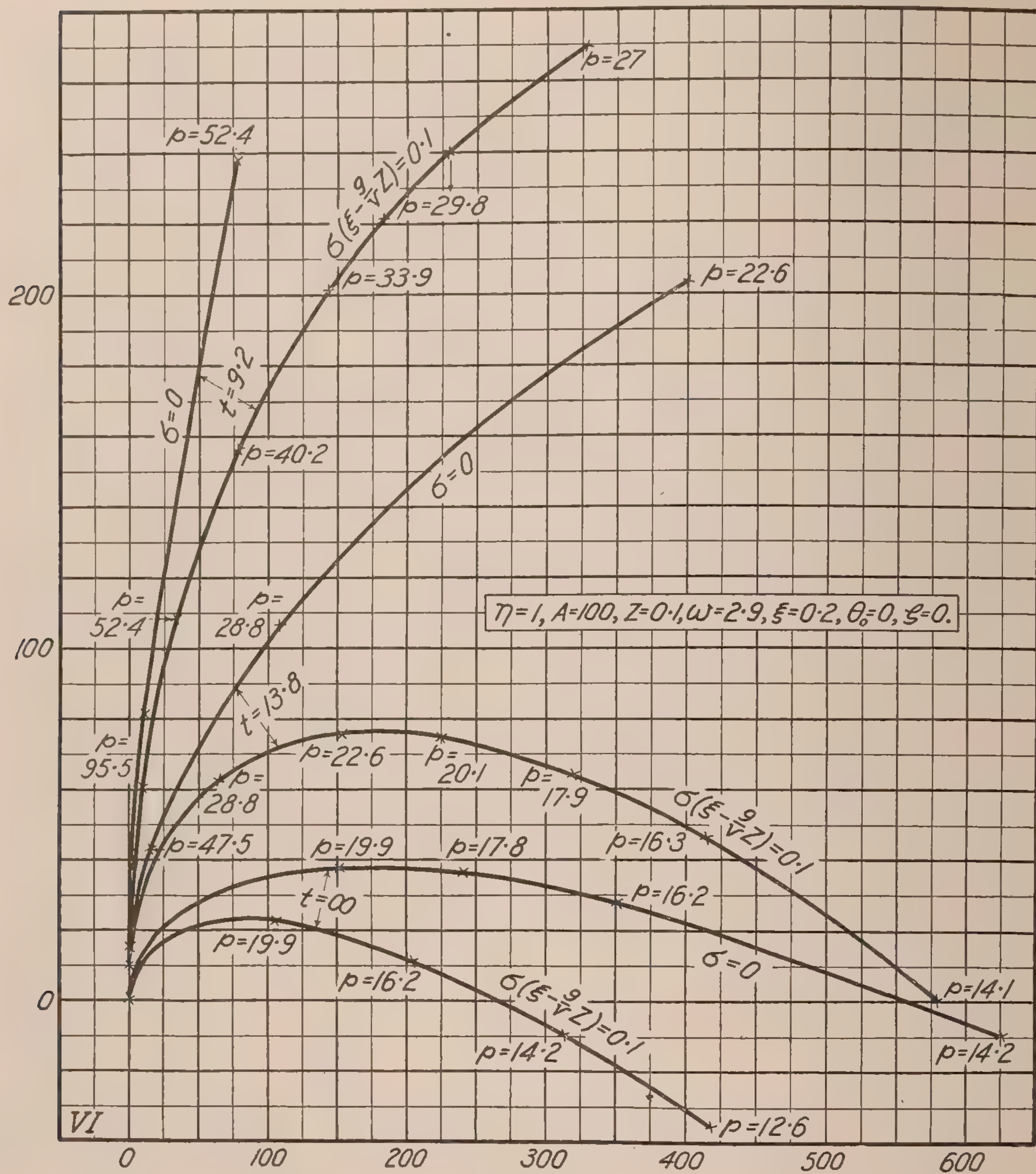
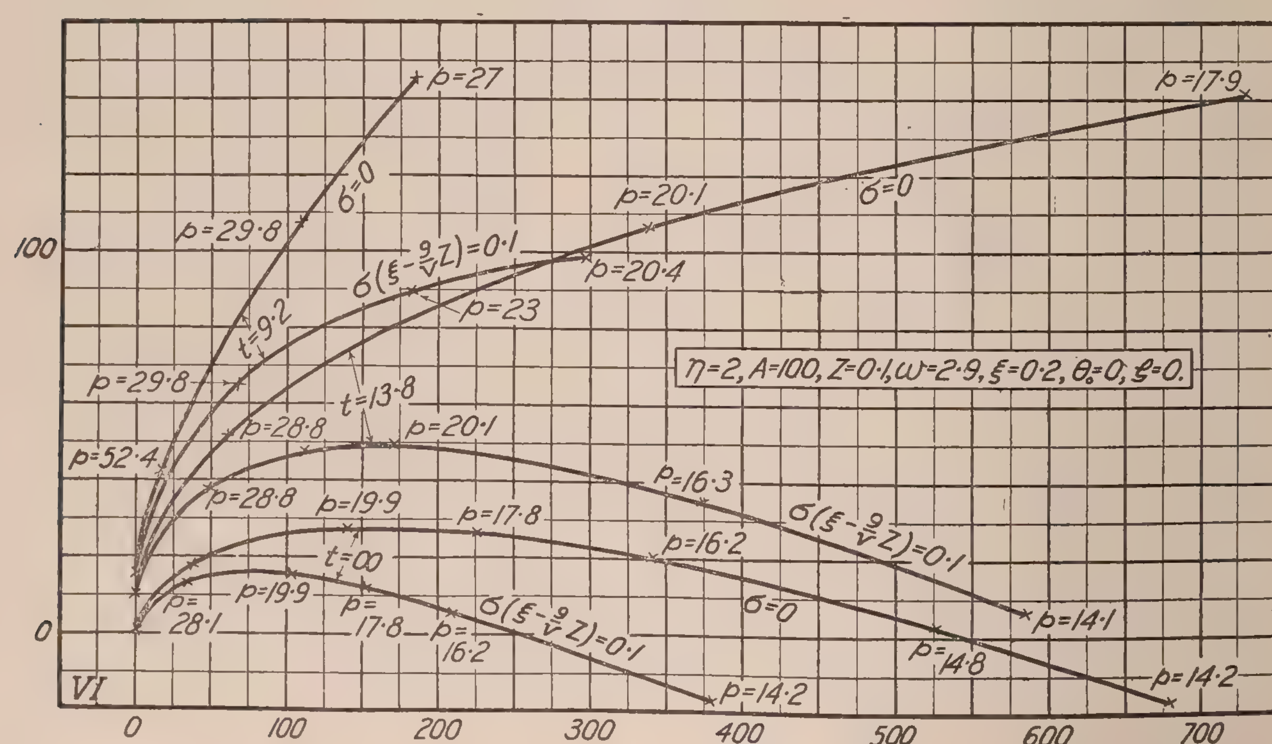
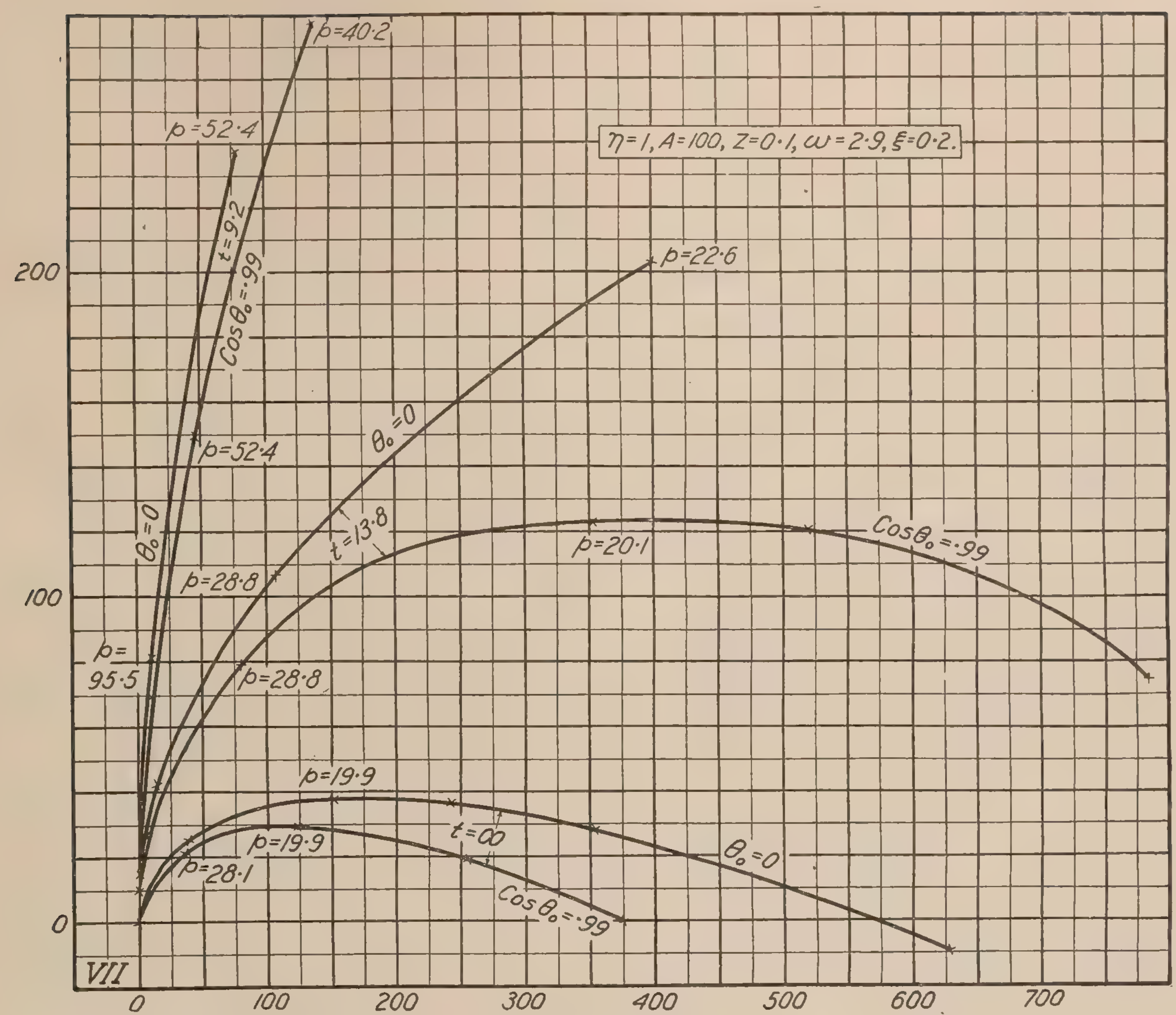
p=12·6
200
100
0
V
0
100
200
300
400
500
600
700
800





NOTE: σ is written in place of δ in this diagram.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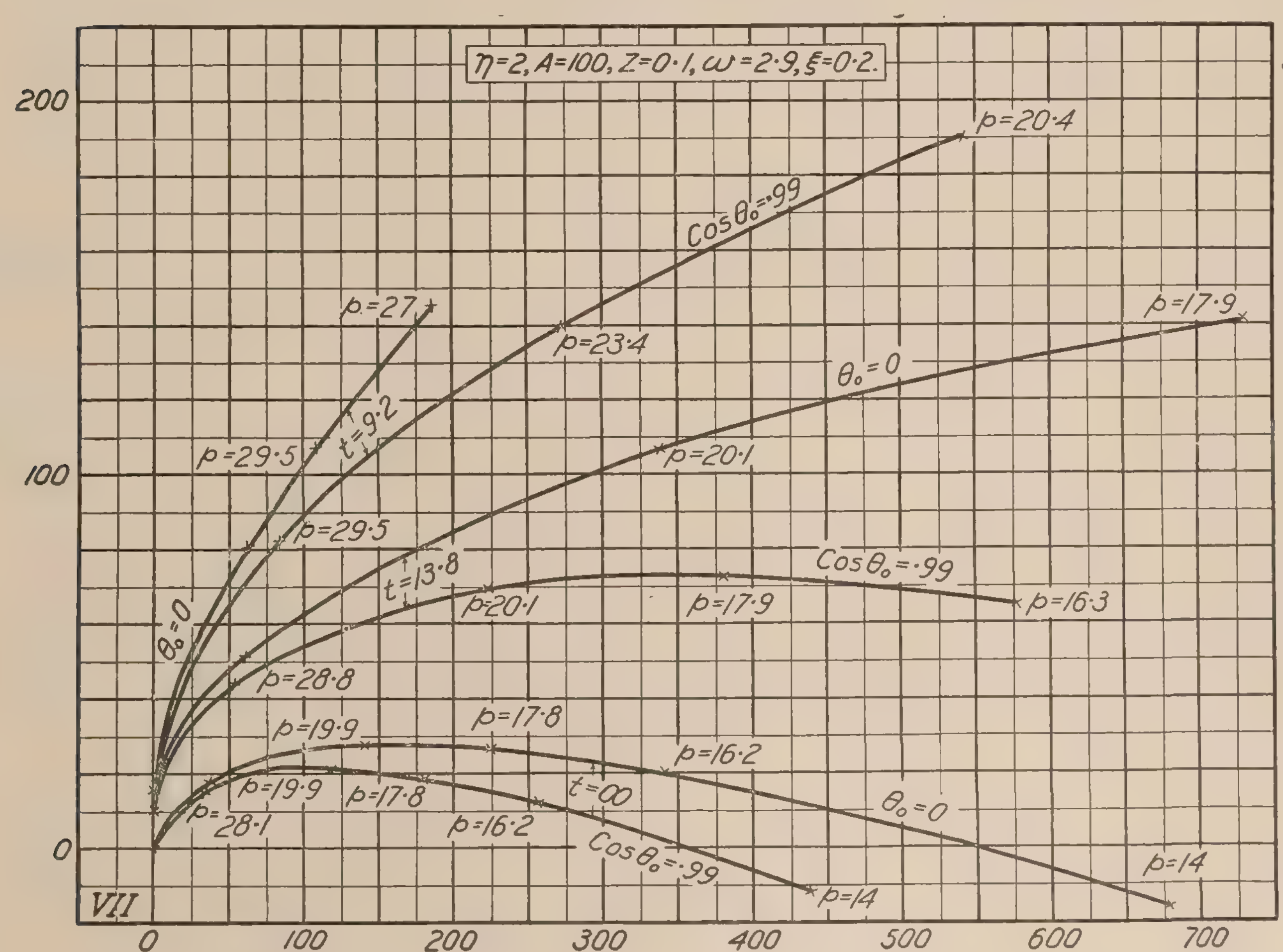
p=52·4
t=9·2
Cos θ₀=·99
p=22·6
θ₀=0
p=52·4
θ₀=0
t=13·8
p=28·8
p=20·1
Cos θ₀=·99
p=
95·5
p=28·8
p=19·9
p=19·9
t=∞
θ₀=0
p=28·1
Cos θ₀=·99
VII
200
100
0
0
100
200
300
400
500
600
700


η=2, A=100, Z=0·1, ω=2·9, ξ=0·2.
p=20·4
Cos θ₀=·99
p=27
p=23·4
p=17·9
θ₀=0
p=29·5
t=9·2
p=20·1
p=29·5
Cos θ₀=·99
t=13·8
p=20·1
p=17·9
p=16·3
θ₀=0
p=28·8
p=17·8
p=19·9
p=16·2
p=19·9
t=∞
p=17·8
θ₀=0
p=28·1
p=16·2
Cos θ₀=·99
p=14
p=14
VII
200
100
0
0
100
200
300
400
500
600
700

Lateral oscillations.—When the lateral oscillations are analytically independent of the pitching oscillations the biquadratic equation which determines the temporal characteristics of the oscillations is

$$\lambda(\lambda+x)\ (\lambda+w)\ (\lambda+\eta)+\zeta\lambda(\lambda+x)-\xi y\lambda(\lambda+w)+z(\lambda+\eta)-\xi z\lambda-y\zeta=o \qquad (1)$$

where [7]

$$\begin{aligned} L_p&=-xK_A^2, & L_r&=-\frac{U}{g}yK_A^2, & L_v&=\frac{z}{g}K_A^2 \\ N_p&=-\frac{g}{U}\xi K_C^2, & N_r&=-\eta K_C^2, & N_v&=\frac{\zeta}{U}K_C^2, \\ Y_p&=o, & Y_r&=o, & Y_v&=-w. \end{aligned} \qquad (2)$$

Writing $\lambda^2-2\alpha\lambda+\theta=o$ and reducing the above equation to a linear one in λ we find, on equating to zero the coefficients of λ and 1, that

$$\begin{gathered} \theta(w+\eta+x+4\alpha)=(x+2\alpha)\ (w+2\alpha)\ (\eta+2\alpha)+\zeta(x+2\alpha)-\xi y(w+2\alpha)+z(1-\xi), \\ \theta^2-\theta[(x+2\alpha)\ (w+\eta+2\alpha)+w\eta+\zeta-\xi y]+z\eta-y\zeta=o. \end{gathered} \qquad (3)$$

With the aid of these equations the curves $t=$constant may be drawn in the (x, y) plane for various sets of values of z, ζ, η, and w. It will be seen from the diagrams that there is a marked difference between the curves for which ξ is positive and those for which ξ is zero or negative.

It has been pointed out recently by Prof. E. B. Wilson [8] that in normal flight at a fairly low angle of incidence the quantity N_p is negative, and that consequently there is no need for a fine adjustment of the values of N_v, L_v, N_p, and L_p in order to secure both spiral stability and stability in the Dutch roll. At larger angles of incidence N_p may be zero and even positive, and then there is more need for a fine adjustment. On this account most of the diagrams have been drawn for zero and positive values of ξ, one diagram being deemed sufficient to indicate the general arrangement of the curves when ξ is negative. It will be noticed that when ξ is negative the period p is rather short but the time of damping t can also be made short and so the short period is not a great disadvantage. By making ξ positive the period can be more than doubled, but this is offset by the lengthening of the time of damping.

It should be noticed that in the diagrams the region of stability is bounded by the curve $t=\infty$ and the line $y=\frac{z\eta}{\zeta}$. If the representative point lies to the left of the curve $t=\infty$, the airplane is unstable in the Dutch roll; while if the point lies above the line $y=\frac{z\eta}{\zeta}$, the airplane is spirally unstable.[9] The shape of the curves $t=$constant is very much the same for the different values of η, ζ and z used in the diagrams, but the position of the line $y=\frac{z\eta}{\zeta}$ varies considerably.

An examination of diagrams VIII–XII indicates that it should be possible to construct an airplane which, when flying at a large angle of incidence, is spirally stable and has a period for the Dutch roll of about 15 seconds with a fairly short time of damping.

It has been remarked that in a moderately stable airplane the period is not sensitive, comparatively large changes in the airplane having small effect on the period and damping.[10] As an illustration of this phenomenon it is of some interest to consider the lines along which both p and t are constant. Such a line is obtained by choosing α and θ so that the two equations (3) are the same. Writing down the conditions

$$\frac{(w+2\alpha)(\eta+2\alpha)+\zeta-\theta}{-\theta(w+\eta+2\alpha)}=\frac{-\xi(w+2\alpha)}{\theta\xi-\zeta}=\frac{z(1-\xi)-\theta(w+\eta+2\alpha)}{\theta^2-\theta(w\eta+\zeta)+z\eta},$$

[7] Cf. J. C. Hunsaker, Dynamical Stability of Aeroplanes, Smithsonian Miscellaneous Collections (Washington), Vol. 62, No. 5, June (1916), pp. 55–57.

[8] Fourth Annual Report of the National Advisory Committee of Aeronautics. No. 26. Washington (1919).

[9] For these terms see Hunsaker (loc. cit.).

[10] Cf. W. S. Farren, "Full Scale Aeroplane Experiments," The Aeronautical Journal, February, 1919, p. 56.

we have two equations to determine α and θ when z, w, ξ, η and ζ are given. If on the other hand ξ, ζ, w, α, and θ be regarded as given, the resulting equations will determine η and z uniquely. The equation for η is

$$-(\eta+2\alpha)(w+2\alpha)\zeta=\xi\theta^2+\theta[\xi w(w+2\alpha)-\xi\zeta-\zeta]+\zeta^2.$$

If $w+2\alpha$ is small, a comparatively small percentage change in θ means a comparatively large percentage change in the value of η. For example if $w=0.1$, $\zeta=0.6$, $\xi=2$, $w+2\alpha=-.05$,

$$\theta=.27,\quad \eta=\ .72,\quad z=1.782$$

$$\theta=.26,\quad \eta=\ .97,\quad z=1.794$$

$$\theta=.25,\quad \eta=1.233,\quad z=1.777$$

In diagram VIII with $\xi=1$, the equation of the straight line along which p and t are constant is $.0884x+.0241y-.3843=o$; and we have $w+2\alpha=-.0241$, $\theta=.4953$, $t=11.12$, $p=8.96$.

It appears from the diagrams that the greater the curvature of the curve $t=$ constant, the more rapid is the variation of p as the representative point moves along it. If the curve is concave to the axis of y, p increases as the point moves up the curve, while, if the curve is convex to the axis of y, p increases as the point moves down the curve.

If we wish an airplane to have resistance derivatives for which the period is not sensitive, a good plan is to bring the representative point as close as possible to the straight line along which p and t are constant. When ξ is negative this line does not seem to appear on the diagram.

A study of diagrams VIII–XII reveals the following properties:

(1) The greater the value of ξ the greater seems to be the effect on the damping of a change in w.

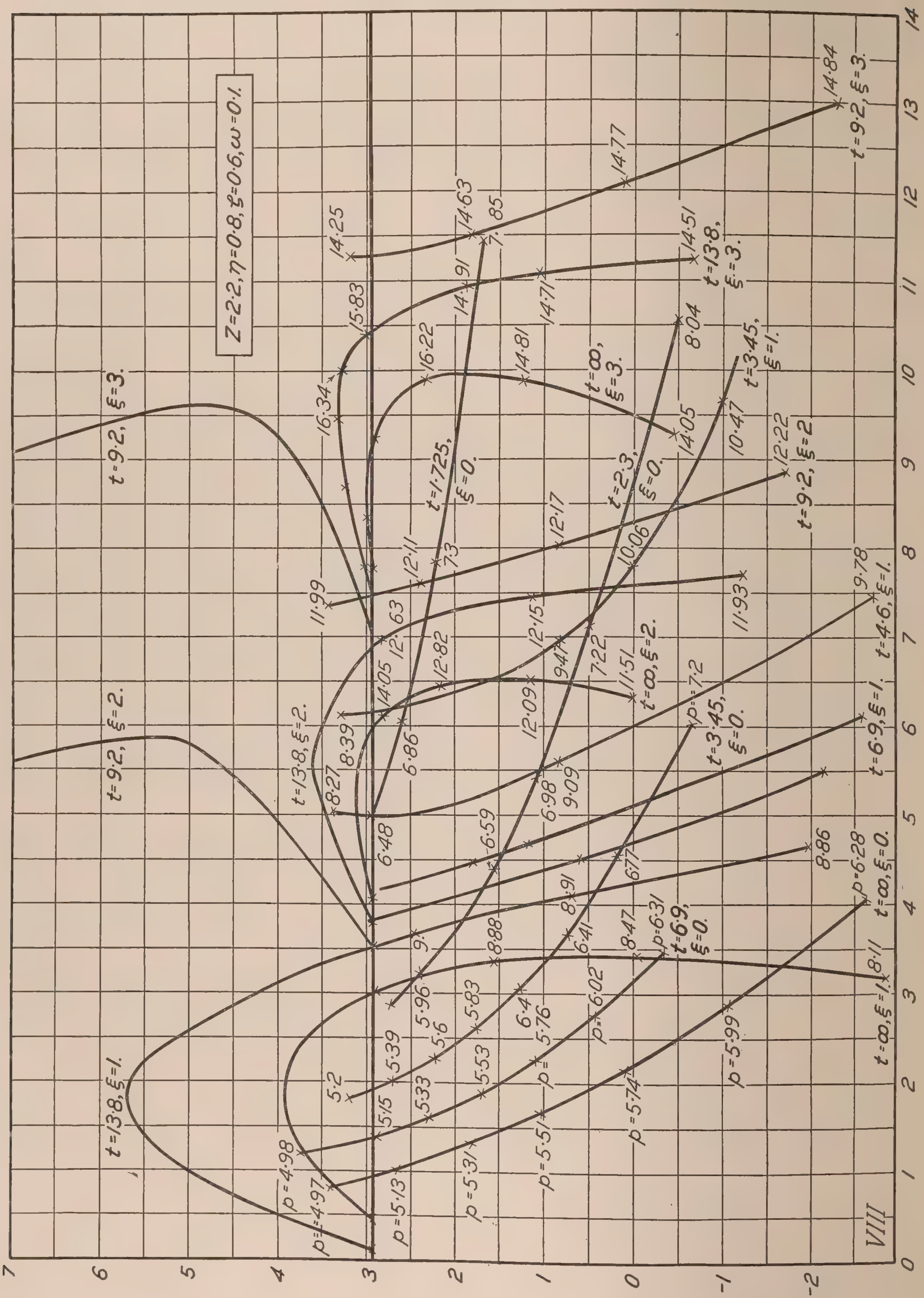
(2) An increase in w decreases the time of damping but does not greatly alter the period.

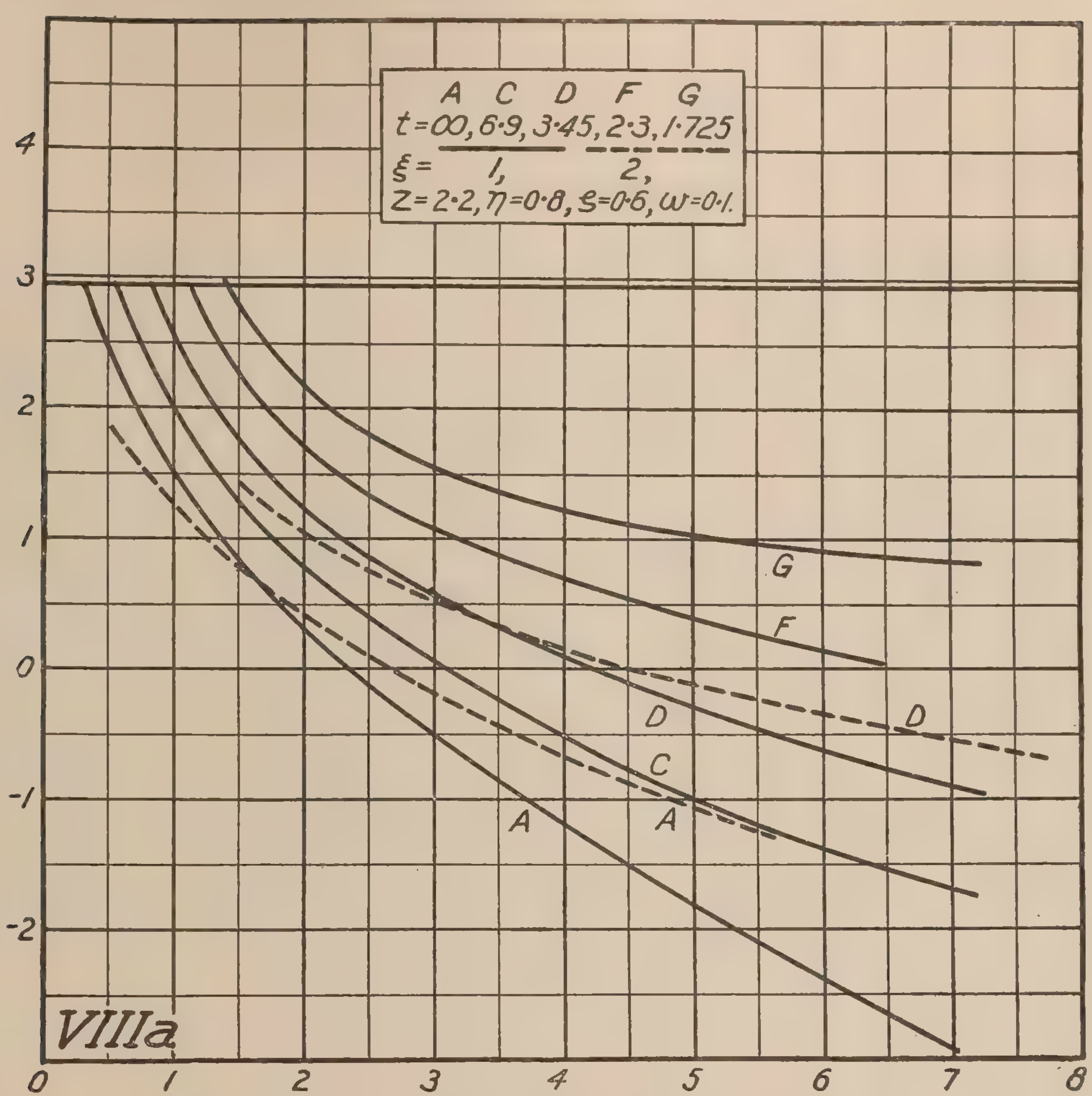
(3) An increase in η decreases the time of damping and increases the period when $\xi=o$; but, when $\xi=1$, 2, or 3, the effect seems to be reverse.

(4) An increase in z seems to widen the gaps between the curves $t=$constant and to greatly increase the period when $\xi=1$ and $\xi=2$.

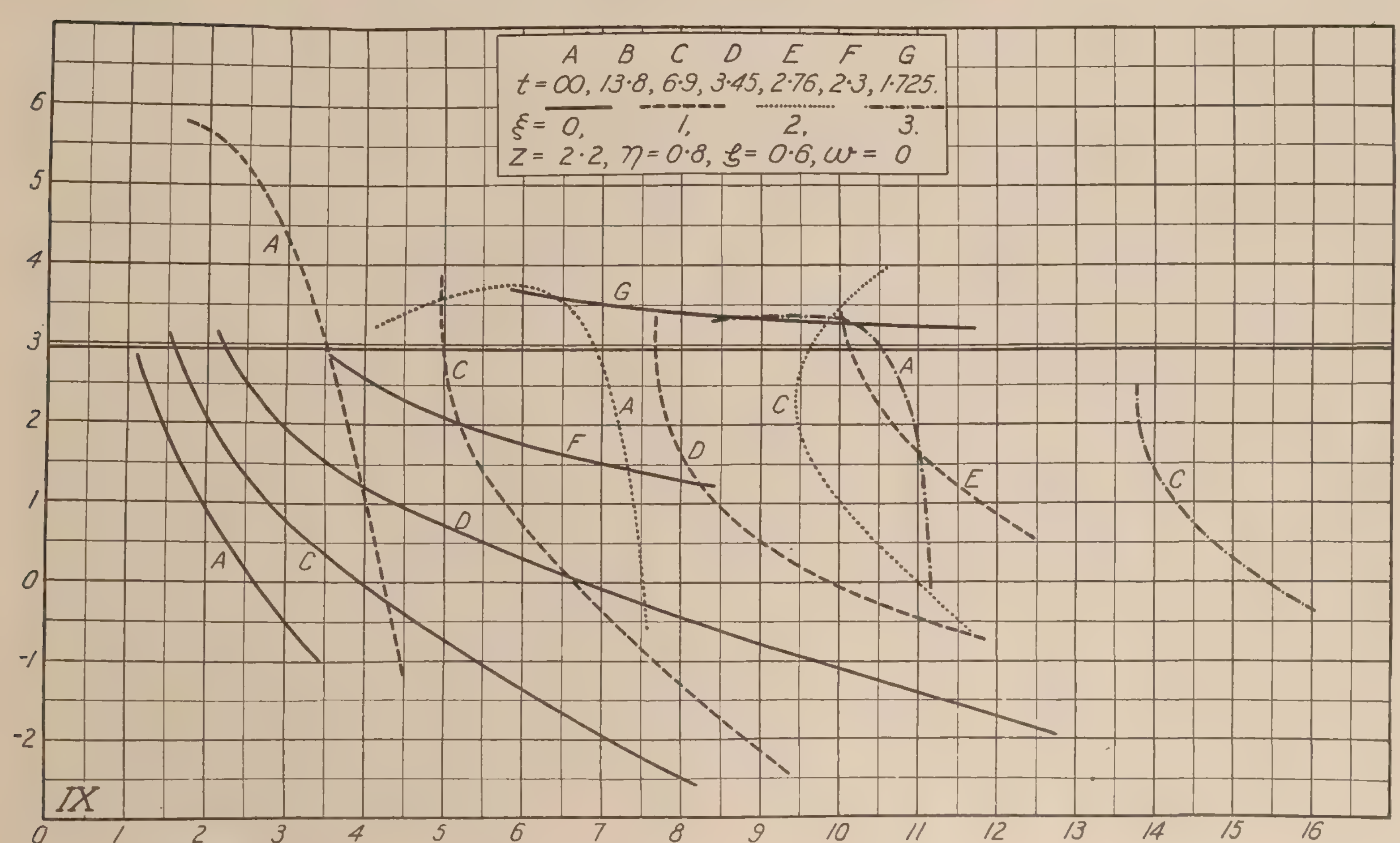
It will be noticed that in diagram XII the numbers on the curves $t=$constant indicate the value of $\frac{p}{t}$.

(5) The chief effect of a decrease in ζ seems to be a slight change in curvature of the curves $t=$constant.


Z=2·2, η=0·8, ξ=0·6, ω=0·1.
VIII



NOTE: $\xi=-1$; -2.



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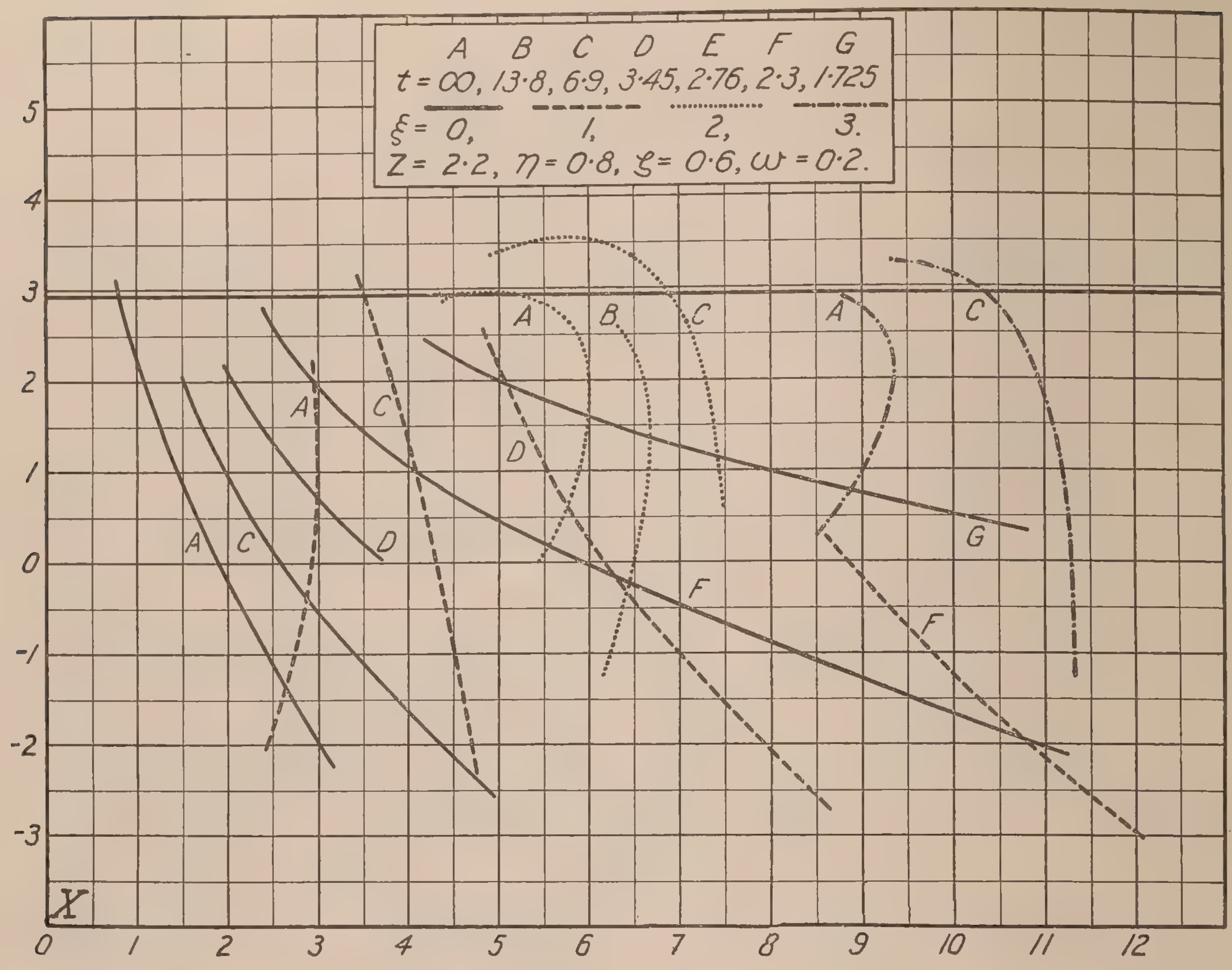
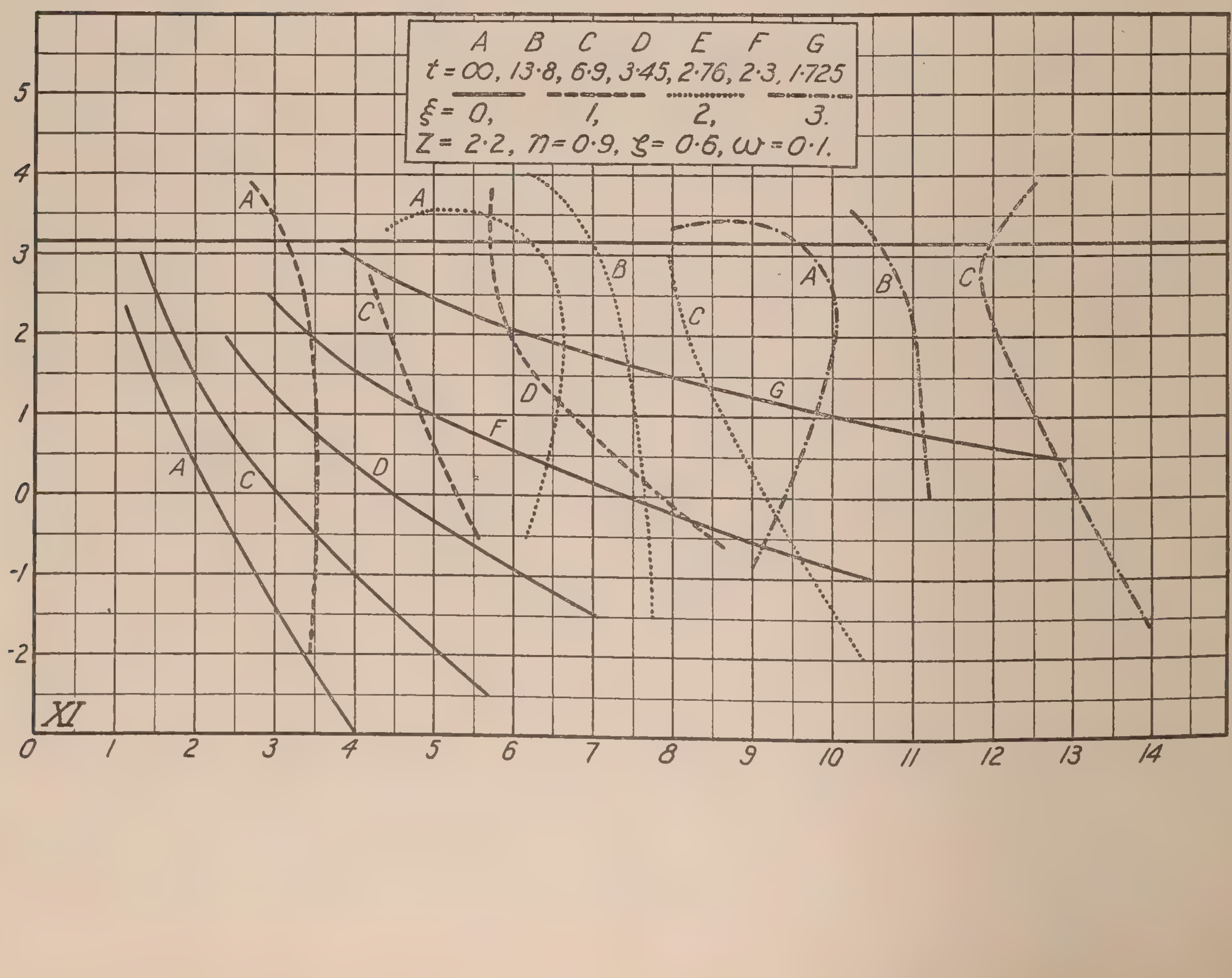
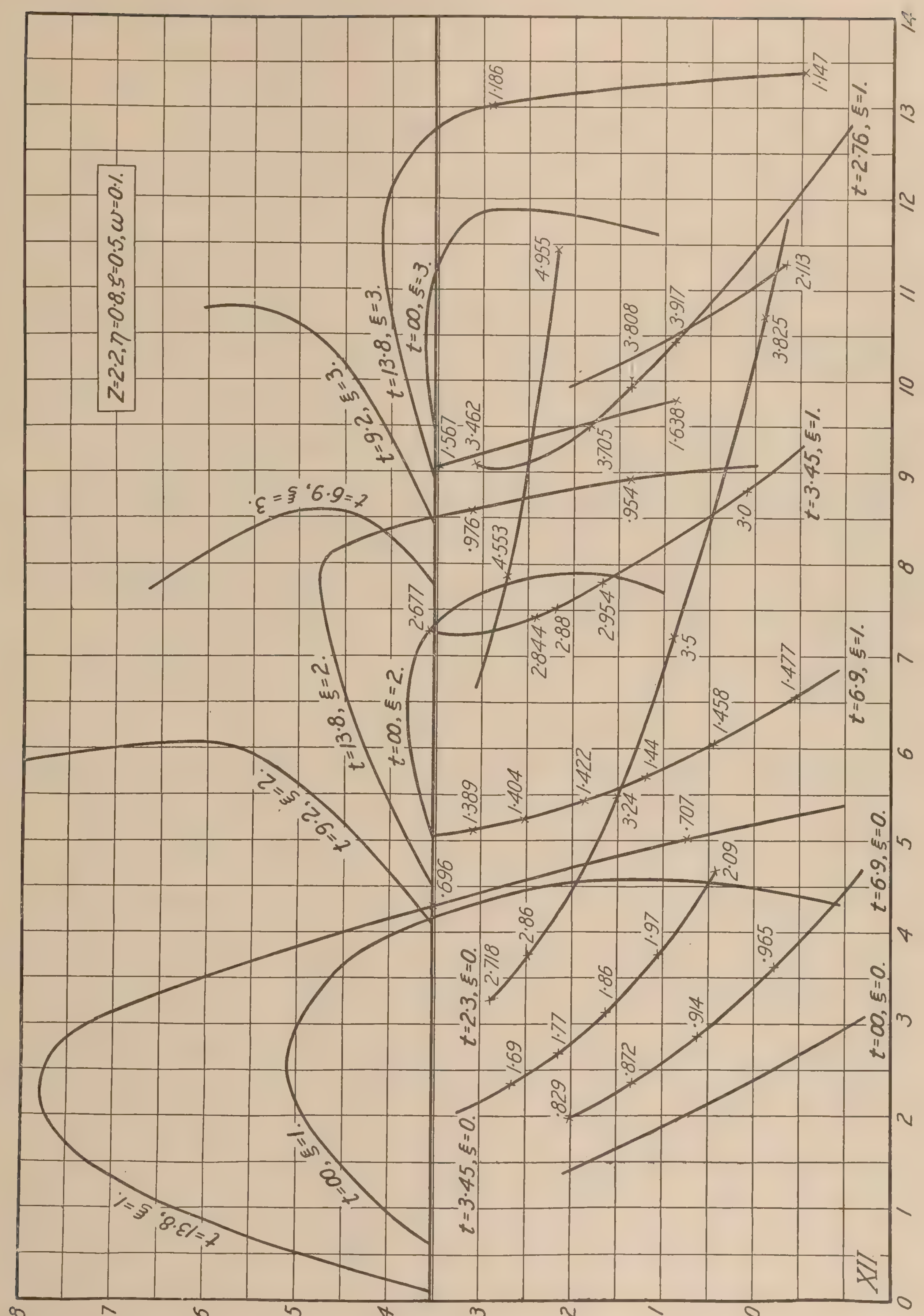

A B C D E F G
t = ∞, 13·8, 6·9, 3·45, 2·76, 2·3, 1·725
ξ = 0, 1, 2, 3.
Z = 2·2, η = 0·8, ζ = 0·6, ω = 0·2.
X


A B C D E F G
t = ∞, 13·8, 6·9, 3·45, 2·76, 2·3, 1·725
ξ = 0, 1, 2, 3.
Z = 2·2, η = 0·9, ζ = 0·6, ω = 0·1.
XI



Z=2·2, η=0·8, ξ=0·5, ω=0·1.
t=13·8, ξ=1.
t=00, ξ=1.
t=9·2, ξ=2.
t=13·8, ξ=2.
t=00, ξ=2.
t=6·9, ξ=3.
t=9·2, ξ=3.
t=13·8, ξ=3.
t=00, ξ=3.
t=3·45, ξ=0.
t=2·3, ξ=0.
t=00, ξ=0.
t=6·9, ξ=0.
t=6·9, ξ=1.
t=3·45, ξ=1.
t=2·76, ξ=1.
XII

APPENDIX TO REPORT NO. 80.

ABILITY OF THE PARACHUTE AND HELICOPTER.

Note I.

The expressions for H and K in terms of $x_1, x_2, x_3, y_1, y_2, y_3$ may be found as follows: Writing

$$f_1(y)=(y+y_1)(y+y_2)(y+y_3)-x_2x_3(y+y_1)-x_3x_1(y+y_2)-x_1x_2(y+y_3),$$

$$f_2(y)=x_1(y+y_2)(y+y_3)+x_2(y+y_3)(y+y_1)+x_3(y+y_1)(y+y_2)-x_1x_2x_3,$$

we find that

$$f_1(y)=\frac{x_1(y+y_1)+x_2(y+y_2)+x_3(y+y_3)}{(x_1+x_2+x_3)^2}f_2(y)-f_3(y),$$

where

$$(x_1+x_2+x_3)^2 f_3(y)=Ax_2x_3(y+y_1)+Bx_3x_1(y+y_2)+Cx_1x_2(y+y_3).$$

The coefficient of y on the right-hand side is the quantity denoted by H.

Again writing

$$f_2(y)=(ay+b)f_3(y)-K$$

where a, b, and K are constants to be determined, we may find the value of K by substituting in the last equation a value of y which makes $f_3(y)=0$. We thus find that

$$H(y+y_1)=x_1(Bn-Cm)=x_1P,\qquad H(y+y_2)=x_2(Cl-An)=x_2Q,\qquad H(y+y_3)=x_3(Am-Bl)=x_3R$$

$$H^2K=x_1x_2x_3[H^2-QR-RP-PQ].$$

Putting $f_3(y)=o$ in the identical relation connecting $f_1(y)$, $f_2(y)$, and $f_3(y)$ we also find that

$$H^2K(x_1^2P+x_2^2Q+x_3^2R)=[H^2(P+Q+R)-PQR]x_1x_2x_3.$$

Hence

$$H^2K(H+ix_1^2P+ix_2^2Q+ix_3^2R)=x_1x_2x_3(H+iP)(H+iQ)(H+iR)$$

or

$$H^2K\left[H^2+(x_1^2P+x_2^2Q+x_3^2R)^2\right]^{\frac{1}{2}}=x_1x_2x_3(H^2+P^2)^{\frac{1}{2}}(H^2+Q^2)^{\frac{1}{2}}(H^2+R^2)^{\frac{1}{2}}.$$

In this relation the positive signs must be taken for the square roots so that K is positive when x_1, x_2, and x_3 are positive.

Note II.

In the case of a helicopter rising or falling vertically it may be sufficient to take into account the fin action and gyroscopic action of the lifting screw.

Writing

$$Q=o,\ E=X_u,\ F=-iX_r,\ J=iN_u,\ K=N_r+iN_p$$

the period equation becomes

$$\lambda(\lambda+u)(\lambda+y+iz)+x(\lambda w+i)=o$$

where

$$x=\frac{g^2N_u}{A},\ y=g\frac{N_r}{A},\ z=g\frac{N_p}{A},$$

$$u=g\frac{X_u}{W},\ w=\frac{V}{g}-\frac{X_r}{W}.$$

Routh's conditions indicate that for stability the quantities

$$p_1 = y + u,$$

$$H = uyz^2 + (y+u)^2(wx+uy) - x(y+u),$$

and

$$K = \frac{x}{H^2}\left[H^2 - Hyz^2u - (y+u)xy^2z^2\right]$$

must be positive.

On the other hand we find that

$$T = \frac{8x}{(y+u)^2}\left[H^2 - Hyz^2u - (y+u)xy^2z^2\right]$$

so that the conditions $K > o$ and $T > o$ are equivalent.

The effect of gyroscopic action on stability has been estimated for the case of an airplane in rectilinear flight and found to be small.[11] The value adopted for N_p was $N_p - I\Omega$ where I is the moment of inertia of the propeller about the axis of y and Ω its angular velocity about this axis.

With $I = 150$ pounds$-ft^2$ and $\Omega = 2\pi \times \frac{1200}{60} = 125.8$ radians per second, this gave a value of N_p of about $15m$ for an airplane of mass $m = 1{,}300$ pounds. In the case of the helicopter, Ω is smaller than for the airplane propeller but I is very much greater if the diameter of the lifting screw is large. It seems likely, then, that the gyroscopic effect on stability will be greater than in the case of the airplane.

[11] L. Bairstow, B. Melvill Jones, and A. W. H. Thompson, British Advisory Committee's Report. 1912-13, p. 166.

REPORT No. 96

STATICAL LONGITUDINAL STABILITY OF AIRPLANES

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REPORT No. 96.

STATICAL LONGITUDINAL STABILITY OF AIRPLANES.

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This report is essentially a continuation and extension of Report No. 70 of the National Advisory Committee for Aeronautics, entitled "Preliminary Report on Free-Flight Testing," the last part of which was devoted to an elementary discussion of the statical stability characteristics of the JN4H and the DH4. Since the completion of Report No. 70 a large amount of experimental work has been done on the JN4H by the committee's staff at Langley Field, in addition to a little on several other types, and the results are presented here, together with a detailed theoretical analysis of statical stability, of the factors which affect it, and of the methods which can be employed for its modification. Some of the results obtained have been discussed in technical Note No. 1 of the National Advisory Committee for Aeronautics, "Notes on Longitudinal Stability and Balance," portions of which are reprinted in this report.

As in the earlier report, stability will be considered under the two entirely distinct heads of stability with locked controls and stability with free controls. The first depends solely on the control position, and is much simpler to analyze and easier to secure than is the second, which depends on the forces or, more accurately, the moments acting on the movable portion of the control surface.

THEORY OF STABILITY WITH LOCKED CONTROLS.

An airplane which is stable with the elevator locked in position so that it forms in effect a part of the fixed tail-plane will tend to return to its original attitude if the longitudinal equilibrium is disturbed by a change of the angle of attack in either direction. The pitching moment about the center of gravity, which is manifestly zero for the equilibrium condition, will therefore be positive for all angles of attack smaller than the equilibrium angle and negative for all angles in excess of that value. The stability with locked elevators is the only true inherent stability, the airplane acting absolutely as a rigid body with no moving parts.

If a stable airplane is in equilibrium at a given angle of attack and it is desired to change the equilibrium condition to a larger angle a stalling moment must be imposed to balance the negative pitching moment which would arise from any increase of the angle of attack. This stalling moment is secured by pulling up the trailing edge of the elevator, so that the algebraic value of the angle at which it meets the air is decreased, and then locking it in this new position. Similarly, in order that the equilibrium angle may be decreased, the angle at which the elevator is fixed must be increased. The direct criterion by which the degree of stability or instability with locked controls can be judged from free-flight tests is then that the angle at which the elevator is set, relative to some line fixed in the airplane, shall diminish as the equilibrium angle of attack increases and the speed of flight decreases. A curve of elevator angle against speed will therefore have a positive slope for a stable airplane, and the magnitude of the slope of such a curve is at once indicative of the degree of stability.

It has been pointed out that stability under any particular condition is assured if the curve of pitching moments crosses the horizontal axis once and only once, the moments being negative for all angles larger than that of equilibrium, positive for all angles smaller. If the angle of elevator setting be changed, everything else remaining as before, the curve of pitching moments is little changed in form, but is slid vertically, remaining approximately parallel to itself, since a change of elevator setting modifies the angle of attack of the tail as a whole and

alters the lift coefficient by very nearly the same amount at all angles. If the curve did not change its form at all and remained exactly parallel to itself throughout it is evident that an airplane stable at all speeds would have a moment curve the slope of which would be negative at all points, since stability demands that the slope be negative where the curve crosses the horizontal axis and the curve can be so shifted, by adjustment of the elevator, as to cross the axis at any point of its length.

The curve of pitching moments would always move parallel to itself, and the criterion just mentioned would be perfect, if the tail as a whole always maintained the same form and if its lift coefficient curve were a straight line, so that a given change of angle would have the same absolute effect on the lift coefficient of the tail, whatever may have been the initial angle of attack. The second condition is very closely observed with all types of tail except when the tail is presented to the relative wind at an abnormally large angle, either positive or negative, and the first holds true when there is no fixed tail-plane, a change in the setting of the elevator therefore meaning a change in the angle of setting of the whole tail. In the much more usual design in which the tail is divided into fixed and movable portions, any change in elevator angle changes the sectional form of the tail, the effective camber becoming deeper as the elevator is turned in either direction away from the prolongation of the chord of the fixed portion. Since the slope of the lift curve is greater for a deeply-cambered tail than for one nearly flat, and since, as will be mathematically demonstrated a little later, the efficiency of the tail in producing stability depends primarily on the slope of the lift curve, it is clear that the stabilizing effect of a tail will be greatest when the angle between the tail-plane and elevator is considerable, or, in other words, when the tail-plane is set at such an angle that the machine is very nose-heavy or tail-heavy and that the elevator has to be held hard up or down in order to maintain equilibrium. While this has a good effect on stability with the controls locked, it makes the airplane very unpleasant to fly with free controls, and also decreases the efficiency of flight and the speed, the drag of the tail being much augmented, for a given lift, by setting the elevator at a considerable angle to the tail-plane. However, even though an airplane may be perfectly balanced under normal conditions, there is but one speed at which the elevator will lie exactly in line with the tail-plane, and at which as a result, the tail as a whole will have the designed section. The angle between the fixed and movable surfaces is greatest at very high and very low speeds, and the airplane is consequently liable to possess, under these extreme conditions, a higher degree of stability then would be prophesied from a wind tunnel test carried out, as such tests practically always are, with the elevator fixed parallel to the tail-plane for all angles of attack. This is especially true on those airplanes which have the gap between the tail-plane and elevator closed in some way. Another factor tending to give greater stability than that shown by model test is that, at extreme angles of attack, the fixed flat tail surfaces employed on wind-tunnel models meet the air at an angle approaching that of maximum lift, and exceeding that at which the lift curve begins to fall away from a straight line. The slope of the lift curve for the tail is therefore less than in steady free flight of the full-sized airplane, as the adjustment of the elevator with changing angle of attack is such, for a stable airplane, that the angle of the relative wind to the line connecting the leading edge of the tail-plane with the trailing edge of the elevator changes less rapidly as the angle of attack is varied than it would if the elevator remained fixed in one position relative to the airplane. Since both of these favorable effects (the effect of the elevator setting on camber of the horizontal tail surface as a whole and its effect on the true angle of attack of that surface) are most marked when the elevator angle varies most with changing speed and angle of attack, or, in other words, when the airplane is most stable with locked controls, it is evident that stability begets stability, and that the stability characteristics at high, low, and intermediate speeds are, to some extent, interdependent. The very act of increasing, by any means whatever, the degree of stability under normal conditions increases the stabilizing efficiency of the tail.

The pitching moment curve for any particular elevator setting can be studied and analyzed mathematically. If it be assumed, as a first approximation, that a negative slope of the pitching moment curve at all points is a sufficient condition of stability, the analysis can be confined to a single elevator setting.

The pitching moment under any conditions is equal to the sum of the moments due to the wings and tail (the effect of the body and chassis is small enough so that it can safely be neglected), and may be written:

$$M = M_1 + M_2$$

Developing each of the components,

$$M_1 = -(x-a) \times L_c \times A_1 \times V^2$$
$$M_2 = -(x'-a) \times L_{c2} \times A_2 \times V^2$$

where x = distance from leading edge of wings to C. P.

a = distance from leading edge of wings to C. G.

L_{c1} and L_{c2} = lift coefficients of wings and tail, respectively.

A_1 and A_2 = areas of wings and tail, respectively.

x' = distance from leading edge of wings to C. P. of tail surfaces.

V = speed of flight.

All distances are expressed, for convenience and uniformity, in terms of fractions of the wing chord.

The total moment is then:

$$M = -[(x-a) \times L_{c1} \times A_1 + (x'-a) \times L_{c2} \times A_2] \times V^2$$

Differentiating,

$$\frac{dM}{d\alpha} = -\left[(x-a) \times A_1 \times \frac{dL_{c1}}{d\alpha} + L_{c1} \times A_1 \times \frac{dx}{d\alpha} + (x'-a) \times A_2 \times \frac{dL_{c2}}{d\alpha} + L_{c2} \times A_2 \times \frac{dx'}{d\alpha}\right] \times V^2$$

The variation of V with regard to α can be neglected, as it would never result in changing the sign of the slope of the curve at its intersection with the horizontal axis.

Since it is the sign of the slope of the moment curve which is of primary interest, the factor V^2 can be disregarded for the present. In order that the airplane may be stable, $\frac{dM}{d\alpha}$ must be negative under all conditions, and the expression inside the brackets in the equation must therefore be positive. With wings and tail of ordinary section, the C. P. moving forward as the angle of attack is increased, the third of the four terms within the brackets is positive, while the second and fourth are negative. The sign of the first term depends on the location of the C. G. It is always positive at very small angles of attack, and some machines have the C. G. far enough forward so that the first term is positive under all conditions. If the lift curve be assumed to be a straight line, so that the value of L_c at any point is equal to the product of $\frac{dL_c}{d\alpha}$ by α (measured from the angle of zero lift) the factors $\frac{dL_{c1}}{d\alpha}$ and A_1 can be taken out of the equation just given for $\frac{dM}{d\alpha}$, and the expression inside the brackets can then be written, with all constant factors ignored:

$$(x-a) + \alpha \times \frac{dx}{d\alpha} + (x'-a) \times \frac{A_2}{A_1} \times \frac{dL_{c2}}{dL_{c1}} + \alpha' \times \frac{A_2}{A_1} \times \frac{dL_{c2}}{dL_{c1}} \times \frac{dx'}{d\alpha} =$$
$$(x-a) + \alpha \frac{dx}{d\alpha} + \frac{A_2}{A_1} \times \frac{dL_{c2}}{dL_{c1}} \times \left[(x'-a) + \alpha' \times \frac{dx'}{d\alpha}\right]$$

where α' is the angle of attack measured from the angle at which the lift of the tail is zero. Lanchester has given a construction[1] for the determination of a sufficient condition of stability, based on this assumption that the lift curve is a straight line and taking into consideration only the first two terms of the expression just given. It is, therefore, the construction for the condition under which the portion of the moment curve due to the wings alone has a negative

[1] The Flying Machine from an Engineering Standpoint, by F. W. Lanchester, N. Y., 1915.

sign, and is decidedly on the safe side. It is obvious that the expression inside the brackets must be positive, since x' is very large, and stability is therefore certain if the sum of the first two terms is positive.

If the tail had the same section, aspect ratio, and general efficiency as the wings $\frac{dL_{c2}}{dL_{c1}}$ would be approximately 0.6, as the rate of change of the angle at which the tail meets the air is diminished by the rate of change of the downwash angle, and this is about 0.4 as great as the rate of change of the angle of attack of the wings. As a matter of fact, however, the efficiency of the tail, as measured by the slope of the curve of lift coefficients, is about half that of the wings, and the value of $\frac{dL_{c2}}{dL_{c1}}$ is more likely to be 0.3 than 0.6.[2]

Since for any equilibrium condition the total pitching moment is zero,

$$(x-a)\times L_{c1}\times A_1=-(x'-a)\times L_{c2}\times A_2.$$

It can then readily be shown that, still making the assumption that the lift curve is a straight line,

$$\alpha'\times\frac{dL_{c2}}{dL_{c1}}\times\frac{A_2}{A_1}=-\alpha\times\frac{(x-a)}{(x'-a)}$$

Substituting the second of these values for the first in the fourth term of the equation for slope of the moment curve, the variable part of that equation becomes:

$$(x-a)+\alpha\left[\frac{dx}{d\alpha}-\frac{x-a}{x'-a}\times\frac{dx'}{d\alpha}\right]+\frac{A_2}{A_1}\times\frac{dL_{c2}}{dL_{c1}}\times(x'-a) \qquad (1)$$

Examining each term of this equation in turn, it appears that the first term is always positive at small angles, when the center of pressure is far back, and may be either positive or negative at large angles. Its algebraic value can be increased to any desired extent by moving the center of gravity forward. The expression inside the brackets is always negative except in those rare cases where the wing cell itself has a "stable" center of pressure travel. The negative value can be reduced by moving the center of gravity forward, by shortening the fuselage, by using a wing section with a more stable center of pressure travel, or by using a tail surface with a more stable C. P. travel if the C. G. is back of the C. P. of the wings, so that $\frac{x-a}{x'-a}$ is negative. If $\frac{x-a}{x'-a}$ is positive it is disadvantageous to stability to have $\frac{dx'}{d\alpha}$ positive (i. e., to have a stable motion of the C. P. of the tail). Since $\frac{x-a}{x'-a}$ is usually positive at some angles of attack and negative at others it is rather difficult to tell what properties should be sought in a tail-plane to give the best stability. In any case, however, the effect of the second term inside the brackets is small as $\frac{x-a}{x'-a}$ is almost always less than 0.05, and the tail section may be chosen from considerations quite unconnected with stability. Finally, the last term in (1) is always large and positive, and can be increased by increasing the length of the body, the area of the tail, or its efficiency as defined by the slope of the curve of lift coefficients.

The virtues, as a stabilizing agent, of a tail-plane set at a negative angle to the wings have been understood for many years, such a disposition of surfaces having been used by Penaud on his rubber propelled models about 1870. It is not so universally comprehended, however, that the inherent direct advantages of a negative tail-plane setting, are slight, arising only from the greater efficiency of the tail under those conditions, with the elevator held at a considerable angle to the tail-plane to give equilibrium, and that the great merit of such a setting is that it permits the center of gravity to be placed very far forward without throwing the airplane badly out of balance. The really crucial points in connection with stability with locked controls are the position of the center of gravity, the size and efficiency of the tail-plane, and the length of body, and that the first of these is by far the most important.

[2] It has been found at the Royal Aircraft Establishment that the tail efficiency for a B. E. 2E ranges from 0.5 at high speeds to 0.75 at low. (Full scale stability experiments on a B. E. 2E with R. A. F. 15. wing section: R. & M. (New Series) No. 326: 1917.

Lanchester's construction has been carried through for a number of representative aerofoil sections, and it has been found that the C. G. must lie (taking the average result for the several wings, which vary only slightly among themselves) not more than 0.23 of the way back on the mean wing chord if stability is to be secured without any assistance from the tail.

A wing having a "stable" center of pressure travel does not necessarily give an airplane complete stability, as shown by the curve of pitching moments. For example, an airplane with flat plate wings and no tail, and with the C. G. anywhere between the leading edge and the middle of the chord, would be stable at some definite angle of attack, so far as small disturbances and small excursions from that angle were concerned. It would, however, be subject to "catastrophic instability" in the event of large disturbances, the curve of pitching moments cutting the horizontal axis at three points within the range of possible flight angles, two of these points corresponding to stable conditions of flight, the third to an unstable condition. To completely insure against such instability it would be necessary to provide a tail and to move the C. G. forward at least to the leading edge of the wing, farther forward than is required with "unstable" wings of cambered section. In the case of an airplane in which the center of pressure of the wings approaches the leading edge as the angle approaches zero, as in the flat plate, the danger of getting into the inverted equilibrium position is greatest when the C. G. is far forward (but still back of the leading edge), as the angles of attack for normal flight and steady upside-down flight are then very close together. An airplane which flies normally at 8°, for example, and which has another point of equilibrium at $-8°$, is much less likely to be thrown into the inverted position by atmospheric disturbance or by an inadvertence on the part of the pilot than it would be if the angles were $+2°$ and $-2°$. In a certain sense, all airplanes are catastrophically unstable, since the curve of pitching moments, being continuous throughout 360°, must cut the horizontal axis at least twice if it cuts it at all. For a flat plate alone, with the C. G. anywhere along the chord, the curve cuts the axis four times during the complete circle. For a typical cambered wing, the curve cuts the axis twice if the C. G. lies in the first 30 per cent of the chord, four times if it lies between 0.3 and 0.5 of the way back. All wings, both flat and cambered, have a point of stable equilibrium at a small negative angle of attack, and it is the function of the tail to shift this point of equilibrium to an angle of positive lift. The other point of intersection, for a wing with the center of pressure far forward or for a complete airplane, is one of unstable equilibrium, and occurs at an angle of approximately 180°, corresponding to the conditions during a tail-slide. Catastrophic instability need then occasion no difficulty if the C. G. is located far enough forward, as it is easy to secure a moment curve which will have a negative slope at all angles from $-40°$ to $+40°$.

If, in (1) 0.3 be substituted for $\frac{dL_{c2}}{dL_{c1}}$, 0.13 for $\frac{A_2}{A_1}$, and 3.75 for $(x'-a)$, these values corresponding roughly to the average dimensions used at the present time, and if it be assumed that the motion of the C. P. of the tail is in the same direction and half as great as that for the wings, the formula becomes:

$$(x-a)+\alpha\times\frac{dx}{d\alpha}\times\left[1-\frac{x-a}{7.5}\right]+.146. \tag{2}$$

The factor inside the brackets is so nearly equal to 1 that it can safely be disregarded for purposes of approximation.

The mean values of the second term and of x determined by wind tunnel tests for a number of commonly-used wing sections are:

α	2°	3°	4°	5°	6°
$\alpha\times\frac{dx}{d\alpha}$	-0.333	-0.181	-0.142	-0.137	-0.128
x	.53	.46	.43	.40	.38

It should be borne in mind that these figures and the results deduced from them are only illustrative, relative to averages of wind tunnel tests and that they are subject to verification

by free flight tests. Substituting these mean values in (2), the value which a must not exceed in order that the moment curve may have a negative slope can at once be found. This value is smallest for the smallest angle, where it is 0.35. In other words, the center of gravity of the airplane must lie not more than 35 per cent of the way back on the mean wing chord if stability is to be secured. If the tail area were decreased 50 per cent, the angle of setting being changed at the same time but the section and plan form remaining fixed, the C. G. would have to be moved forward until a became less than 0.28. If, on the other hand, the tail efficiency, as measured by $\frac{dL_{c2}}{dL_{c1}}$ be increased 50 per cent without change of area (a feat which should not be very difficult to accomplish in some present-day airplanes), the C. G. could be moved back to a point about 40 per cent of the chord from the leading edge without causing the airplane to become unstable. This backward movement of the C. G. decreases the load on the tail surfaces and improves the general efficiency of flight.

SLIP-STREAM EFFECTS.

The analysis so far has proceeded on the assumption that all parts have the same speed relative to the air through which they pass. This assumption is correct in the case of gliding flight, but it is very far from the truth with the throttle open, the mean air-speed at the tail being much higher than that over the wings, since nearly the whole tail lies in the slip-stream on most airplanes. If the slip-stream velocity varied in the same manner and proportion as the speed of the airplane relative to the undisturbed air the higher velocity would operate only to make the tail more effective and so to make the airplane more stable. Unfortunately, however, this condition does not prevail. It has been shown by theory and by experiment [3] [4] that the ratio of slip-stream velocity to air-speed increases as the air-speed decreases, and, in fact, that the speed in the slip-stream is almost independent of the air-speed.

The pitching moment equation with allowance for the slip-stream, if it be assumed that the whole of the tail, but no part of the wings, lie in the slip-stream, is:

$$M = -[L_{c1} \times A_1 (x-a) \times V^2 + L_{c2} \times A_2 \times (x'-a) \times V_s^2]$$

where V_s is the slip-stream velocity. In this case the effect of a change in elevator setting is, as before, to slide the moment curve vertically, but it slides the curve parallel to itself only if the velocity across the tail does not change. It will be recalled that this same condition was laid down in the case where slip-stream effect was ignored, and that the analytical work was accordingly carried through on the assumption that the velocity remained constant. Similarly in the present instance it will be necessary to assume that the slip-stream velocity passing over the tail remains constant while the speed of flight varies. Differentiating M with respect to the angle of attack, treating V as a variable:

$$-\frac{dM}{d\alpha} = \left[\frac{dL_{c1}}{d\alpha} \times (x-a) \times V^2 + \frac{dx}{d\alpha} \times L_{c1} \times V^2 + 2V\frac{dV}{d\alpha} \times L_{c1} \times (x-a)\right] \times A_1$$

$$+ \left[\frac{dL_{c2}}{d\alpha} \times (x'-a) + \frac{dx'}{d\alpha} \times L_{c2}\right] \times A_2 \times V_s^2$$

Strictly speaking, $\frac{V}{V_s}$, instead of V, should be taken as an independent variable, but, as has already been pointed out, V_s varies so little with airspeed in normal flight with wide-open throttle that it can be considered in an approximate treatment as actually remaining constant.

[3] Preliminary Report on Free-Flight Testing, by E. P. Warner and F. H. Norton: Report No. 70, National Advisory Committee for Aeronautics, Washington, 1920.

[4] Slip-Stream Corrections in Performance Computations, by E. P. Warner: Report No. 71, National Advisory Committee for Aeronautics, Washington, 1920.

Numerous reports of the British Advisory Committee for Aeronautics also deal with this subject.

In order that the airplane may be in equilibrium, the condition

$$(x-a)\times L_{c1}\times A_1\times V^2 = -(x'-a)\times L_{c2}\times A_2\times V_s^2$$

must hold true. If, as before, the lift coefficient curves be assumed to be straight lines, so that

$$L_c=\alpha\times\frac{dL_c}{d\alpha},$$

the equation of equilibrium becomes:

$$\frac{dL_{c2}}{d\alpha}\times(x'-a)\times A_2\times\alpha' = \frac{-(x-a)\times A_1\times V^2\times\frac{dL_{c1}}{d\alpha}\times\alpha}{V_s^2}$$

Substituting the expression on the right-hand side of this equation for that on the left in the expression for the slope of the moment curve, so as to eliminate α',

$$-\frac{dM}{d\alpha}=\left[(x-a)\times V^2+\frac{dx}{d\alpha}\times\alpha\times V^2+2V\frac{dV}{d\alpha}\times\alpha\times(x-a)\right]\times A_1\times\frac{dL_{c1}}{d\alpha}$$
$$+\left[\frac{dL_{c2}}{d\alpha}\times(x'-a)\times A_2\times V_s^2\right]-\frac{x-a}{x'-a}\times A_1\times V^2\times\frac{dL_{c1}}{d\alpha}\times\alpha\times\frac{dx'}{d\alpha}$$

Taking A_1, $\frac{dL_{c1}}{d\alpha}$, and V_s^2 out as factors, the condition of statical stability becomes:

$$\left[(x-a)\times\left(\frac{V}{V_s}\right)^2\right]+\left[\frac{dx}{d\alpha}\times\alpha\times\left(\frac{V}{V_s}\right)^2\right]+\left[2\frac{V}{V_s^2}\times\frac{dV}{d\alpha}\times\alpha\times(x-a)\right]+$$
$$\left[\frac{dL_{c2}}{dL_{c1}}\times\frac{A_2}{A_1}\times(x'-a)\right]-\left[\frac{x-a}{x'-a}\times\left(\frac{V}{V_s}\right)^2\times\alpha\times\frac{dx'}{d\alpha}\right]>0.$$

If the airplane is in level or approximately level flight (inclination not in excess of 20°), as is usually the case when the throttle is open,

$$W=L_{c1}\times A_1\times V^2=\alpha\times\frac{dL_{c1}}{d\alpha}\times A_1\times V^2$$

neglecting the lift on the tail, where W is the total weight of the machine. Then

$$\frac{W}{\alpha}=\frac{dL_{c1}}{d\alpha}\times A_1\times V^2$$

$$-\frac{W}{\alpha^2}d\alpha=\frac{dL_{c1}}{d\alpha}\times A_1\times 2VdV$$

$$\frac{dV}{d\alpha}=-\frac{W}{\alpha^2\times\frac{dL_{c1}}{d\alpha}\times A_1\times 2V}=-\frac{V}{2\alpha}$$

Substituting this value for $\frac{dV}{d\alpha}$ in the stability equation, the third term exactly cancels the first, and:

$$\alpha\times\left(\frac{V}{V_s}\right)^2\times\left[\frac{dx}{d\alpha}-\frac{x-a}{x'-a}\times\frac{dx'}{d\alpha}\right]+\left[\frac{dL_{c2}}{dL_{c1}}\times\frac{A_2}{A_1}\times(x'-a)\right]$$

must be positive for stability. It will be noted that the term depending directly on the relation between the position of the C. G. and that of the C. P. of the wings does not appear in this equation, and that C. G. position has only a very slight effect on stability. In fact, the stability

can only be increased, if the velocity of flow over the tail be constant, by increasing the area or efficiency of the tail surfaces or the length of the body.

It is evident that it is much more difficult to secure stability when the velocity across the tail is constant than when it varies in the same manner as that across the wings. The actual condition always lies somewhere between these two extremes, and the stability is improved as the tail is brought out of the slip-stream in whole or in part, thus approaching more nearly the second limiting condition.

It can be seen from physical reasoning that, if the slip-stream velocity is kept constant while the air-speed varies in such a manner as to keep the lift constant, the stability must be nearly independent of C. G. position, as the form of a curve of moments due to a series of parallel forces is independent of the position of the moment axis if the sum of the upward forces be equal to the weight of the machine at every angle of attack, and a shift of the C. G. therefore changes the wing moment by the same amount for all angles.

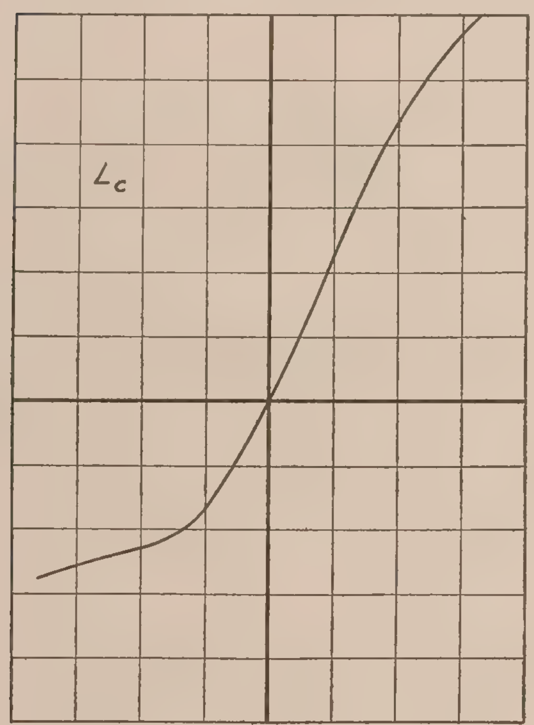

Fig. 1.

It has been shown that it is always advantageous to increase as much as possible the efficiency of the tail surfaces as measured by the slope of the curve of lift coefficients. If a section flat on one side and cambered on the other be tested at both positive and negative angles (measuring angles from the zero lift position and defining their signs on the assumption that the cambered surface is uppermost), it is found that the curve of lift coefficient against angle of attack has the general form shown in figure 1, and that the slope of the curve at the point corresponding to any given positive lift coefficient (except a very small one) is materially greater than that at the point where there is a negative L_c of the same absolute magnitude. The tail-plane should therefore be so set, for best efficiency, as normally to work at a positive angle. Since the C. P. of the wings is behind the C. G. at all times on some airplanes, and at all except very low speeds on all, the load on the tail-plane is normally downward. In order that there may be a downward force while the tail is set at a positive angle of attack, using the term positive angle to denote the condition in which the flat surface of the tail experiences a larger normal pressure (algebraic value) than the cambered surface, the tail must be inverted, with the flat surface on top. It appears from the analysis that this disposition, which has been employed in the Pfalz and numerous other machines, possesses distinct advantages. The increase in stability by inversion of the tail should be greatest at high speeds, as it is at high speeds that the normally placed tail-plane meets the air at a large negative angle where the slope of the lift curve is small, and there is consequently more room for improvement under those conditions than under any others. At intermediate speeds, where the load on the tail is small and where there is little difference in the form and slope of the lift curve for equal positive and negative coefficients, it should make little difference whether the upper or lower surface is the cambered one. The position of the C. G. and the tail area can therefore be modified to change the stability of the airplane at all speeds, while the relation between stability at high speeds and at low can be controlled to some extent by altering the sectional form.

EXPERIMENTS ON STABILITY WITH LOCKED CONTROLS.

The experiments made to verify the above theory and to determine the degree of stability fall into two classes. In the first the airplane was flown under the control of the pilot, the elevator angles were dertermined for several air-speeds and a constant throttle setting, and the angles thus measured were plotted against air-speed. In the second series of experiments the airplane was actually flown with the elevator locked in several different positions and the nature

of its motion was observed. In order that the experiments might not be complicated by any interaction of longitudinal and lateral stability the locking device was so arranged as to permit the stick to be moved from side to side while keeping it in the same fore-and-aft position. The pilot was thus able at all times to make use of the rudder and ailerons to keep the airplane on an even keel laterally. The measurement of elevator angles and the curves based on those measurements admittedly are not very accurate in form, as the constant fluctuations in position due to minute air disturbances are large in comparison with the changes of angle as the speed changes, the stability being very near to neutral in most instances. However, while the accuracy is not great enough to permit of the making of delicate determinations of the point at which the instability appears or of refined analysis of the effect of changes in the airplane, it is still sufficient to show any large variations in stability and to check the analysis approximately.

The variables which were changed in these experiments were:

(*a*) Horizontal position, or *X*-coordinate, of the C. G.

(*b*) Stagger.

(*c*) Angle of setting of the tail-plane.

(*d*) Sectional form of tail-plane.

(*e*) Vertical position, or *Z*-coordinate, of the C. G.

A JN4H was used in all these experiments. Assembly drawings of this airplane were given in Report No. 70. The DH4 is the only other type of airplane on which any full-scale experiments on stability with locked controls or on elevator positions have been carried on in America. (*b*) has the same effect as (*a*) in that it moves the C. G. relative to the wings, but changing the stagger also has a direct effect, as it modifies the travel of the center of pressure of the wings. It is necessary to reduce the stagger below normal if the C. G. is to be brought very far forward on the mean chord, as the attachment of enough weight at the nose to move the C. G. to a distance of less than one-third of the chord from the leading edge of the mean chord would bring the C. G. so close behind the axle as to entail serious danger of nosing over. The tail-plane angle was modified by placing blocks under the leading or trailing edge, the fin being cut away at the bottom to provide clearance. The sectional forms tested were three in number. The standard tail was tested both in normal position and inverted, and the third arrangement was a tail of symmetrical section made by attaching convex fairings on the flat lower surfaces of the ribs of the standard tail. Only a single alteration was made in the vertical position of the C. G., a weight being attached to the axle during one test.

The data permitting direct study of the effect of C. G. position without the introduction of any other complicating factors are unfortunately rather sparse. As already noted, the C. G. can not be moved far forward of its normal position without danger of nosing over, and movement to the rear through more than 3 or 4 per cent of the chord length makes the airplane tail-heavy and tiresome, if not dangerous, to fly. Tests with the C. G. position coefficient (the ratio of the distance between the leading edge of the mean chord and a line through the C. G. perpendicular to the thrust line to the chord length) at 0.365 and 0.335, the stagger being 13 inches in both cases, show an improvement of stability with the throttle closed as the C. G. is moved forward. With the throttle open, the difference between the two cases is negligible.

This is strictly in accordance with theoretical deduction. As in the cases detailed in Report No. 70, the airplane is stable at large angles of attack and unstable at all speeds beyond a certain point. Instability with the throttle closed appears at a mean speed of 78 m. p. h. with a C. G. coefficient of 0.365, at 82 m. p. h.when the coefficient is 0.335. This difference, while distinct, is much smaller than might have been predicted. It is probable that one reason for the small apparent effect of the C. G. position is that the tail-plane in this series of tests was blocked up to a negative angle (2.9° to the top longeron), that the elevators had to be pulled down to maintain equilibrium, and that they were pulled down farther when the C. G. was back and the machine was tail-heavy than when it was forward. The combination of tail-plane and elevator then acts roughly as a cambered surface, and the camber is deepest and the effectiveness of the surface in producing lift is highest when the C. G. is farthest back. This increased

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effectiveness of the tail unit partially counterbalances the less stable form of the curve of moments due to the wings. Not even with a C. G. coefficient of 0.29, this being the farthest forward position that was tried, was there complete stability at high speeds. The prediction from the model test that stability at all speeds would be obtained with a C. G. coefficient of 0.35 is thus shown to be incorrect for this machine, and it is evident either that the travel of the C. P. of the wings in free-flight is different from that found from a model test or that the efficiency of the tail is less than was estimated. The latter is very probable, as the aspect ratio of the tail is low and the section thin.

Changes in stagger, like those in C. G. position, had very little direct effect. It is necessary to use a positive stagger of at least 50 per cent of the chord if any improvement in stability is to be secured by modification of the nature of the center of pressure travel, and the maximum stagger used in any of these tests on the JN4H was the normal amount, 27 per cent.

The effect of modification in the tail-plane angle was much larger than had been anticipated. As a concrete instance, three tests made with different tail-plane settings may be compared. With the tail plane set flat on the top longerons, the airplane became unstable at 57 m. p. h. with throttle open and at 62 m. p. h. when gliding. With the tail-plane set at $-1.4°$ to the top longerons the corresponding figures were 67 and 72 m. p. h., and when the angle of setting was increased to $-2.9°$ the critical speed was 75 m. p. h., both with open and with closed thr ttle. It can not be claimed that these speeds are correct to any high order of accuracy, but they are probably good to within a maximum error of 6 m. p. h. and a probable error of 3 m. p. h. The apparent change of stability with change of tail-plane setting is large enough so that, despite the considerable errors which may be present, the general trend of the variation, at least, is fairly certain. It will be seen that the range of speed in which the airplane is stable constantly increases as the rear of the tail-plane is raised. This points to a considerable indirect advantage in moving the C. G. forward, as the elevator angle for zero force on the stick (a condition always to be sought for when in equilibrium at normal speeds, even if stability must be sacrificed to obtain it) probably is nearly independent of the tail-plane setting, and the angle between tail-plane and elevator when properly balanced is therefore greatest when the C. G. is far forward and when the tail-plane has to be set at a large negative angle to keep the nose up. It has already been pointed out that setting the two portions of the surface at a large angle to each other improves the tail efficiency. Study of the results of wind tunnel tests on wings with hinged rear portions set at various angles does not indicate a change in slope of the lift coefficient curve sufficient to account for the magnitude of the effects observed in the present experiments, and it is probable that the direct effect of tail-plane setting would be much less if the tail were of reasonably thick and efficient section than it was in the present case where the section of the tail approximated to that of a flat plate. The effect of tail-plane setting on stability was much less marked in the DH4 than in the JN (see Report No. 70).

The effect of sectional form is to change the relative stability at different speeds, as was predicted from the theoretical analysis. The building up of the tail-plane to a symmetrical form increases the stability at high speeds while decreasing it at low, and the inversion of the tail-plane has the same effect in a still more marked degree. The inversion of the tail-plane raised the speed at which instability appeared by 5 m. p. h., and made the instability much less marked when it did appear, the elevator angle with throttle closed, with 6 inches stagger, and with the tail-plane at $-1°$, varying through a total range of less than $\frac{1}{2}°$ at all speeds from 60 to 91 m. p. h.

The next experiments dealt with the effect of lowering the C. G., the object being to reduce the difference between the balance with throttle open and with throttle closed by bringing the C. G. below the thrust line and so causing the thrust to produce a diving moment counterbalancing the stalling moment due to the action of the slipstream on the controls. The C. G. was lowered about $1\frac{1}{4}$ inches in the only experiment of this series so far conducted, and no effect was apparent, presumably because the change was not large enough. Calculations of a necessarily very approximate nature indicate that the C. G. would have to be lowered about a foot to bring the curves for all throttle settings into coincidence.

Summarizing the results of these experiments, it may be said that they check extremely well with the theory except that the effect of C. G. position is less, that of tail-plane setting more, than was expected.

As an index of the magnitudes of the total effects of these modifications, the curves for the JN with the standard rigging are plotted in figure 2 together with those for the most stable arrangement tried (6 inches stagger, coefficient of C. G. position 0.290, tail-plane inverted and set with chord at $-1°$ to top longerons).

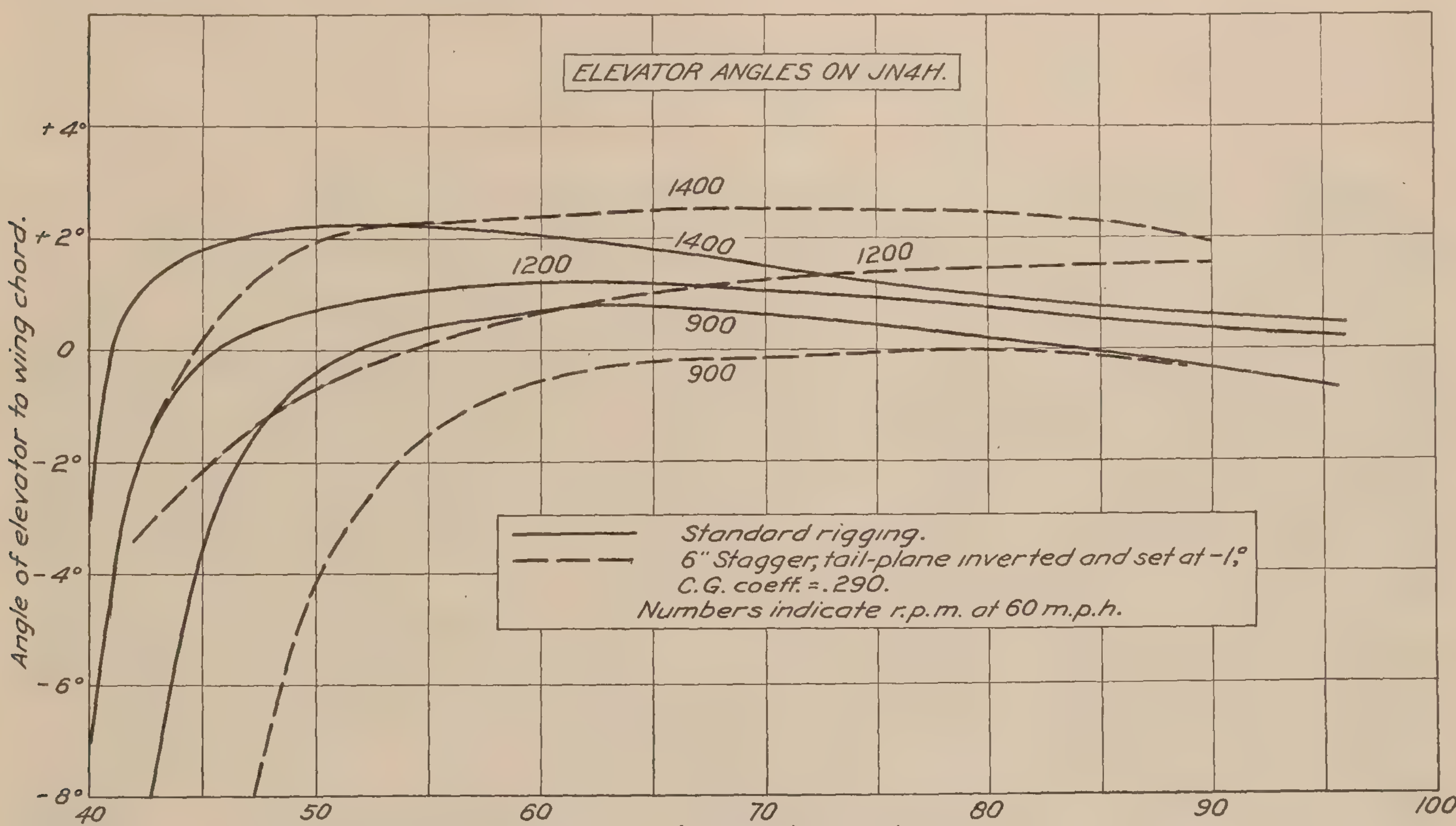


Fig. 2.

EXPERIMENTS ON ACTUAL FLIGHT WITH LOCKED CONTROLS.

To secure the most direct possible check on all of this theoretical and experimental work, a number of attempts to fly the airplane with the elevator controls locked were made. It was found that, as indicated by the angle curves, the standard JN was quite unstable except at very low speeds. When the stagger was decreased to 13 inches and the tail-plane set at $-2°$ to the top longeron the airplane was stable with locked controls throughout the range of normal flight speeds when the throttle was closed and at speeds up to 65 m. p. h. with open throttle. The "peak" of the curve of elevator angles, supposed to represent the point where instability begins, is at approximately 72 m. p. h., but the curve is so flat between this point and 65 m. p. h. that a "bump" or other disturbance is likely to throw the airplane over the "peak," to an angle corresponding with a speed greater than 72 m. p. h. When this happens, the speed continues to increase, and the machine would presumably ultimately go over on its back if the pilot did not resume control. This smallness of the reserve of statical stability for the purpose of resisting atmospheric or other disturbing factors is an objectionable feature of all arrangements in which the stability curve is very nearly horizontal for a considerable distance (as, for example, in the case of the inverted tail-plane, already discussed).

The airplane could be set oscillating, with the controls locked, by quickly opening and closing the throttle. When this was done anywhere within the stable range of speeds the airplane started oscillating with a period of about 20 seconds, the motion being well damped and dying out quickly. In normal weather the flight with locked elevator, aside from such artificially-produced irregularities, is very steady, comparing favorably in that respect with fully-controlled flight by the average pilot.

The conclusions to be drawn from all this experimental work are that the C. G. should be far forward, certainly not over 30 per cent of the mean chord back of the leading edge of that chord, that the tail-plane should be at a negative angle such that the machine balances without any force on the stick at the best climbing speed, that the C. G. should be as low relative to the thrust line as is possible without disarranging the essentials of the design or decreasing its usefulness in any way, and that the tail should be of thicker section than is the common practice at present, with at least a part of the convex camber on the lower surface. It is impossible to be sure until more data on numerous different types of airplanes have become available, but it is not believed that it will ordinarily be advisable to go to the extreme of using a flat upper surface with a convex lower.

STABILITY WITH FREE CONTROLS.

Stability with free controls is much more difficult to treat theoretically than is that with controls fixed, but it is easier to secure accurate experimental data for the first condition than for the second.

As pointed out in Report No. 70, it is impossible to predict accurately the behavior of an airplane with free controls except after an exhaustive series of tests on the pressure distribution over the tail, as the moment about the hinge is governed by the position of the center of pressure of the elevator and the motion of the center of pressure on a surface hinged to the rear of another surface is a very uncertain quantity, especially when the elevator is set at an angle close to that of zero lift, as is usually the case with a properly balanced airplane. An approach to the theory of stability with free controls can best be made by considering separately several simplified cases.

The simplest possible case is that in which, as on the old Grade and Wright model B, the Salmson, and some of the Halberstadts and Moranes, there is no tail-plane, the whole horizontal tail-surface moving as one piece (or, in the Wright and Grade, flexing). The air load on such an elevator must be downward, thus acting with the weight of the elevator, whenever the center of pressure of the wings is behind the C. G. With the C. G. located in accordance with the present practice, the total moment about the hinge is likely to be such as to require a pull on the stick at all times, the airplane not being truly balanced at any speed unless a spring or elastic is attached to the stick to hold it back and reduce the effort required from the pilot. This expedient is employed on the Salmson. Obviously, since the center of pressure moves farther to the rear of the C. G. as the angle of attack decreases, the download carried by the elevator increases as the speed increases. On the other hand, since, as shown in Report No. 70, the criterion of stability with free controls is that the pull on the stick must decrease, and ultimately become a push, as the speed increases, the moment about the hinge must decrease with increasing speed. If the moment is to decrease while the force increases, it is evident that the center of pressure on the elevator must move forward as the speed rises.

At this point in the analysis three cases must be recognized and treated separately, two relating to gliding conditions and one to flight with the throttle open. It has just been seen that $L_{c2}\,A_2V^2$, where L_{c2} is, as before, the lift coefficient for the tail, increases with increasing speed. The rate of change of the force on the tail, or the amount of change for a given alteration in speed, is independent of the position of the C. G., but the relative change, the ratio of the forces for any given pair of speeds, is governed entirely by that factor. Writing the complete equation for force on the elevator, the symbols having the same significance as in the part of the report dealing with locked controls:

$$L_{c2}\,A_2V^2 = -L_{c1}\,A_1V^2 \times \frac{x-a}{x'-a} = -W \times \frac{x-a}{x'-a}$$

$$L_{c2} = -\frac{W}{A_2V^2} \times \frac{x-a}{x'-a} = L_{c1} \times \frac{A_1}{A_2}\,\frac{x-a}{x'-a}$$

Differentiating this with respect to the angle of attack, neglecting variations in x':

$$\frac{dL_{c2}}{d\alpha}=-\frac{A_1}{A_2\times(x'-a)}\times\left[\frac{dL_{c1}}{d\alpha}\times(x-a)+\frac{dx}{d\alpha}\times L_{c1}\right]$$

Assuming $L_{c1}=\alpha\times\frac{dL_{c1}}{d\alpha}$, as before:

$$\frac{dL_{c2}}{d\alpha}=-\frac{A_1}{A_2}\times\frac{1}{x'-a}\times\frac{dL_{c1}}{d\alpha}\times\left[(x-a)+\alpha\times\frac{dx}{d\alpha}\right]$$

If the expression inside the brackets is positive, the lift coefficient of the elevator must decrease in absolute value as the angle of attack decreases and the speed of flight increases, while the change is in the reverse direction if the sum of the bracketed terms is negative. It is therefore necessary for stability with free controls that the center of pressure of the tail move forward as the lift coefficient decreases if the C. G. is forward of a certain critical point, the location of which depends on the characteristics of the wing, and must move forward with an increasing L_c if the C. G. is behind that point. In other words, the C. P. travel on the elevator must be "stable" in the first case, "unstable" in the second. In wings of normal form for which the calculation has been made, the farthest forward location of the point just alluded to ranged from 20 per cent to 25 per cent of the way back on the mean chord (assuming the strict applicability of wind tunnel results). At low speeds the point lies about 30 per cent of the chord length from the leading edge. If the C. G. were exactly coincident with the critical point at any instant, the angle between the elevator and the relative wind would not change at all as the speed changed slightly. The center of pressure on the tail therefore could not move, and it would be utterly impossible to secure stability with free controls as shown by the curve of stick forces (see Report No. 70).

There remains to be considered the case of flight with open throttle. This, as for locked controls, will be treated on the extreme assumption of a constant velocity in the slip-stream. Since the load on the tail increases with increasing speed of flight, it is evident that the lift coefficient for the tail must increase if the speed in the slip-stream remains constant. The reasoning is then the same as for the case with throttle closed and C. G. back, and the travel of the center of pressure must be "unstable." If the C. G. is forward of the critical point, then, the requirements for stability with open and with closed throttle are diametrically opposed and absolutely incompatible. In this type of machine (one with no fixed tail-plane) the stability with free controls is actually injuriously affected by moving the C. G. forward beyond a certain point. It will be shown later that other arrangements are not subject to this disadvantage, or at least not in the same degree, and the use of an elevator without a tail-plane is therefore to be avoided if stability is desired, entirely apart from its disadvantages in respect of ease of control, other things being equal.

The next illustrative case to be analyzed is that in which there are a separate tail-plane and elevator working entirely independently of each other, being placed side by side, as in the Bleriot XI bis, instead of in tandem, as is the present practice. In this case the tail-plane carries a down load at high speeds and an up load at low. If the tail-plane be made large enough, and be set at a large enough negative angle, to give statical stability by the locked-control criterion and to balance the airplane at some angle in the normal flying range without any assistance from the elevators (that is, if a wind tunnel test of a model with the elevators removed gives a curve of pitching moments which has a negative slope everywhere and which cuts the axis of zero moment somewhere between 0° and 12°), it is evident that, if the elevator section be assumed to be symmetrical about its center line and if the effect of the elevator's weight be neglected or be assumed balanced by a spring or counterweight, the airplane will fly with no force on the stick at the same angle and speed at which it was found to be in equilibrium with the elevators removed. Furthermore, the maintenance of equilibrium at any higher speed will require that the elevator furnish a diving moment to counteract the stalling moment due to the inherent stability without the elevators, and there will therefore be an upward load on the ele-

vators and a push on the stick. At all speeds lower than the normal trimming speed, similarly, there must be a pull on the stick. It is then certain that the airplane will be completely stable with free controls at the normal trimming speed, but it does not follow that the slope of the stick force curve is everywhere negative, as would be necessary if there were to be stability at all speeds. If the elevator section is not symmetrical, but is of aerofoil form, the elevators will take up a position for which the moment about the hinge is zero. There will then be a downward force if the upper surface of the elevator is convex and the lower one flat or at least less convex, as the lift coefficient for an aerofoil section is always negative when the moment about the leading edge is zero. The elevator will therefore give a stalling moment, and the airplane will fly in equilibrium with free controls at a larger angle of attack than that at which the moment is zero with the elevators removed.

If the weight of the elevators be taken into account it is clear that they will hang down at such an angle that the moment about the hinge due to the air forces is equal to that due to the weight. In general, this means that there will be a small upward force on the elevators

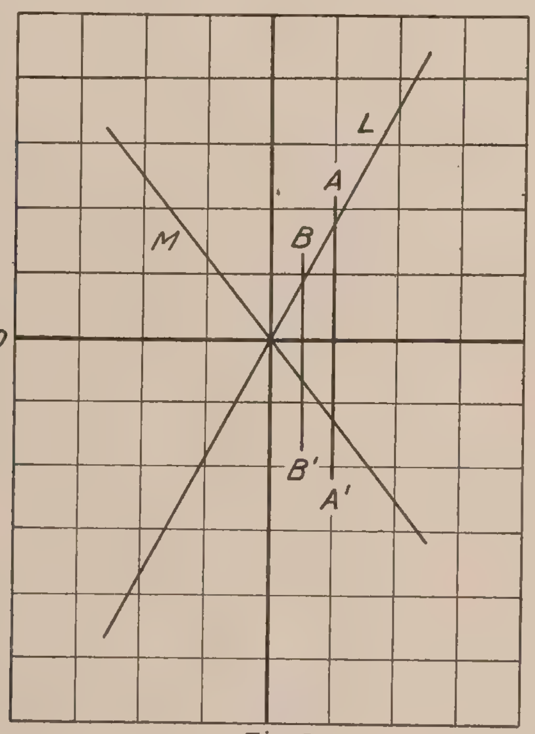


Fig. 3.

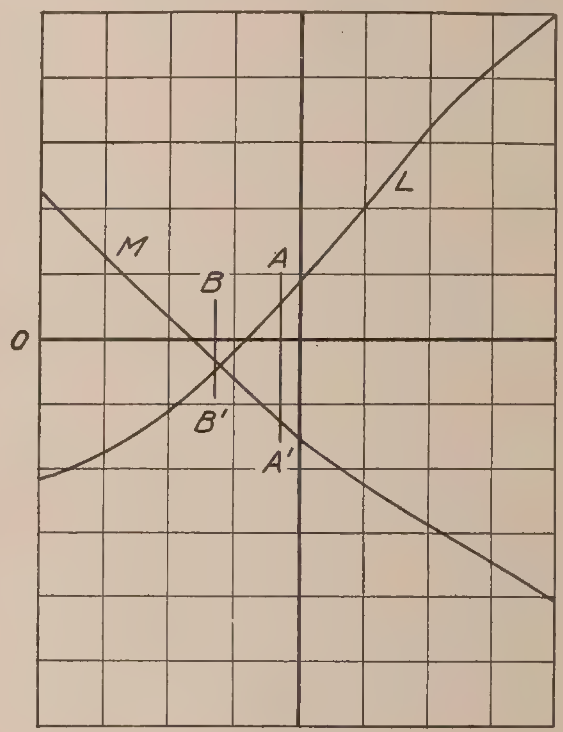


Fig. 4.

and that a diving moment will act on the airplane as a result. The effect of elevator weight on stability and on the form of the stick force curves can best be shown with the aid of a graph. The curves of lift coefficient and of moment coefficient (moment about the leading edge) for a symmetrical section are diagrammatically shown in figure 3, and those for a representative aerofoil section are similarly shown in figure 4. All of these curves are straight lines, to a first order of approximation, in the neighborhood of the zero lift angle. Let it be supposed that on an airplane with an elevator of symmetrical section the elevators hang at the angle represented by the line AA' when flying at the speed V_1, and that, at the higher speed V_2, they would hang freely at the angle indicated by BB' (of course there can actually be only one trimming speed for steady flight with free elevators), the moment coefficients at the two angles being inversely proportional to the squares of the corresponding speeds since the total moment must be constant and equal to the moment of weight. Since the curves of L_c and M_c are both straight lines passing through the origin, the ratio of the two is the same at all angles of attack. Then, since M_c is inversely proportional to V^2 the total lift on the elevators, or product of L_c and V^2, is constant. The lift is the same at BB' as at AA', the diving moments in the two cases are therefore the same, and the analysis carried through for the weightless elevator holds good without change. It is still true that an increase of speed requires a diving moment from the elevator, that this in turn exacts an increase in the up load on the surface, and that an increasing up load means an increasing moment about the hinge and a pull on the stick, so that there is stability with free controls at least at the trimming speed. It is now, however, apparent, since each successive increase of lift entails a further increase of moment about the

hinge, that the curve of stick forces has a positive slope and a stable form throughout the whole range of flight speeds.

Passing now to the case of the aerofoil section, where the curves of L_c and M_c no longer intersect at the origin, it is evident that it is no longer true that total lift and total moment are in a fixed ratio to each other. In passing from AA' to BB' in figure 4, choosing BB' so that $M_c V^2$ will be constant, L_c actually changes sign. For an aerofoil section right side up, as the speed increases and M_c decreases the algebraic value of $\frac{L}{M}$ grows less and the diving moment due to the free-hanging elevator decreases and becomes at very high speeds a stalling moment. This is desirable from the standpoint of stability, as the free elevator works with the tail-plane to return the machine to its original attitude, and the force which must be exerted on the stick to fly the airplane at any speed other than its trimming speed is thereby increased. If the elevators were flat above and cambered below the condition would be reversed, and the lift for constant hinge moment would change in a manner disadvantageous for stability.

In the case of an airplane where the elevator is hinged to the rear of a tail-plane it is only possible to reason by analogy from the simpler type of tail surface just discussed. The relationship between the moment and lift coefficients is now dependent in a rather indeterminate manner on the angle of attack. In general, however, it is sufficient for stability that the curve of pitching moments should have a negative slope at all points when tested without the elevators and that a curve of coefficients of moment about the elevator hinge plotted against lift coefficient should have a negative slope at all points for all angles of the tail-plane. If the first of these conditions is observed both with throttle open and with throttle closed, the airplane will be stable with free controls under both of these conditions of operation. These specifications are not absolutely rigorous, as the force on the tail-plane and its effect on stability are somewhat affected by the presence of the elevator, especially if the elevator is a heavy one. The efficiency of the tail is, as already noted, greatest when the elevator is set at a considerable angle to the tail-plane. A heavy elevator, which hangs down below the line of the relative wind and which requires that the tail-plane be set at a larger negative angle to maintain equilibrium at any given speed than would be necessary with a lighter control member, offers some advantage in this respect. Other things being equal, and neglecting the direct effect of C. G. position on stability, a tail-heavy airplane would have a more "stable" curve of stick forces than would one properly balanced, as the elevator has to be pulled down to preserve equilibrium on the tail-heavy machine and this increases the efficiency of the tail-plane. To secure a true measure of the effect of a change in C. G. position the tail-plane should be adjusted, after the change, to such an angle that the airplane will trim with free controls at the same speed as before, and it should be found, if this is done, that there nearly always is an improvement of stability by moving the C. G. forward, the exceptions being machines with very small tail-planes.

Another reason, in addition to that just mentioned, for the increasingly stable form of the stick force curve as the negative angle of tail-plane setting is increased, is that a given change of setting means a change of lift coefficient for the tail-plane which is approximately the same for all angles of attack. The total stalling moment due to the change is then proportional to the square of the speed, and the additional upward force on the elevator and decrease (algebraic) of stick force necessary to produce a diving moment to balance this stalling moment is accordingly greater at high speed than at low. The effect therefore is to move the stick force curve downward, as a whole, but also to tilt it so that negative slopes are increased, positive slopes decreased. This phenomenon was discussed in Report No. 70, already referred to on several occasions, in connection with the testing of a DH-4 with several different settings of the adjustable tail-plane. The condition connecting L_c and M_c for the elevator is observed on the BE-2A[5], BE-2C[6], and the JN-2[7], the only machines for which hinge moment tests are available.

[5] Report of British Advisory Committee for Aeronautics, 1912-13, Rep. No. 74, p. 123.

[6] Full Scale Experiment on the Moment about the Hinge of the Air Forces on an Elevator; British Advisory Committee for Aeronautics, R. & M. No. 284, 1916.

[7] Bulletin of the Airplane Engineering Department, U. S. A.: Dec., 1918.

In fact, any elevator which did not have a lift-moment curve with a negative slope at all points would be overbalanced. The chief deduction to be drawn from this analysis is that models should be tested for stability in the wind tunnel with the elevators removed, and that, if stability with free controls is desired, the tail-plane should be large enough so that the curve of pitching moments from such a test will have a negative slope at all points, both with and without the slipstream effect. This points directly to the advantage to be gained by the use of a large tail-plane and small elevators. If possible, the tail-plane and elevator should be of such sectional form that there is a downward force on the elevator when the moment about the hinge is zero.

Heavy elevators are to be avoided for several reasons, chief among which is the effect of accelerations on the stick force required. To give a concrete instance, the pull on the stick required to balance the weight of the elevators in a JN is 8½ pounds. In pulling out of a loop with an acceleration of 3g, the stick force would be 25 pounds, even if there were no air load at all on the elevator. In the VE–7 the pull under the same conditions would be only about 8 pounds. A heavy elevator increases both the natural period of oscillation of the airplane with free controls and the damping of the motion, as the accelerations of the elevator turn it down during the lower part of an oscillation, up during the upper part, always moving so as to oppose the existing pitching motion of the airplane.

Before passing on to the discussion of experiments on stick forces something should be said with regard to balanced controls. Overbalance may be defined as the condition in which the curve of coefficient of hinge moment against lift coefficient for the elevator has a positive slope at some points for some tail-plane settings, and it is quite possible that some types of elevator may be overbalanced when hinged at the leading edge, although such a state of affairs would be rare. If an elevator is much overbalanced the airplane is usually unpleasant, although not necessarily dangerous, to fly. Curiously enough, the best stability with free controls if the elevator hinges are too far back is obtained if the airplane is extremely deficient in stability with locked controls. If the machine is statically unstable when tested without the elevators there must be an upward force on those members at speeds below, a downward force at those above, the equilibrium speed. The elevator being overbalanced at all speeds, this gives a push on the stick at high speeds and a pull at low. Actual flight with free controls would hardly be possible, however, as the stick has no equilibrium position when the controls are overbalanced, but moves quickly to one or the other of its extreme limits of travel as soon as released. Flight with free controls would be possible only if the elevators were fitted with stops confining their oscillations between very narrow limits.

Intentional and extreme overbalancing forms the basis of the "automatic rudders" invented by Col. Crocco. The "automatic rudder" consists of a tail plane hinged at the rear and with the leading edge free to move vertically but restrained by springs. If the aircraft noses down the top load on the tail plane is increased and the leading edge moves downward, still further increasing the downward force on the tail plane and the righting moment derived therefrom. The efficiency of the tail plane as a stabilizing factor can be trebled or quadrupled in this way. This device has been successfully employed on some Italian airships, and it theoretically is equally applicable to airplanes, but it would probably be rendered unsatisfactory in service by excessive vibration of the tail plane and because of the relatively short natural periods of oscillation of an airplane. Overbalanced surfaces of any sort should in general be avoided at all costs.

EXPERIMENTS ON STABILITY WITH FREE CONTROLS.

The methods of conducting these experiments were explained in report No. 70. The results obtained with free controls are relatively more accurate than those with locked controls, largely because no communication between pilot and observer is required and because the personal equation of only one individual enters into the result. Also, more data have been obtained with free controls because stick-force measurements can be made on any machine without making the slightest change or installing any special equipment, and such measurements were therefore made on several airplanes on which there was no opportunity to install an angle indicator. In all the tests on the JN the machine was piloted by Mr. R. G. Miller.

The tests made on the JN4H included a series dealing systematically with the effects of C. G. position and tail-plane setting. The curves for three different C. G. positions and a tail-plane angle of $-2.4°$ to the top longerons are plotted, both for open and for closed throttle, in figure 5. It will be observed that, as prophesied from the theory, the machine is most stable with the C. G. back when the throttle is open. The position of the C. G. with the throttle closed seems to have very little effect, much less than would be expected.

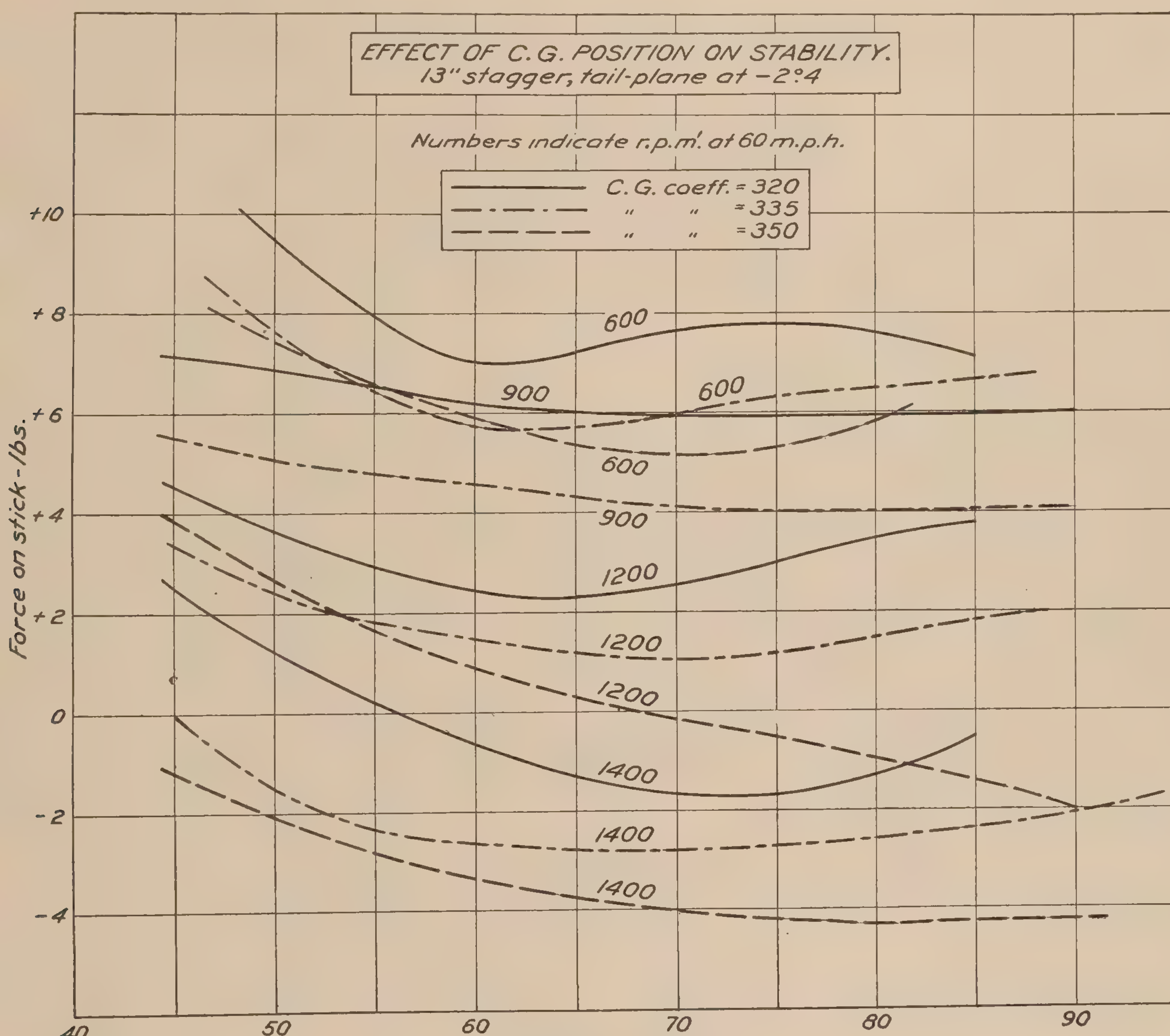


Fig. 5.

The effect of tail-plane setting is shown in figure 6, where the curves for three different settings with a constant C. G. position are given. The curves show that the stability is much better with the tail plane at $-2.4°$ to the top longerons than with it set at $-1.4°$, and that a further increase of angle of setting to $-2.9°$ produced still further improvement when gliding, but had comparatively little effect when the throttle was more than half opened. Apparently the most efficient camber for the tail as a whole is nearly if not quite reached when the tail plane is set at $-2.4°$ and the elevators are pulled down enough to balance the machine with the C. G. 35 per cent of the way back on the mean chord.

The next group of tests dealt, as in the case of locked controls, with the effect of sectional form of the tail. It is rather difficult entirely to separate the effect of sectional form from such complicating factors as angle of setting. It is obvious that data for tails of different types can not be made directly comparable by simply setting the tail planes in all cases with their chords at the same angle to the wings. The best means of obtaining a comparison appears

to be to set the several tails at such angles that the force on the stick at economical speed will be the same in all cases. This has been done approximately for the standard, the inverted, and the symmetrical tails in figure 7. The curves show, as was deduced from theory, that the cambering of the lower surface of the tail increases the stability at high speed while decreasing that at low. The range of stability is not increased, but the curve is flattened.

Experiments on the effect of the vertical coordinate of the C. G. were not carried far enough to be conclusive. Theoretically, lowering the C. G. relative to the thrust line should decrease the effect of opening the throttle and should increase the stability, since the thrust is largest at low speeds and a lowering of the C. G. produces the largest additional diving moment and re-

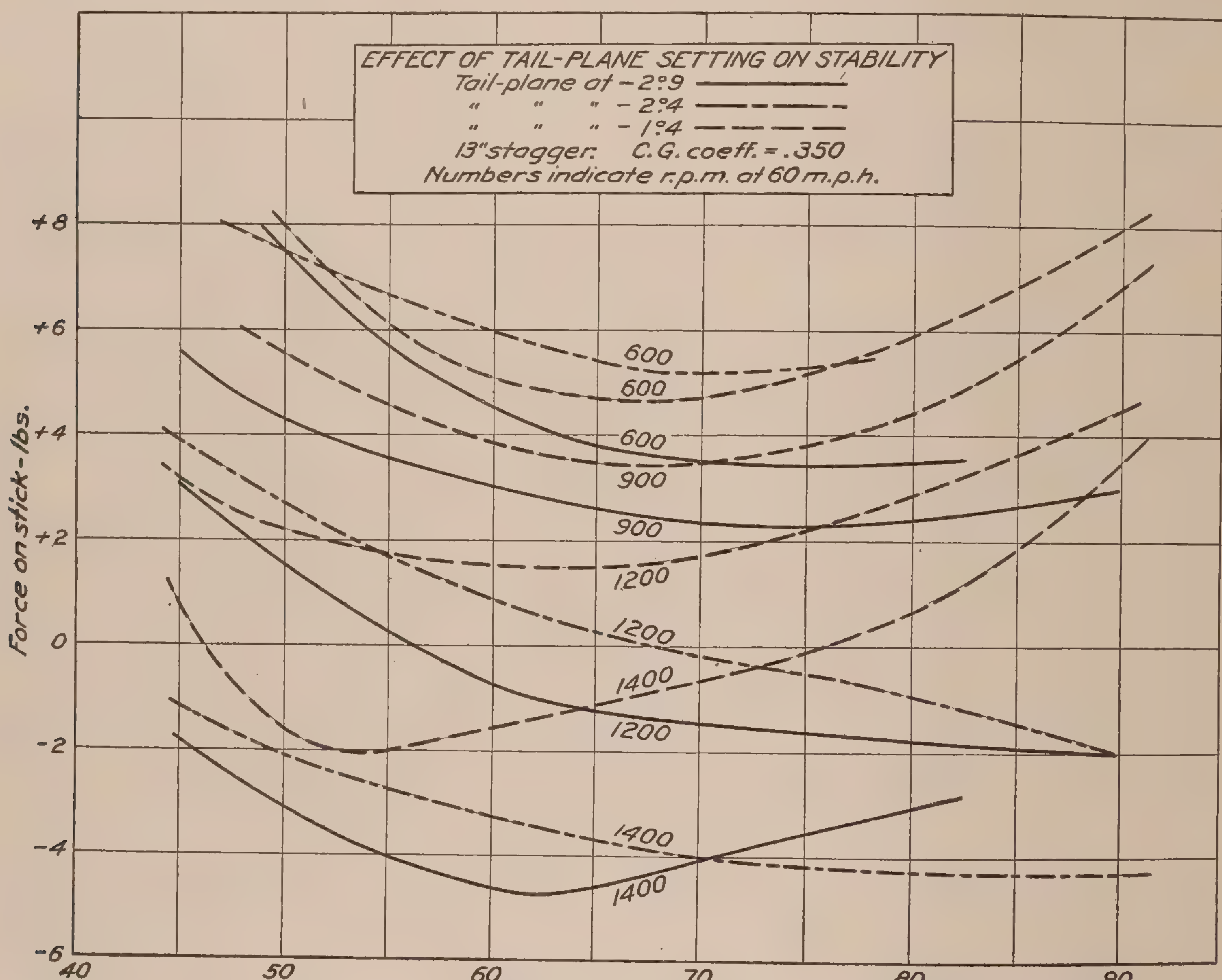


Fig. 6.

quires the largest additional pull on the stick under those conditions. Actually, however, neither of these effects appeared when the C. G. was lowered about an inch by the attachment of 50 pounds of lead to the axle. The propeller thrust on a JN4H at 60 m. p. h. is 470 pounds (calculated from a wind-tunnel test of the propeller). A lowering of the C. G. by 1 foot then produces a diving moment of 470 pounds-feet. Then, assuming the center of pressure of the tail to be distant 18 feet from the C. G., the down load on the tail is increased by 26 pounds. Part of this additional load comes on the tail-plane, as the pulling up of the elevator "banks up the air" on the tail-plane and increases its lift coefficient. Assuming that the additional force is equally distributed between the fixed and movable portions of the surface, and that the center of pressure of the elevator alone lies at 32 per cent of its chord behind its leading edge, this being the value determined in wind-tunnel tests on the pressure distribution on a JN tail,[8] the change in moment about the elevator hinge due to lowering the C. G. by 1 foot would be 139

[8] Bulletin of the Airplane Engineering Department, U. S. A., December, 1918, p. 38.

pounds-inches, corresponding to a change in stick force of 5.7 pounds. Since the mean separation between the curves for open and closed throttle at 60 m. p. h. is 8.1 pounds it would theoretically be necessary to lower the C. G. by 1.4 feet in order to bring the curves to coincidence. This would manifestly be impossible without a complete change in the type of the airplane.

Another possible method, suggested by Mr. F. H. Norton, for reducing slip-stream effect on the controls is to tip the engine down at the front so that the slip-stream makes a smaller angle with the tail-plane than does the relative wind when there is no slip-stream. This was tried out by placing tapered blocks between the engine and its bearers so as to incline the thrust line at 2° to the top longerons. Some improvement resulted from this change, but the gain

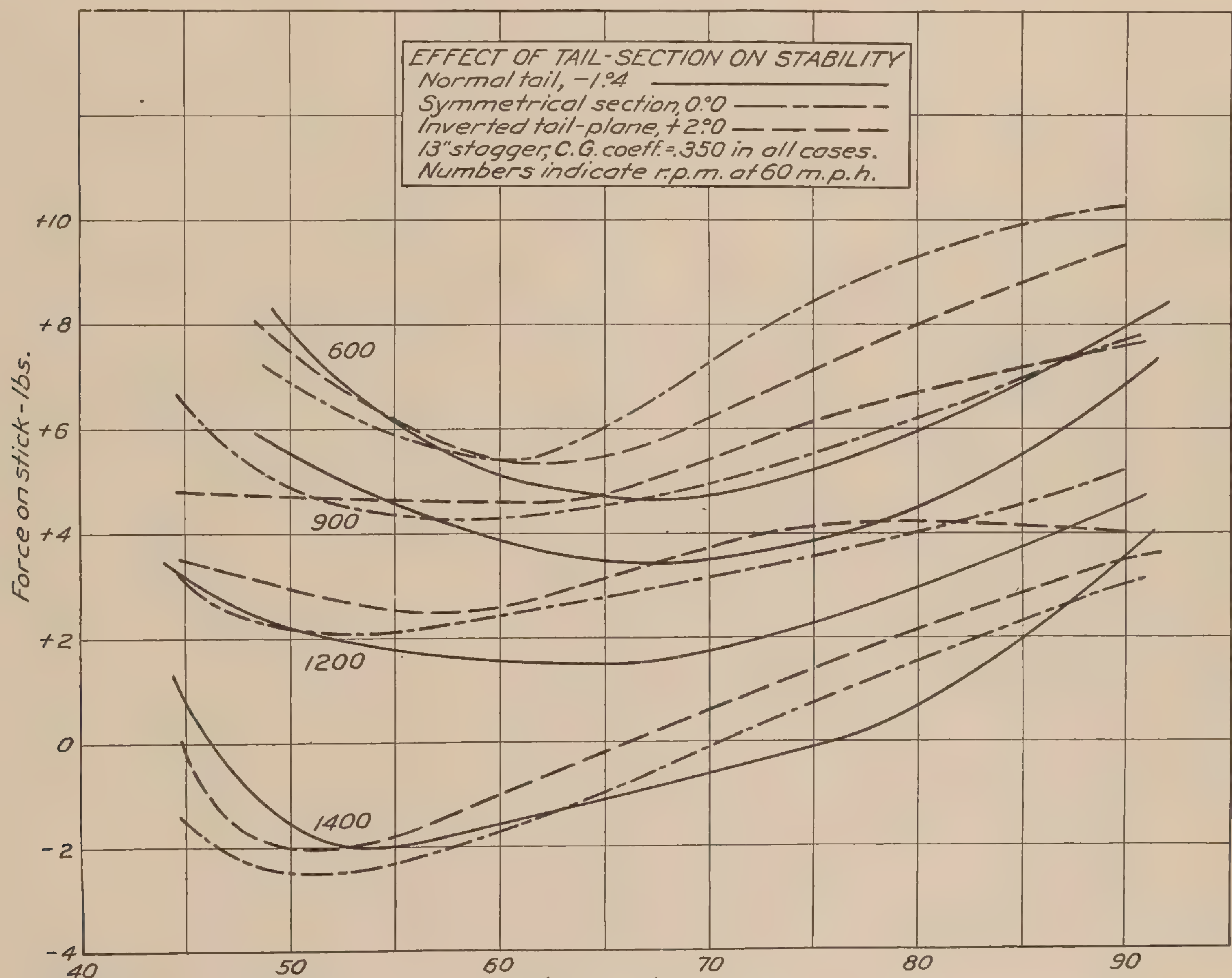


Fig. 7.

was not marked enough to justify the recommendation of such an inclination of the engine as a regular feature of design. An inclination large enough to be of much use in neutralizing the slip-stream effect on the controls would be distinctly detrimental to efficiency at maximum speed. The best way that has yet appeared to reduce slip-stream effect on a single-engined machine is to use a tail of large aspect ratio so that a considerable portion of it will lie outside of the slip-stream.

In closing the treatment of the experiments on the JN, as an indication of the net improvement of stability which has resulted from all this work, there are plotted in figure 8 the curves for the standard JN and for the best arrangement finally arrived at (6 inches stagger, tail-plane inverted and at −2° to the top longerons). It will be observed that there is a great improvement in stability, especially at high speeds, and that the danger of the stick force in a dive increasing to a point where it would be impossible to pull the machine out has entirely disappeared.

TESTS ON OTHER AIRPLANES.

In addition to the DH4, the results for which were discussed in report No. 70, stick-force determinations have been made, through the courtesy of the Airplane Engineering Department at McCook Field and particularly of Col. T. H. Bane and Lieut. Col. V. E. Clark, on the VE7 (Vought), U. S. A. C11 (Lepere Biplane), and Martin Transport. The assembly drawings of these three airplanes are reproduced in figures 9, 10, and 11. All three of these airplanes were flown during the tests by Lieut. H. R. Harris. The stick-force curves for the three machines

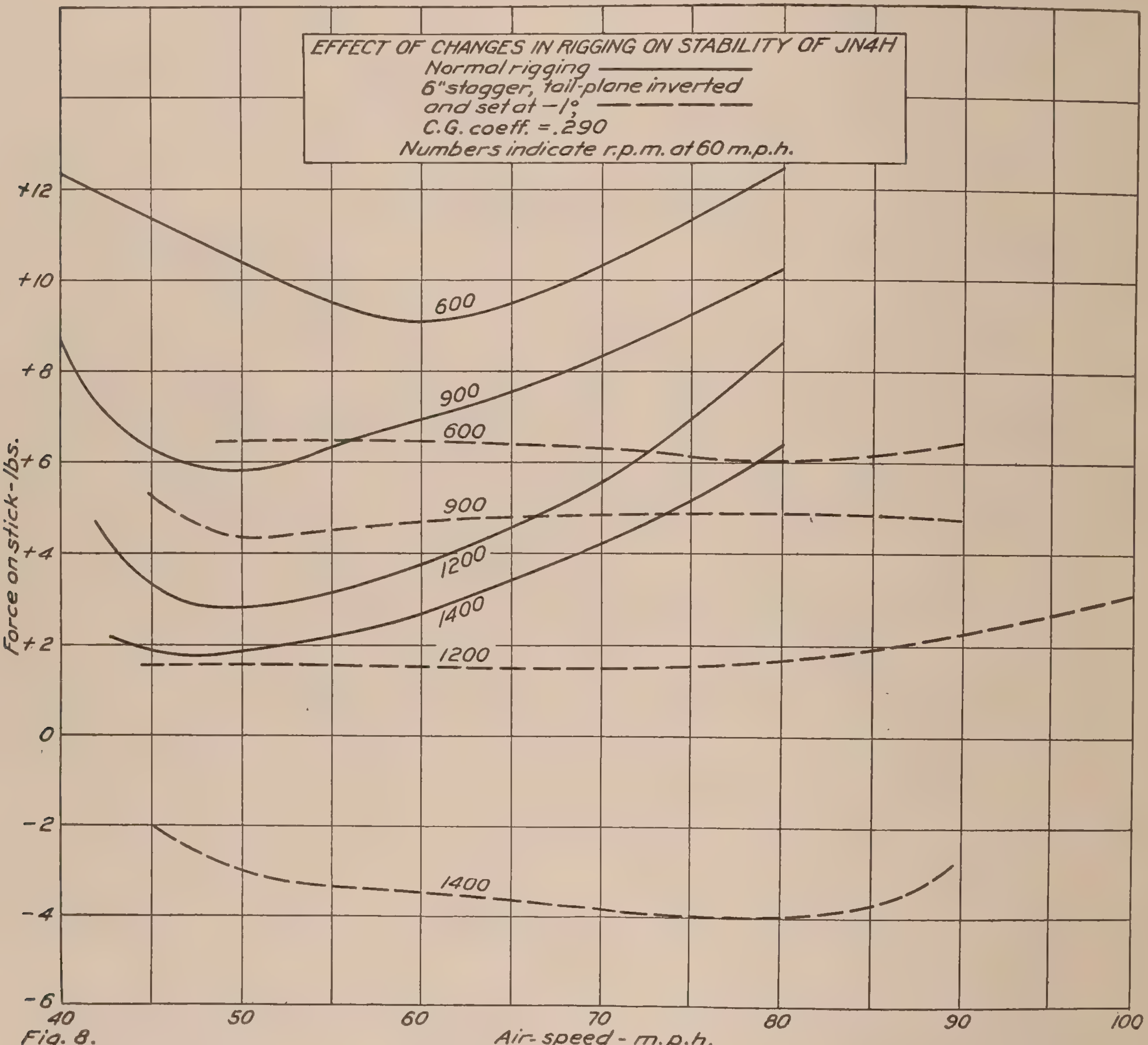


Fig. 8.

are given in figures 12, 13, and 14. The curves for the Martin must be regarded with some suspicion, as the friction in the control system (of the column type) was so great as to make it impossible to be sure of the forces within 2 or 3 pounds.

The stability of the VE7 is virtually ideal. This machine had the C. G. 30 per cent of the way back on the mean chord and one-half inch below the thrust line. The tail-plane is convex on both surfaces, the upper camber being about twice the lower. Comparisons between different machines show a remarkable divergency in the location of the point of maximum stability, and that location seems to be largely controlled by the section of the tail. The DH4 and Lepere have tails of virtually symmetrical section and are much more stable at high speeds than at low. The JN has all its camber on the upper surface and is much more stable at low speeds than at

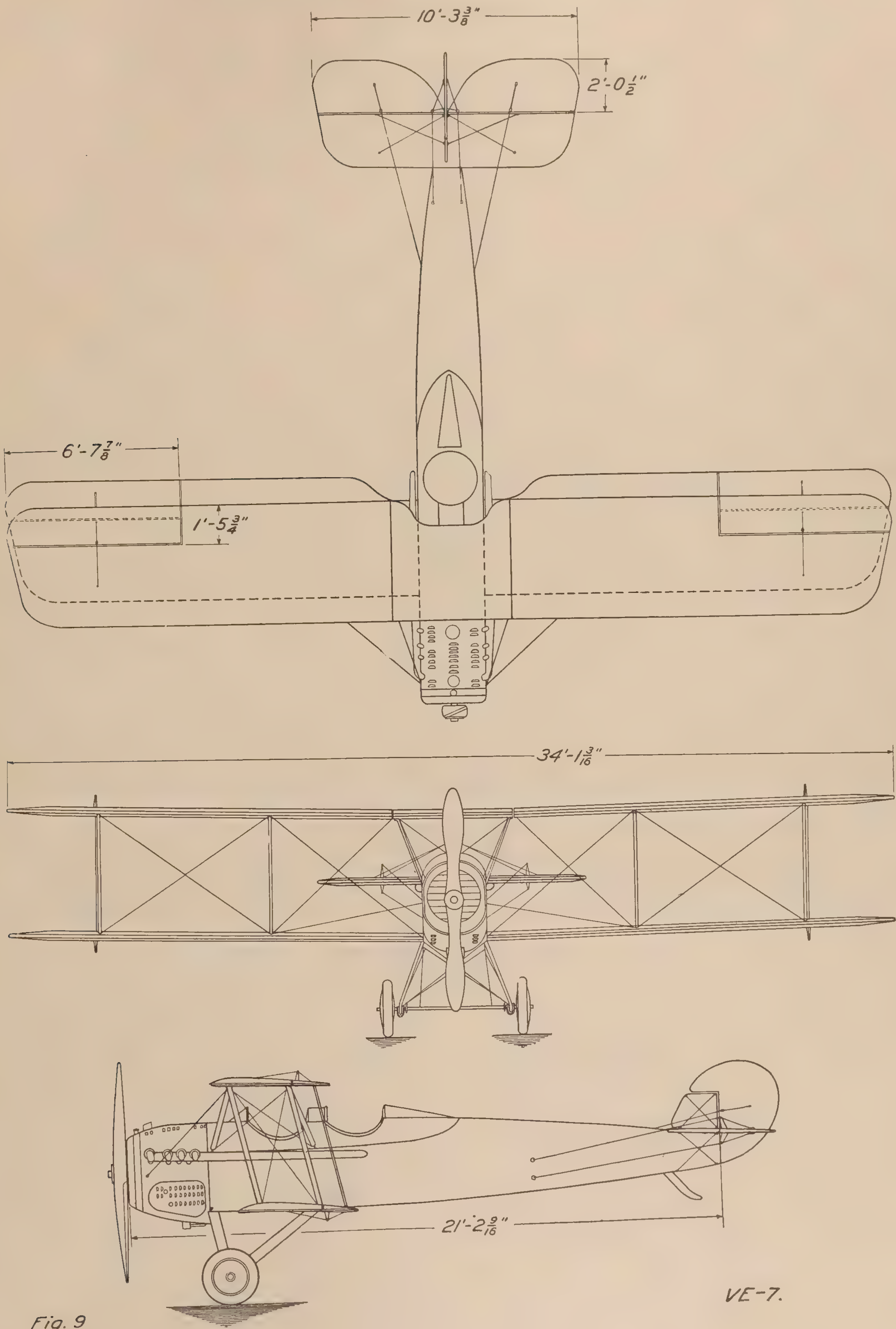


Fig. 9

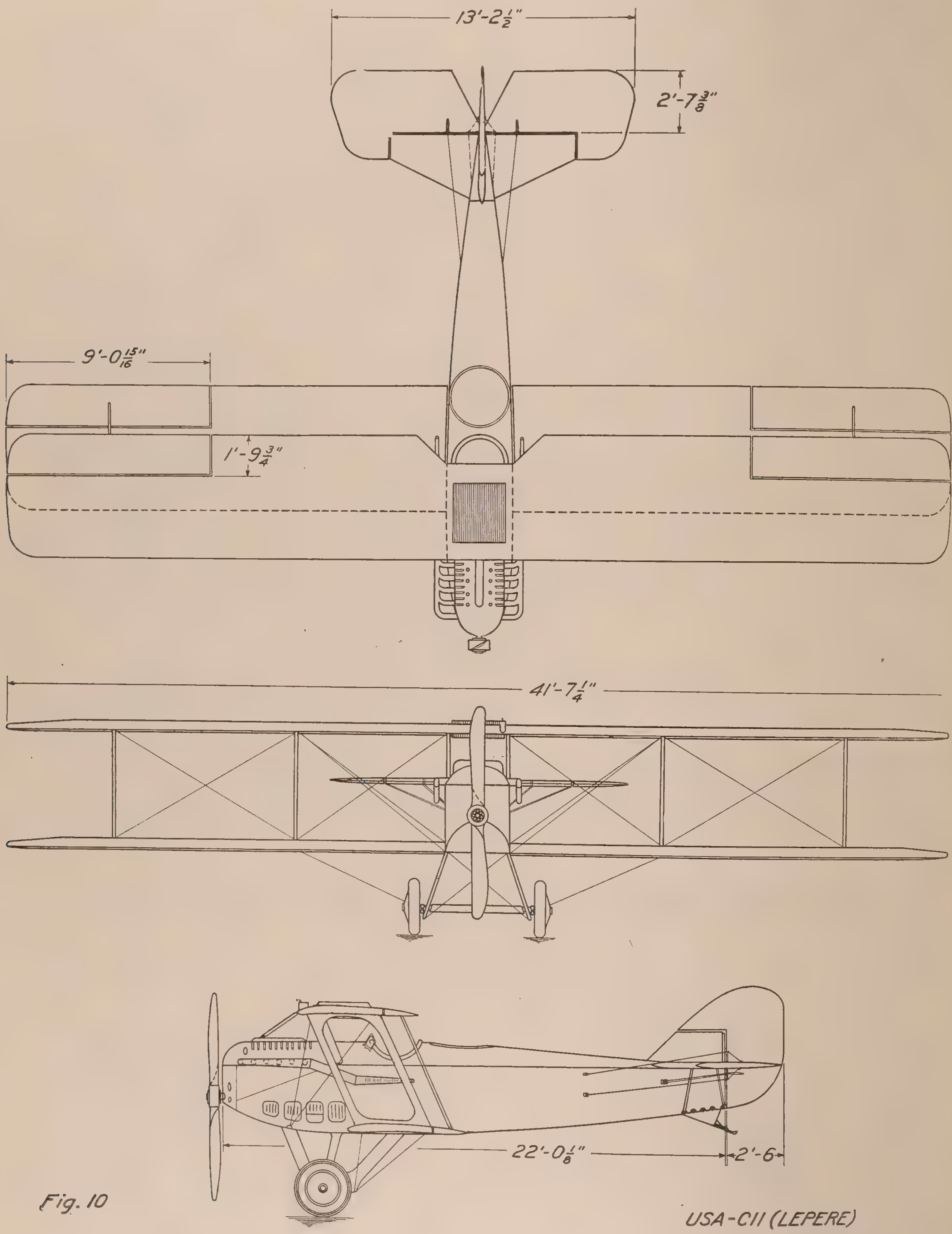


Fig. 10

USA-C11 (LEPERE)

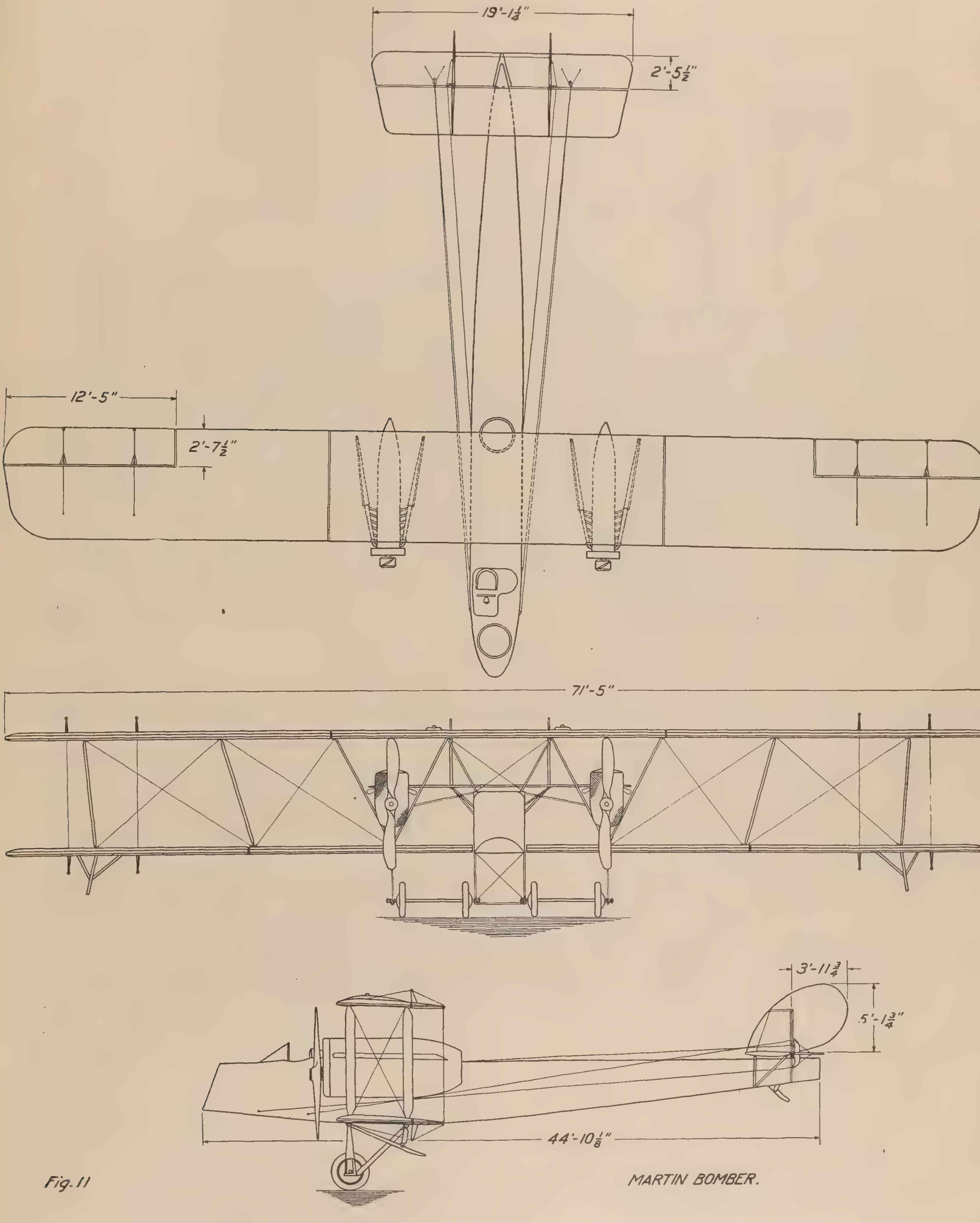


Fig. 11 MARTIN BOMBER.

high, while the VE7, which has twice as much camber on the upper surface as on the lower is equally stable at all speeds. That the section should exercise an influence in the general direction that it does is of course predictable from theory, but the magnitude of the effect found in comparing those four machines is much greater than would be expected either from theory or from the experiments on the effect of sectional form of the tail in the JN. At least it is possible to say definitely that the tail should not have a flat lower surface. All experiments and theories agree on that point.

The control surfaces both in the VE–7 and in the Lepere are much lighter than in the JN, a pull of only 2½ pounds on the stick being required to hold up the elevator on the VE–7 when at rest. In a loop or a tight spiral the pull required on the stick would then be about 18 pounds less on the Vought than on the JN, from this cause alone, and this factor contributes in no small degree to the remarkable controlability of the former machine.

The slip-stream effect on the controls still appears in the Martin, notwithstanding the fact that it is a twin-engined machine with the thrust line high relative to the C. G. It is probable, although direct experiments on the point have not yet been made, that the slip-streams on a twin-engined machine tend to approach each other and to draw along by viscous drag the air which lies between them, and that the portion of the tail which lies in the slip-streams is therefore actually larger than is usually assumed. Two possible methods of reducing slip-stream effect in a twin-engined airplane are to "toe in" the engines, setting them at an angle to the plane of symmetry so that the slip-streams will diverge and miss the tail, and to turn the propellers in opposite directions, the upper blade of each propeller moving away from the center line of the machine so that there is an upward component of race rotation in that portion of the slip-stream which strikes the tail, thus reducing or annulling the additional downward force due to the slip-stream.

The experiments on the Vought, Martin, and Lepere have been discussed at much greater length in Technical Note No. 1 of the National Advisory Committee for Aeronautics. Certain interesting points in connection with the balancing of the Lepere tail surfaces have been treated in that note and need not be repeated here.

FLIGHT TESTS WITH CONTROLS FREE.

Needless to say, the final test of stability with free controls is to release them while in flight and observe the subsequent motion, and this has been tried with two of the five machines for which stick force measurements were made. In order that lateral and longitudinal motions might be kept entirely separate, as in the case of locked controls, a short vertical stick was mounted directly on the longitudinal tube to which the regular control-stick is pivoted and which carries at its ends the sectors to which the aileron cables are attached. This secondary stick then permits the pilot to operate the ailerons without any possibility of affecting the elevator.

With 13 inches stagger and with the tail-plane set at from $-2°$ to $-3°$ to the top longerons the JN airplane would fly indefinitely with elevator control free for a small range of engine speeds. The factor limiting the range of r. p. m. was not the appearance of instability but the large separation of the force curves for different throttle settings. With the throttle closed there is a pull on the stick for all speeds at which it was considered safe to dive, and the airplane would therefore go into an approximately vertical dive, if not actually over on its back, if the

controls were released and left free for a long enough time with the throttle closed. From about 1,000 to 1,300 r. p. m., however, the flight was more steady than with locked controls and more steady than it could be held by the use of the controls by any pilots except those of the most exceptional skill. In fairly smooth air (not ideal, but not unduly bumpy) the elevator moved continuously through a total angular range of about 0.5°. Most of the trials were started by releasing the stick while the airplane was diving steadily at about 80 m. p. h. With the throttle wide open the nose began to come up at once and continued to rise until the longitudinal axis was vertical, at which time the pilot resumed control. The machine still had plenty of speed and it is possible that, if left to itself, it would complete a loop with free controls. With the throttle partly opened the nose rose to a definite point and then began to drop again, com-

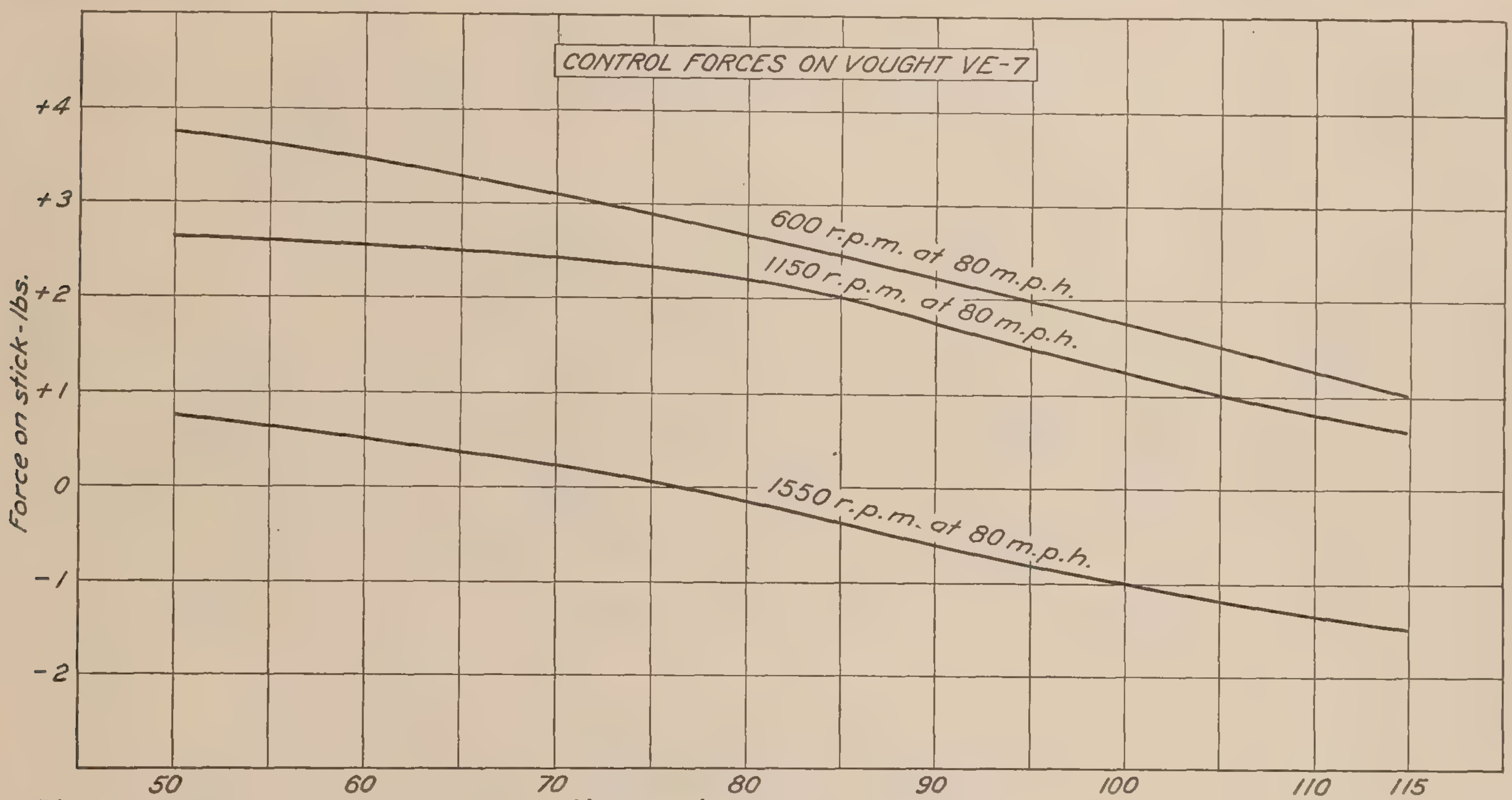
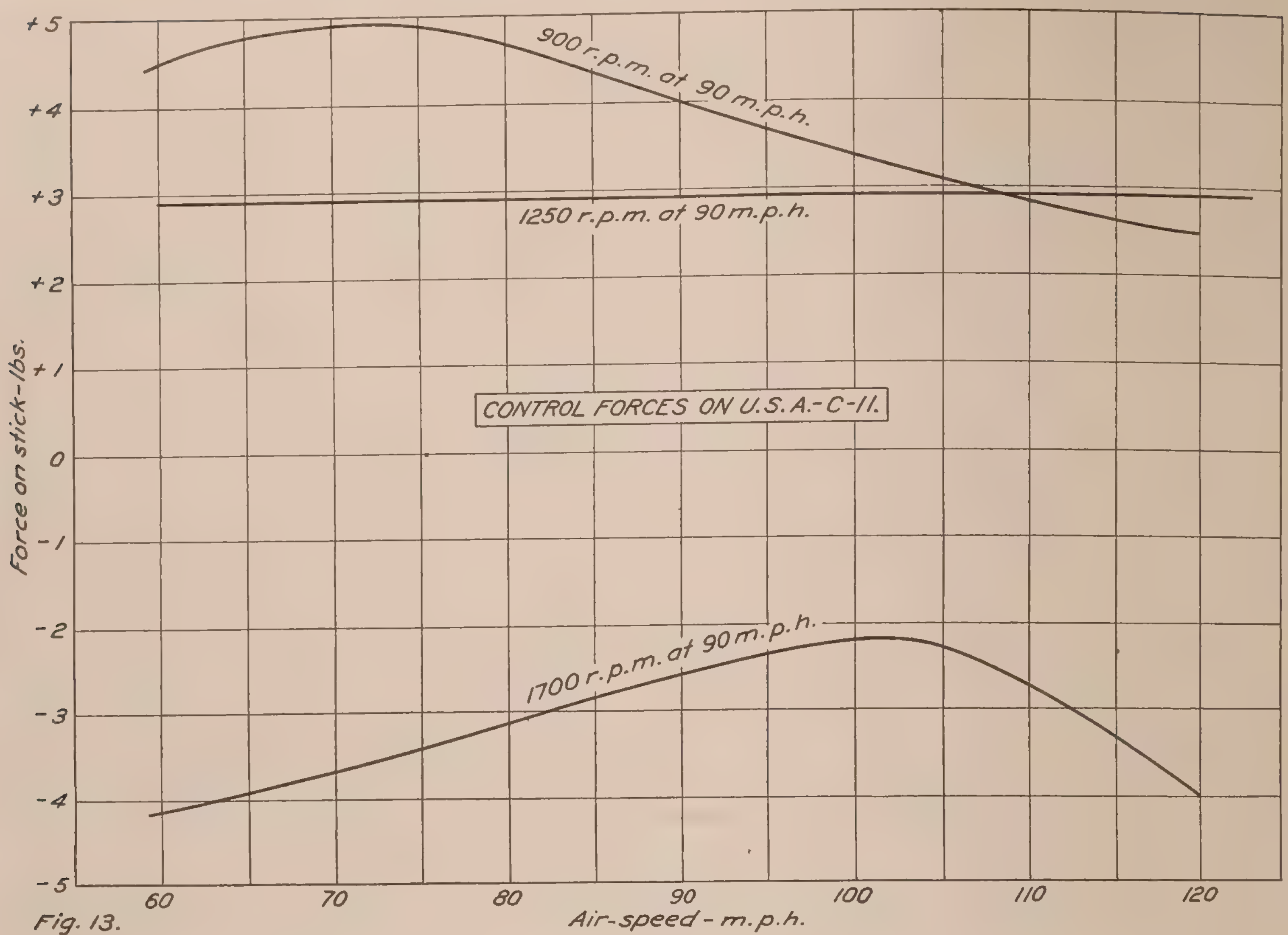


Fig. 12.

ing to an equilibrium position after two or three oscillations. The airplane was also flown in a circular path with angles of bank up to 15° and with the elevator control entirely free. The steadiness of flight when circling, although sufficient, was inferior to that with free controls.

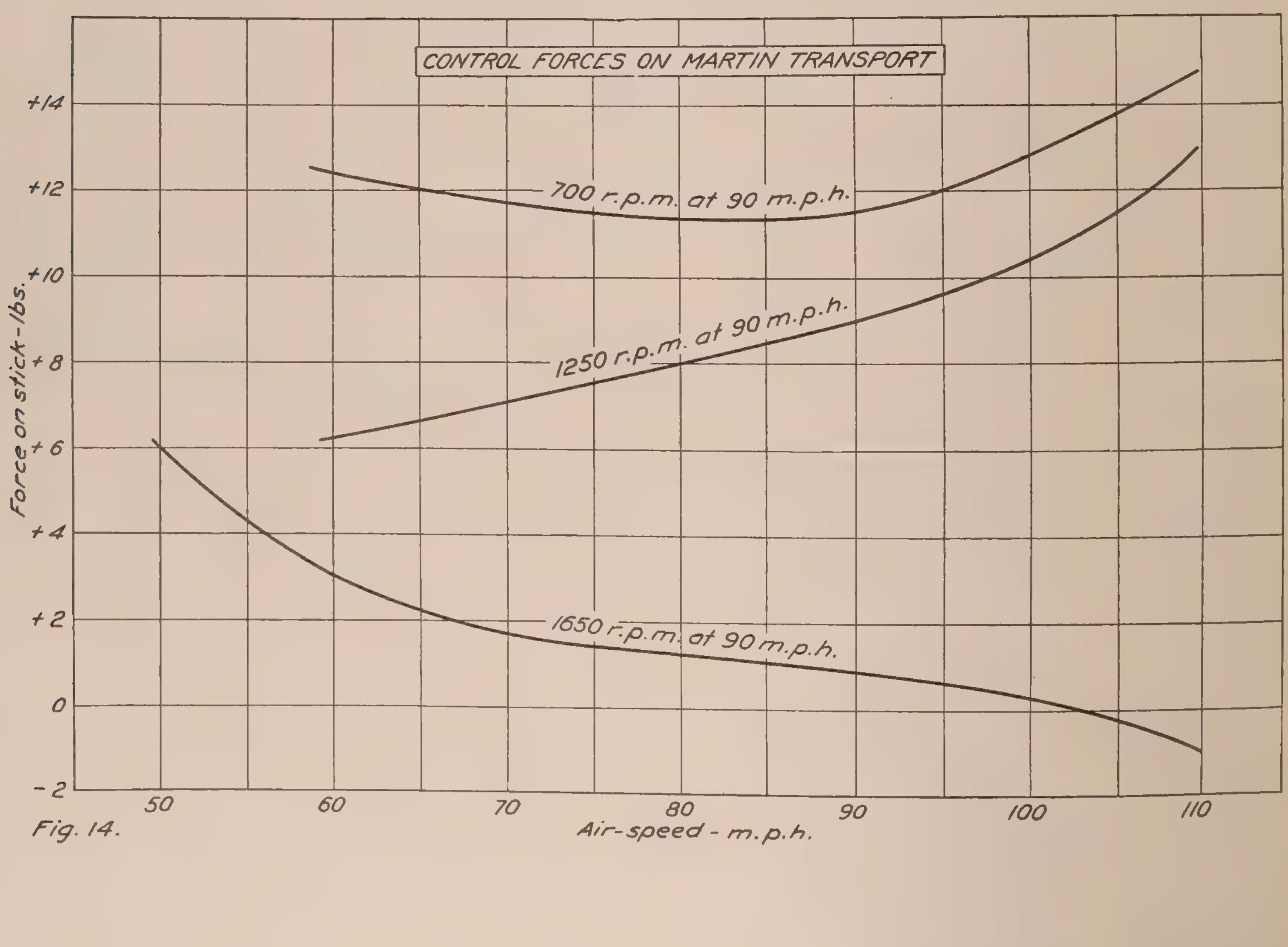
The subject of dynamical stability will be treated at length in a subsequent report, but a few observations will be noted here. The dynamical stability of the JN proved to be excellent, the oscillations being heavily damped except in a few instances. The periods measured ranged from 25 to 28 seconds, and the oscillations were by no means simple harmonic in form, the nose rising much more slowly than it dropped, and seeming to creep gradually up to the most stalled position, hang there for two or three seconds, and then drop abruptly.

The VE–7 was also flown with free controls and was also found to be very steady, although not quite so good as the JN in this respect. The period of oscillation was from 14 to 17 seconds, being shorter than on the JN chiefly because of the smaller moment of inertia.



CONTROL FORCES ON U.S.A.-C-11.
900 r.p.m. at 90 m.p.h.
1250 r.p.m. at 90 m.p.h.
1700 r.p.m. at 90 m.p.h.
Force on stick-lbs.
+5
+4
+3
+2
+1
0
-1
-2
-3
-4
-5
60
70
80
90
100
110
120
Air-speed - m.p.h.

Fig. 13.



CONTROL FORCES ON MARTIN TRANSPORT
700 r.p.m. at 90 m.p.h.
1250 r.p.m. at 90 m.p.h.
1650 r.p.m. at 90 m.p.h.
Force on stick-lbs.
+14
+12
+10
+8
+6
+4
+2
0
-2
50
60
70
80
90
100
110
Air-speed - m.p.h.

Fig. 14.

REPORT No. 97

GENERAL THEORY OF THE STEADY MOTION OF AN AIRPLANE

IN SEVEN PARTS

▽

By GEORGE DE BOTHEZAT
National Advisory Committee for Aeronautics

REPORT No. 97.

INTRODUCTION.

I hope it may be interesting to the reader to learn briefly, as it were, the history of the method here proposed for the study of steady motion, one which is different from other methods used. In his course of 1909–1910 at the "École Superieure d'Aeronautique," M. Paul Painlevé showed how convenient the drag-lift curve was for the study of airplane steady motion. His treatment of this subject can be found in "La Technique Aeronautique" No. 1, January 1, 1910. In my book "Etude de la stabilité de l'aéroplane," Paris, 1911, I had already added to the drag-lift curve, the curve I call speed curve, which permits a direct checking of the speed of the airplane under all flying conditions. But the speed curve was still plotted in the same quadrant as the drag-lift curve. Later, with the progressive development of the new aeronautical science, with the continual increasing knowledge about engines and propellers, when seeking a convenient method of airplane design that really took account of all the particulars of the subject, I was brought to add the three other quadrants to the original one quadrant, and thus was obtained the steady motion chart described in detail in this paper, a method which I have been using since 1914. This chart is the most convenient method I know for the complete representation of the airplane steady motion performance. This method allows an easy survey of all the mutual interrelations of all the quantities involved in the question and this is accomplished—the chart once plotted—without any computations or graphical tracings. The chart, therefore, permits one to read directly, for a given airplane, its horizontal speed at any altitude, its rate of climb at any altitude, its path inclination to the horizon at any moment, its ceiling, its propeller thrust, revolutions, efficiency, and power absorbed—that is, the complete set of quantities involved in the subject, and to follow the variations of all these quantities both for variable altitude and for variable throttle. At the same time, one can follow the variation of all of the above quantities in flight, as a function of the lift coefficient and of the speed. It is the possibility of doing this that constitutes the most important property of my steady motion chart and makes its use so convenient for any purpose or question connected with steady motion.

I have considered it necessary not to limit myself in this paper to the general exposure of the method proposed, but to give at the same time a general discussion of the main principles connected with the subject, about which so many misunderstandings are still widespread.

Thus, the question of the interreaction of the airplane and propeller through the slip stream will be found discussed here. Several authors have talked much about the great increase of airplane drag produced by the slip stream. The trouble is that the additional pressure on the airplane due to the slip stream is an interior force for the airplane system, and it thus can not be purely and simply added to the airplane drag, which in our statement of the problem is an exterior force. The way in which the momentum theorem is applied to the airplane must be well remembered in the present case. The airplane in flight is considered inclosed in a closed surface invariably connected to the airplane and it is the component along the flying speed of the fluid momentum that flows out of this surface that measures the drag of the whole airplane. But in the value of this momentum the additional pressure on the fuselage due to the slip stream and the additional thrust of the propeller, which is the direct reaction to the last additional pressure, appear with opposite signs, and thus only their difference affects the drag. I hope that those who will carefully follow the general treatment of this problem here given, will not have the slightest doubt about the real nature of the question.

The question of the properties of the engine-propeller system and its dependence upon the properties of the engine considered alone and of the propeller considered alone will be found treated here in the generality demanded by actual aeronautical engineering practice. When a given propeller is considered by itself, its characteristics are functions of the ratio of its translational speed to its revolutions. When an engine is considered by itself, its power characteristics are functions of the revolutions and throttle opening. But when a propeller is connected to a certain engine the propeller's revolutions have to adjust themselves to the translational speed of the engine-propeller system and its characteristics will be functions only of the translational speed and throttle opening.

These preliminaries to the study of airplane steady motion is completed by the discussion of the question of the standard atmosphere. It is the opinion of the author that this last question has, in general, been greatly misunderstood. The entire performance of an airplane depends upon the density and temperature of the air in which the airplane flight takes place. It is a property of the airplane to be able to reach a certain limiting atmospheric layer specified by a certain density, above which the airplane can not fly any more, which is called its ceiling. The altitude at which this atmospheric layer can be found is very variable with the meteorological conditions. Thus the airplane ceiling can not be specified by an altitude value, but only by a density value. The forces of air resistance depend only upon the density and are independent—in practical limits—of temperature; the lift, the drag, and propeller thrust depend only upon density; it is the power of the airplane engine alone that is affected by temperature. Thus at constant density only the engine power will be influenced by the temperature; and, when selecting a standard law connecting atmospheric temperatures with atmospheric densities, it is only the selection of standard working conditions for the engine that will be concerned. The temperature acts on the engine somewhat as a throttle variation. The last fact understood, it is clear that there is no reason for adopting a fantastic relation between temperature and densities for engine standard working conditions, and the adoption of a constant standard temperature for all densities becomes quite natural. It is in such a way that we are brought to the general conclusion that, for the standardization of airplane performance, it is the isothermic atmosphere that should be adopted. It is the proposition of the author to adopt the isothermic atmosphere of zero degrees centigrade as standard atmosphere. The tremendous advantages and great simplicity that result from such a selection will be found discussed in this paper. The isothermic atmosphere of zero degrees centigrade has also in its favor the fact that it satisfies all demands quite as well as any other "standard atmosphere." (See fig. 13.) The public has curiosity about the height at which an airplane is flying; but, from an engineering standpoint, we can only speak about the density reached by an airplane.

It is thus beyond discussion that, from the standpoint of aviation engineering, the isothermic atmosphere of zero degrees centigrade is the only one that can be reasonably adopted as the standard atmosphere.

For some special purposes we need to know the actual altitude at which an airplane is flying. But this is a totally different question, and no "standard atmosphere" can help us in such a case to obtain an accurate determination of the altitude. The question of the altitude determination from the knowledge of the atmospheric pressure and temperature is a special question in itself, totally independent of the conditions adopted for the standardization of airplane performances. The foregoing questions are discussed in the first three parts.

In Part IV the general theory of the steady motion of an airplane is developed. After the basic equations have been established and the method to be used for their discussion described, a general survey of the properties of an airplane in steady motion is given. I call attention to the detailed discussion of climbing phenomenon that will be found here and to the general formulae established for the rate of climb and time of climb, which quantities, under the simplest assumptions, appear as hyperbolic functions of the ceiling. It is also shown as a consequence of what conditions one can derive the law of linear variation of the rate of climb with altitude as practically observed. The influence of throttle variation on airplane per-

formance is also submitted to a detailed study and the influence of the mechanical losses of the engine on the airplane when gliding is discussed.

The complete study of the properties of an airplane in steady motion is made by the same uniform method, and the complete representation of the entire performance is reached. It is the last fact that constitutes the main advantage of the method developed.

In Part V is discussed the question of the first checking of airplane performances, starting with a minimum of data available concerning the airplane considered. This question is of great practical interest, but certainly the performance is predicted only as a first approximation.

Part VI gives the general outlines of the author's method of airplane free flight testing, which permits the most complete and rigorous airplane tests. The whole system of airplane characteristics, including the separate determination of the engine and propeller characteristics as given by free flights, is obtained from a set of climbs and glides made at constant indicated air speeds. The horizontal speeds at all altitudes, the best rates of climbs, and the ceiling are found with great accuracy without the pilot having to fly under these conditions, which practically can never be reached with complete certainty. On the contrary, the flying at nearly constant indicated air speeds can be realized by the pilots fairly well and with ease. That is why the present method of free flight testing is so convenient in practice.

A last part is devoted to the study of the problem of soaring. This question of soaring has been since long a matter of great interest and discussion. The phenomenon is a direct consequence of the existence in the atmosphere of ascending currents of air, and all other explanations of it are devoid of any serious foundation. Soaring is only possible if the upward vertical wind component is equal to or greater than the glider's rate of descent. Gliders of very small rate of descent can be built with ease; special attention has only to be paid to their stability and maneuverability. On the other hand, as is explained in this paper, it is the opinion of the author that ascending winds in the atmosphere must be considered as a common occurrence; this being a result of the instability of the vortex sheets formed between air layers of different velocities, and which must break into the Karman stable system of quincunx vortex rows. Between such vortices, traveling in space, we must meet at equal intervals ascending and descending currents. Direct computations show that the vertical components of these air currents are sensible fractions of the speed difference between the atmospheric layers which have originated these quincunx vortex rows. We are thus brought to a general understanding of the soaring phenomenon and the possibility of its practical utilization. The great interest of the practical realization of soaring airplanes is, I hope, beyond discussion.

At the end of this report is added a sheet of drawings giving a general survey of some fundamental characteristics of the atmosphere. I owe to the amiability of Dr. C. F. Marvin the remarkably complete data concerning the constitution of the atmosphere with altitude.

It is a special pleasure for me to address my best thanks to Mr. W. F. Gerhardt, aeronautical engineer at McCook Field, and to express my appreciation of the critical judgment he has shown in preparing most of the figures for this report. This last has given me the opportunity to discuss with him many details of this paper, which has helped me to clarify several of them.

Figure 13, relating to the computation of the standard atmospheres has been prepared by Mr. C. V. Johnson, aeronautical engineer at McCook Field, and I also address him my most sincere thanks for his kind assistance.

This paper has been written during my stay at McCook Field, when introducing my method of airplane free flight testing. I am specially pleased to have this opportunity to address my heartiest thanks to Maj. T. H. Bane, chief of McCook Field, for the interest he has always shown in my work and for all the necessary assistance he has placed at my disposal for its successful achievement.

G. DE BOTHEZAT,
Aerodynamical Expert, National Advisory Committee for Aeronautics.

DAYTON, OHIO, *July, 1920.*

REPORT No. 97.

GENERAL THEORY OF THE STEADY MOTION OF AN AIRPLANE.

By George de Bothezat.

PART I.

PRELIMINARY.

The following report on the Steady Motion of an Airplane was prepared by Dr. George de Bothezat, aerodynamical expert for the National Advisory Committee for Aeronautics, with the assistance of the technical staff and the approval of Major T. H. Bane, of the Engineering Division, Air Service of the Army, McCook Field, Dayton, Ohio.

Let us consider an airplane of any type or system, which, like most actual airplanes, has a plane of symmetry and constitutes a rigid system. By considering the airplane to be rigid, we only mean that we neglect the variations in the distribution of weight produced in the airplane by its small deformations and by the displacements of its rudders. The main influence of these last factors is to produce variations in the forces of air-resistance.

We will say that the airplane considered has reached one of its *states of steady motion* when the motion of the airplane proceeds with *a speed constant in magnitude and direction,* the plane of symmetry of the airplane being vertical, and the machine maintaining an invariable orientation relative to its rectilinear trajectory.

Let us consider the airplane moving in a mass of uniform air which in general may have the velocity $\bar{v}$ relative to the earth. The velocity $\bar{v}$ is the wind velocity in that part of the atmosphere where the airplane is actually flying.

The velocity of the airplane relative to the ground will be designated by $\bar{W}$ and called *ground speed* or *absolute speed,* because the earth can be considered, with sufficient approximation, as an absolute reference system in the present case.

The velocity of the airplane relative to the air mass containing it will be designated by $\bar{V}$ and called the *air-speed* or *self-speed.*

The velocities $\bar{v}$, $\bar{W}$ and $\bar{V}$ are vector quantities and are therefore characterized by their magnitudes, directions, and senses. Their magnitudes will be designated by v, W, and V. Between the velocities $\bar{v}$, $\bar{W}$, and $\bar{V}$ there always exists the relation:

$$\bar{W} = \bar{V} + \bar{v} \qquad \text{(geometrical sum)}$$

which expresses the fact that the airplane, so to say, flies in the wind with its self-speed $\bar{V}$ and is carried by the wind with the velocity $\bar{v}$. In case of no wind,

$$\bar{v} = 0; \quad \bar{W} = \bar{V}$$

The airplane will move with a self-speed of translation $\bar{V}$, constant in magnitude and direction, when all the forces acting on the airplane have a resultant equal to zero, and when the resulting moment of these forces, relative to the center of mass, are also equal to zero. The last conditions are direct consequences of the theorems of momentum and moments of momentum.

The forces acting on an airplane are: The weight, $\bar{P}$; the propeller thrust, $\bar{Q}$; the total air resistance, $\bar{R}$. The foregoing forces include all the forces acting on the airplane.

The first condition of steady motion of an airplane is expressed by the relation:

$$\bar{P} + \bar{Q} + \bar{R} = 0 \qquad \text{(geometrical sum)}$$

Let us designate by $\bar{M}$ the resulting moment of all the forces acting on the airplane. The second condition of steady motion of the airplane is expressed by the relation

$$\bar{M} = 0$$

As we consider only those motions of the airplane for which its plane of symmetry is vertical, the moment $\overline{M}$ is always normal to the plane of symmetry.

In the discussion of the conditions that make $\overline{M}=\overline{0}$, two cases must be distinguished:

The first case is when the thrust $\overline{Q}$ of the propeller passes through the center of mass of the airplane considered. In this case, as the weight P always passes through the center of mass, *the moment $\overline{M}$ reduces itself to the moment of the forces of air-resistance.* These last forces are proportional to the square of the self-speed $\overline{V}$ and the angle of attack α of the airplane, and for a given state of steady motion of the airplane can be changed only by the displacement of the elevator, the orientation of which will be supposed fixed by an angle β. We can thus write

$$M=mV^2\varphi(\alpha,\beta)$$

The angle of attack for which $M=0$ will thus be fixed by the condition:

$$\varphi(\alpha,\beta)=0$$

The function $\varphi(\alpha, \beta)$ in the flying interval must be a uniform function; thus to each value of β, i. e., for each position of the elevator, there must be a corresponding value of the angle of attack α for which $M=0$. The curve of α plotted against β can be called the curve of the elevator sensitivity. We are thus brought to the fundamental conclusion:

When the propeller thrust of an airplane passes through its center of mass—provided the action of the slipstream on the elevator can be neglected and the mass distribution considered as invariable—the angle of attack, for a state of steady motion of the airplane, can be changed only by displacement of the elevator. Any other conditions that can change in the flight can not alter the value of the angle of attack of the state of steady motion under consideration.

That is why I say that the angle of attack is the variable which the pilot is holding in his hand.

The second case is when the thrust $\overline{Q}$ of the propeller does not pass through the center of mass. This case is far more complicated than the first one. For a discussion of it, I will refer to my investigations of the question [1] and will mention here only the following: *In the case of the propeller decentration, a change in the angle of attack may be produced by acting on the throttle of the engine, as well as by changing the position of the elevator.*

I shall first give a general survey of the forces acting on the airplane. I shall afterwards deduce the consequences which follow from the condition that the resultant of the forces acting on an airplane is equal to zero when it has reached a state of steady motion. This will bring us to those fundamental references without which the understanding of airplane testing is impossible.

We shall use the metric units exclusively. Their use has been authorized in the U. S. Army by an act of Congress, and in practice tremendous advantages result from the use of these units.

We shall use the engineering metric units, i. e.,

kilogram-weight; meter; second

In these units, considering the gravitational acceleration as equal to $g=9, 81$ mt/sec^2, a body having a weight equal to 9.81 kg. has a mass equal to unity. For,

$$1 \textit{ kilogram-weight} = \textit{mass of a kg.} \times g.$$

and accordingly,

$$\textit{mass of a kilogram-weight} = \frac{1}{g}$$

Thus a body of g kilogram-weight will have a mass equal to unity. We shall call this last unit of mass the *Newton.*

[1] See Dr. G. de Bothezat's "Etude de la Stabilité de l'Aéroplane," Paris, 1911, p. 164, and "Revue de Mécanique, août, 1913." "Théorie Générale de l'Action Stabilisatrice des Empennages Horizontaux ...." Also, "Introduction to Airplane Stability," p. 137 (in Russian), Petrograd, 1912.

REPORT No. 97.

GENERAL THEORY OF THE STEADY MOTION OF AN AIRPLANE.

By George de Bothezat.

PART II.

THE FORCES ACTING ON AN AIRPLANE.

1. THE WEIGHT.

We shall designate by P the total normal *weight* that a given airplane is supposed to lift. The total weight of an airplane always acts vertically and passes through the center of mass of the airplane. The total normal weight is constituted of the following parts:

The structural weight of the airplane	P_a
The weight of the engine	P_m
The weight of the fuel	P_c
The useful weight	P_u

The sum of the first two constituent weights will be designated by P_{am}. We thus have:

$$P_{am} = P_a + P_m \tag{1}$$

The weight P_{am} is the minimum limit of the total weight of the airplane considered.

The sum of the last two constituents weight will be designated by P_{cu}. We thus have:

$$P_{cu} = P_c + P_u \tag{2}$$

The total normal weight is thus equal to

$$P = P_{am} + P_{cu} \tag{3}$$

For each airplane tested it it useful to note the value of the ratios:

$$P_a/P;\ P_m/P;\ P_c/P;\ P_u/P \text{ and } P_{am}/P = p_{am};\ P_{cu}/P = p_{cu}.$$

For large weight-carrying and low-ceiling airplanes, p_{cu} is close to 50 per cent, and for small high-speed and high-ceiling airplanes, p_{cu} is around 25 per cent.

2. THE FORCES OF AIR RESISTANCE.

We will resolve the total air resistance $\bar{R}$ of the whole airplane into two components:

The drag R_x directed along the self-speed $\bar{V}$ of the machine, but always in the inverse sense, and the lift R_y perpendicular to its direction. We have

$$R^2 = R_x^2 + R_y^2$$

All experimenters in aerodynamics fully agree that for the flying range of variation of the speed V, the drag and the lift can be considered as being of the form

$$R_x = k_x \delta A V^2 \tag{4}$$

$$R_y = k_y \delta A V^2 \tag{5}$$

where A is the area, δ is the air density (expressed in Newtons), and k_x and k_y are the drag and lift coefficients, which are functions of the angle of attack only. The angle of attack measured from any fixed reference line in the plane of symmetry of the airplane will be desig-

nated by α. The angle of attack measured from the *zero lift direction* will be designated by i. To a first approximation, for the flying range of variation of i, the coefficients k_x and k_y may be considered as being of the form

$$k_x = k(ai^2 + bi + c) \tag{6}$$

$$k_y = ki \tag{7}$$

The empirical coefficients k, a, b, and c have to be determined from the empirical curves for k_x and k_y by the method of least squares. Thus, to a first approximation, we may consider the drag R_x and the lift R_y as being of the form

$$R_x = k\delta A V^2(ai^2 + Ci + c) \tag{8}$$

$$R_y = k\delta A V^2 i \tag{9}$$

The air resistance $\bar{R}$ here considered, components of which are the drag R_x and the lift R_y, is the total air resistance of the whole airplane, the propeller or propulsive system excluded.

For all the fundamental conceptions relating to the laws of air resistance, the reader is referred to the author's "Introduction into the Study of the Laws of Air Resistance of Aerofoils," published by the National Advisory Committee for Aeronautics, Washington, D. C., Report No. 28.

3. THE PROPELLER THRUST.

In modern airplanes the propeller thrust is produced by a blade-screw propeller driven by a gas engine.

We shall call the system composed of the propeller and the engine the *engine-propeller system*. Its properties, which are a result of the combined properties of the propeller and engine used are, however, different from the properties of the propeller considered alone and of the engine considered alone.

I shall first give a short survey of those properties of the propeller and the engine, the knowledge of which is necessary for a complete understanding of the properties of the engine-propeller system.

A. PROPERTIES OF THE PROPELLER.

Let us consider a given propeller of a diameter D. When this propeller makes N revolutions per second, i. e., when it has the angular velocity $\Omega = 2\pi N$, and moves with the uniform velocity V *mt/sec.* along its axis, it will produce a thrust of Q *klg* when a torque of C *klg. mt.* is applied to its axis.

The thrust power L_u, or useful power developed by the propeller, is equal to

$$L_u = QV \tag{10}$$

The torque power L_a, or power absorbed by the propeller, is equal to

$$L_a = C\Omega \tag{11}$$

The efficiency of the propeller is equal to

$$\eta = \frac{L_u}{L_a} \tag{12}$$

We will designate by μ, and call it the *advance per turn*, or shorter, *advance*, the ratio

$$\mu = \frac{V}{N} \tag{13}$$

The thrust Q of a propeller has for its general expression

$$Q = \delta V^2 F_1(\mu) = \delta N^2 F''_1(\mu) \tag{14}$$

The torque power developed by a propeller has for its general expression

$$L_a = \delta V^3 F_2(\mu) = \delta N^3 F_2''(\mu) \tag{15}$$

In the last expressions, the quantities

$$F_1''(\mu) = \mu^2 F_1(\mu); \quad F_2''(\mu) = \mu^3 F_2(\mu)$$

are functions of the advance μ, only. These functions can be considered either as explicit functions which can be calculated from the screw dimensions[1] and its aerodynamical characteristics, or can be considered as empirical functions determined by direct experiment.

Using the values (14) and (15) for Q and L_a, it is easy to see that η has for its general expression

$$\eta = \frac{F_1(\mu)}{F_2(\mu)} \tag{16}$$

i. e., the efficiency η is a function of the advance μ only.

The thrust Q_0 produced and the power L_0 absorbed by a propeller working at a *fixed* point have for their general expressions

$$Q_0 = \delta N^2 C_1' \tag{17}$$

$$L_0 = \delta N^3 C_2' \tag{18}$$

where C_1' and C_2' are two constants that represent the limiting values which F_1'' and F_2'' take when V tends toward zero.

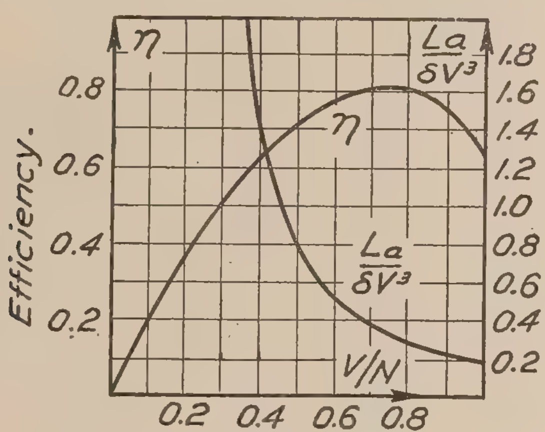


Fig. 1.

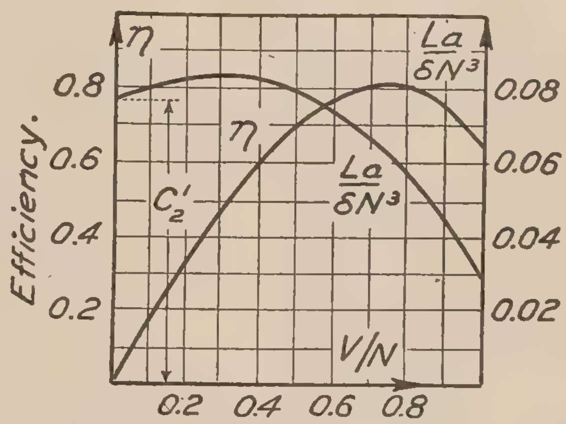


Fig. 2.

The deformation of the propeller blades and the deviation of the fluid resistance from the square law produce some departures from the foregoing laws. But these laws hold perfectly well when the variations of V and N are limited to certain intervals for which the constants appearing in expressions (14), (15), (16), (17), and (18) have been specially determined.

For the complete specification of the properties of a given propeller, two characteristic curves are necessary. We will use as such either

$$\eta(\mu) \text{ and } \frac{L_a}{\delta V^3} = F_2(\mu) \tag{19}$$

or

$$\eta(\mu) \text{ and } \frac{L_a}{\delta N^3} = F_2''(\mu) \tag{20}$$

which have the advance μ as argument.

The general courses of these curves are represented on figure 1 and figure 2, which correspond to a propeller of the Dorand type tested by G. Eiffel.[2]

All the foregoing refers to a propeller working in free air. In some cases different bodies disposed in the neighborhood of the propeller can interfere with the working of the propeller. As the neighborhood conditions require a slight generalization of the ordinary conceptions relating to propellers, I shall consider somewhat in detail the relations that hold in this case.

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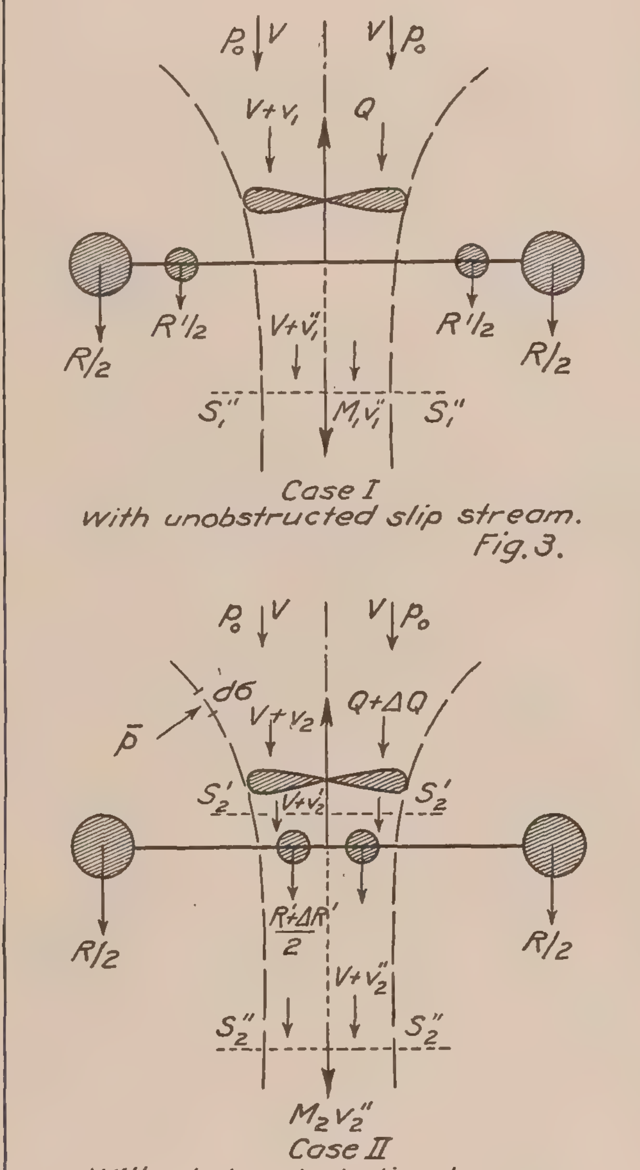
[1] For the explicit expressions of these functions, and methods of their calculations, see pp. 58 and 59 of "The General Theory of Blade Screws," by Dr. G. de Bothezat. Report No. 29, published by the National Advisory Committee for Aeronautics, Washington, D. C.

[2] G. Eiffel, "Nouvelles Recherches sur la Résistance de l'Air et l'Aviation," Paris, 1914. Atlas, plate XXXIII, propeller No. 11.

Let us consider any vehicle of locomotion, in our case an airplane flying under an angle of attack i and a speed V. In order to secure the flight of the airplane, i. e., to overcome the air resistance, it is necessary to supply a certain amount of power, which we will call power utilized L_u by the vehicle. For a given airplane with invariable load, at constant altitude and with motor throttle kept at constant opening, the power L_u is a function of the flying speed V only. This is because, as will be seen later, the angle of attack i under such conditions is a function of the speed V only. The power L_u is delivered to the vehicle by a propulsor, in our case a screw blade propeller. It is self-evident that the power delivered by the propulsor to the vehicle is the same thing as the *power utilized by the vehicle.* But in order to make the propulsor able to deliver the power L_u to the vehicle, we must always deliver to the propulsor a power L_a greater than L_u, called *power absorbed by the propulsor.*



Case I
With unobstructed slip stream.
Fig. 3.

Case II
With obstructed slip stream.
Fig. 3.

It is the ratio

$$\eta = \frac{L_u}{L_a} \tag{21}$$

which we shall call the efficiency of the *propulsor.*

It is easy to understand that the same propulsor applied to different vehicles will generally show different efficiencies on account of the neighborhood conditions interfering with the work of the propulsor. A propeller must be especially adapted to the vehicle under consideration in order to give a high efficiency. In order that we may have a complete understanding of the circumstances that here occur, let us compare the working conditions of two identical screw-blade propellers, applied to two airplanes, identical from the standpoint of air resistance, but in one case with an unobstructed slipstream and in the other case with some of the parts of the airplane disposed in the slipstream created by the propeller. These two cases, which we will call Case I and Case II, are represented schematically on Figure 3.

In Case I, when the airplane has reached a speed V, the total drag is equal to $R+R'$, where R' is the resistance of those parts which in Case II are in the slipstream, and the thrust is equal to Q. When a state of steady motion is reached, applying the momentum theorem *to the airplane,* we find

$$Q = R + R' \tag{22}$$

If we designate by L^{I}_a the power absorbed by the propulsor, the efficiency of the propulsor is equal to

$$\eta_1 = \frac{L_u}{L^{\mathrm{I}}_a} = \frac{(R+R')V}{L^{\mathrm{I}}_a} = \frac{QV}{L^{\mathrm{I}}_a} \tag{23}$$

Applying the momentum theorem to the slipstream, we find that the momentum $M_1 v_1''$ communicated to the fluid that crosses the propeller, measured in the section $S_1''S_1''$ where the exterior pressures on the boundary surface of the slipstream balance, is equal to

$$M_1 v_1'' = Q = R + R' \tag{24}$$

In Case II the bodies having the air resistance R' have been brought inside the slipstream. Their resistance changes then to $R'+\Delta R'$. When the same speed V is reached, the resistance of the bodies outside the slipstream will be unaffected, but the propeller thrust Q, on account of changed neighborhood conditions, will have been changed to a certain value equal to $Q+\Delta Q$. When a state of steady motion is reached, we have

$$Q+\Delta Q=R+R'+\Delta R' \tag{25}$$

and on account of (22)

$$\Delta Q=\Delta R' \tag{26}$$

Designating by L^{II}_a the power absorbed by the propeller in this second case, we have

$$\eta_2=\frac{L_u}{L_a^{\text{II}}}=\frac{(R+R')V}{L_a^{\text{II}}} \tag{27}$$

If in Case II we had measured the thrust of the propeller in free flight by a thrust meter we should have found for its value $Q+\Delta Q$. It is very tempting to take as a measure of the propeller efficiency the expression

$$\eta_2'=\frac{(Q+\Delta Q)V_1}{L_a^{\text{II}}} \tag{28}$$

but this, as it is easy to see, will give an overestimate of the propeller efficiency, because $(Q+\Delta Q)>(R+R')$. There is nothing astonishing in this, because it must not be forgotten that $(Q+\Delta Q)$ is in reality only an interior force in relation to our airplane system. It expresses only the stress state between engine-propeller set and airplane fuselage, and not the resultant exterior force securing the propulsion.

Let us calculate the momentum $M_2\, v_2''$ that crosses a section $S_2''\ S_2''$ of the slipstream taken behind the body of resistance R'. In this case we can not in general assume that the exterior pressure on the outside boundary surface of the slipstream, counted up to the section $S_2''\ S_2''$, balances, and will therefore designate by $\Sigma''\ \bar{p}\ d\sigma$ the resultant of this exterior pressure which on account of symmetry is necessarily directed along the slipstream axis. In the last expression $d\sigma$ is an element of the slipstream boundary surface counted up to section $S''_2\ S''_2$ and $\bar{p}$ the outside vector pressure considered normal to each corresponding surface element $d\sigma$. Applying the momentum theorem to the slipstream counted up to the section $S_2''\ S_2''$ we find:

$$M_2\, v_2''=Q+\Delta Q-(R'+\Delta R')+\Sigma''\ p\ d\sigma \tag{29}$$

or, since $\Delta Q=\Delta R'$ and $Q=R+R'$

$$M_2\, v''_2=R+\Sigma''\ \bar{p}\ d\sigma \tag{30}$$

In general $\Sigma''\ \bar{p}\ d\sigma<R'$, thus $M_2\, v_2''<M_1\, v_1''$.

It is natural to try to find out in what relation the power L^{II}_a stands to the power L^{I}_a. The whole thing depends upon the values of the efficiencies η_2 and η_1. We have

$$L_u=\eta_1\, L^{I}_a=\eta_2\, L^{II}_a \tag{31}$$

thus,

$$\frac{L^{I}_a}{L^{II}_a}=\frac{\eta_2}{\eta_2} \tag{32}$$

I immediately remark that by no means is it necessarily true that $\eta_2<\eta_1$ and it can even happen that for a given propeller we may get $\eta_2>\eta_1$. An examination of the comparative losses that take place in both cases will show the nature of the question.

Applying in Case I the theorem of kinetic energy to the slipstream, we find:

$$\tfrac{1}{2}\, M_1 (V+v_1'')^2 + \tfrac{1}{2}\, I_1''\, \omega_1''^2 - \tfrac{1}{2}\, M_1\, V^2 = Q(V+v_1) + C_1 \omega_1$$

or, since $Q = M_1\, v_1''$

$$Q(V+v_1) + C_1\, \omega_1 = VQ + \tfrac{1}{2}\, M_1\, v_1''^2 + \tfrac{1}{2}\, I_1''\, \omega_1''^2 \qquad (33)$$

where:

M_1 = fluid mass that crosses the propeller disk in a unit of time.
v_1'' = slip velocity in section $S_1''\ S_1''$.
I_1'' = moment of inertia of the fluid mass M_1 in section $S_1''\ S_1''$.
ω_1'' = race rotation in section $S_1''\ S_1''$.
v_1 = slip velocity in the plane of propeller rotation.
C_1 = torque acting on the propeller axis.
ω_1 = race rotation in the plane of the propeller.

The flow conditions in the slipstream are assumed uniform for sake of simplicity.

On the other hand we have,

$$L_a^I = C_1\, \Omega_1 = Q(V+v_1) + C_1\, \omega_1 + p_r^I \qquad (34)$$

where

Ω_1 = angular velocity of propeller rotation.
p_r^I = losses by impact and friction of the fluid against the propeller blades.

We thus finally find:

$$L_a^I = VQ + \tfrac{1}{2}\, M^1\, v_1''^2 + \tfrac{1}{2}\, I_1''\, \omega_1''^2 + p_r^I \qquad (35)$$

and

$$\eta_1 = \frac{L_u}{L_a} = \frac{L_a^I - [\tfrac{1}{2}\, M_1\, v_1''^2 + \tfrac{1}{2}\, I_1''\, \omega_1''^2 + p_r^I}{L_a^I} \qquad (36)$$

Applying in Case II the theorem of kinetic energy, we find:

$$\tfrac{1}{2}\, M_2 (V+v_2'')^2 + \tfrac{1}{2}\, I_2''\, \omega_2'' - \tfrac{1}{2}\, M_2\, V^2 - S_2''(V+v_2'')\, (p_0 - p_2'') + (Q+\Delta Q)\ (V+v_2) + C_2\, \omega_2 - (R'+\Delta R')\ (V+v_2')$$

or since $Q+\Delta Q = M_2\, v_2'' + (R'+\Delta R') - \Sigma''\, \overline{p}\, d\sigma$ and considering $\Sigma_1''\ p\ d\sigma \cong S_2''\ (p_o - p'')$ where S_2'' is the area of the section $S_2''\, S_2''$ of the slipstream, p_0 the outside pressure, and p_2'' the pressure in section $S_2''\ S_2''$ we get:

$$(Q+\Delta Q)\ (V+v_2) + C_2\, \omega_2 = V(Q+\Delta Q) + \tfrac{1}{2}\, M_2\, v_2'' + \tfrac{1}{2}\, I_2''\, \omega_2''^2 - S_2''(p_0 - p_2'')v_2'' + (R'+\Delta R')v_2' \qquad (37)$$

where M_2, v_2'', I_2'', ω_2'', C_2, ω_2 have meanings analogous to Case I; $S_2''\ (V+v_2'')\ (p_0 - p_2'')$ represents the work of the resultant exterior pressure $\Sigma_1''\ \overline{p}\ d\sigma$ considered as built up from the work of the pressures in a section far in front of the propeller and in section $S_2''\ S_2''$, where p_0 and p_2'' are the corresponding pressures; $(V+v_2')$, a mean velocity included between $(V+v_2)$ and $(V+v_2'')$ whose product by $(R'+\Delta R')$ represents the work corresponding to that resistance.

But as:

$$L_a^{II} = C_2\, \Omega_2 = (Q+\Delta Q)\ (V+v_2) + C_2\, \omega_2 + p_r^{II} \qquad (38)$$

(with Ω_2 and p_r^{II} having meanings analogous to Case I) we finally find:

$$L_a^{II} = V\ (Q+\Delta Q) + \tfrac{1}{2}\, M_2\, v_2''^2 + \tfrac{1}{2}\, I_2''\, \omega_2''^2 - S_2''\ (p_0 - p_2'')\ v_2'' + (R'+\Delta R')\ v_2' + p_r^{II},$$ or, since $\Delta Q = \Delta R'$

$$L_a^{II} = VQ + \tfrac{1}{2}\, M_2\, v_2''^2 + \tfrac{1}{2}\, I_2''\, \omega_2''^2 + R'\ v_2' + \Delta R'\ (V+v_2') - S_2''\ (p_0 - p_2'')\ v_2'' + p_r^{II} \qquad (39)$$

and

$$\eta_2=\frac{L_u}{L_a^{II}}=\frac{L_a^{II}-[{}^1/_2\ M_2\ v_2''^2+{}^1/_2\ I_2\ \omega_2''^2+R'\ v_2'+\Delta R'\ (V+v_2')-S_2''\ (p_0-p_2'')\ v_2''+p_r^{II}]}{L_e^{II}} \tag{40}$$

For the ratio of L_a^I to L_a^{II} we find the value

$$\frac{L_a^I}{L_a^{II}}=\frac{Q\ (V+v_1)+C_1\ \omega_1+p_r^I}{(Q+\Delta Q)\ (V+v_2)+C_2\ \omega_2+p_r^{II}} \tag{41}$$

In general $(C_1\ \omega_1+p_r^I)$ and $(C_2\ \omega_2+p_r^{II})$ are of the same order of magnitude; $(Q+\Delta Q)>Q$, but generally $v_2<v_1$. We thus can not decide *a priori* between $L_a^I \gtreqless L_a^{II}$.[3]

I would warn those who think that the losses $[R'\ v_2'+\Delta R'(V+v_2')]$ can be estimated easily, As a matter of fact: First, the velocity in the slipstream, when some bodies are introduced into it, is totally changed in comparison with a free slipstream; second, the slipstream is not a uniform current, but a current of variable velocity along its axis; third, as the slipstream is a stream with free boundaries, the formulae and coefficients of fluid resistance deduced from experiments in fluids of infinite boundaries can not be applied to it, especially when the bodies considered do not have small cross sections in comparison with the cross section of the slipstream.

The efficiency η_1 will be called *free efficiency*, and designated in the following by η_f. The efficiency η_2 will be called *propulsive efficiency* and designated by η_p. We shall designate by f, and call it *neighborhood factor*, the ratio of the propulsive efficiency to the free efficiency.

We thus write:

$$\eta_p=f\eta_f \tag{42}$$

It is understood that the neighborhood factor f can be

$$f\gtreqless 1$$

As has been mentioned already, the free efficiency η_f is a function of the advance $\mu=V/N$ only. But as the slipstream created by a given propeller is also a function of μ only, the neighborhood factor, for a given propeller and given neighborhood conditions, can be a function of μ only. *Thus the propulsive efficiency must be a function of the advance μ only.*

In airplane testing, it is the propulsive efficiency η_p that has to be measured in order to evaluate the propeller in the actual working conditions.

One could raise the following two questions:

a. In what relation does the thrust $(Q+\Delta Q)$ of Case II stand to the momentum $M_2\ v_2'$ in section $S_2'\ S_2'$ (see fig. 3)? It is easy to see that

$$Q+\Delta Q=M_2\ v_2'+\Sigma'\ \overline{p}d\sigma$$

where $\Sigma'\ \overline{p}d\sigma$ is the resultant of the outside pressure on the whole boundary of the slipstream counted up to the section $S_2'\ S_2'$. Between the momentum $M_2\ v_2'$ and $M_2\ v_2''$ we have the relation:

$$M_2\ v_2''+(R'+\Delta R')-\Sigma''\ \overline{p}d\sigma=M_2\ v_2'+\Sigma\ \overline{p}d\sigma$$

b. What would the momentum in section $S_2''\ S_2''$ be if the whole resistance $R+R^1$ had been put in the slipstream? It is easy to see from relation (30) that we simply have:

$$M_2\ v_2''=\Sigma''\ \overline{p}d\sigma$$

because the resistance left outside the slipstream is in this case equal to zero.

I shall not go into a more detailed study of this important question of the propulsive efficiency. This would carry us too far into the propeller theory. Those who would like to

[3] For experimental data referring to the slipstream effect see "The design of screw propellers," London, 1920, pp. 192–196, by Henry C. Watts.

have a deeper understanding of the foregoing discussion are referred to the author's "General Theory of Blade Screws," previously mentioned.

B. PROPERTIES OF THE ENGINE.

Many discussions have been brought about by the question of how the brake horsepower L_m of a given gasoline engine, as actually used on airplanes, varies with the altitudes. Such discussions are rather a misunderstanding, because the power L_m does not depend on the altitude, but depends only upon:

1. The number of revolutions N at which the engine is running.
2. The throttle opening x.
3. The density δ and temperature T of the air in which the engine is working.
4. The quality of the gasoline used.

The question as to how density and temperature are connected with altitude depends exclusively upon meteorological conditions, which as known, are variable through the day, as well as through the year. In the following chapter the question of the standard atmosphere will be discussed briefly.

Since for a given mass of air, its pressure p, density δ and absolute temperature T are connected by the Claperyon relation $p/\delta = gRT$ where R is the gas constant, the brake horsepower can be as well considered as a function of the pressure p and temperature T. But since the propeller thrust and the forces of air resistance depend on the density δ, it is more convenient to relate the power L_m directly to the density δ.

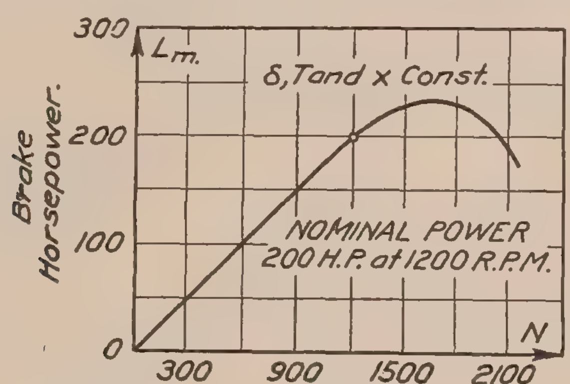


Fig. 4.

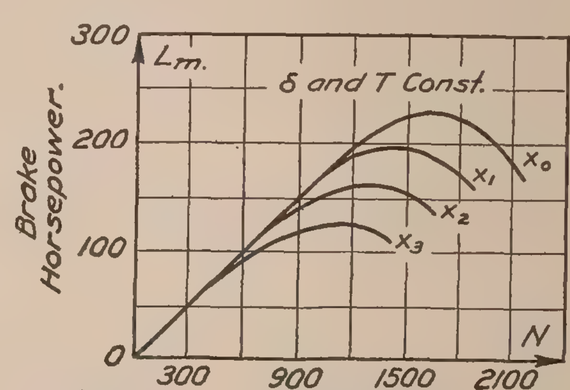


Fig. 5.

In figure 4 is represented the general course of the brake horsepower curve of a gas engine as a function of the revolutions N for δ, T and x constant. The power L_m generally first increases very closely proportionally to the revolutions, but afterwards, when the piston speed becomes too high, the power begins to drop, mainly on account of an incomplete filling of the cylinders by the carburetted air, whose flow speed is limited by the size of the suction pipes. This phenomenon is expressed generally by saying that the volumetric efficiency begins to drop, starting from a certain number of revolutions. The family of brake horsepower curves for δ and T constants, but for variable throttle openings x—a typical set of which is represented in figure 5—show well that the drop of power starting from a certain value of the engine revolutions is due to the drop of volumetric efficiency, because the smaller the throttle opening the earlier the power drop starts. In the figure, $x_0 > x_1 > x_2 > x_3$ - - - -.

In figure 6 is represented the general course of the power curve of a gas engine as a function of the density δ for N, T and x constants, and which for most gasoline engines is very close to a straight line. The density δ' (see fig. 6) is the small density at which the motor delivers just enough indicated power to compensate the mechanical losses, so that the brake horsepower is zero. For densities less than δ' power has to be applied to the engine in order to keep it rotating at a constant number of revolutions; p_0 indicates the mechanical losses of the engine when $\delta = 0$.

The general shape of the last power curve finds its explanation in the fact that the indicated horsepower is very closely proportional to the density δ; and, if the mechanical losses are considered as depending only slightly upon the density, *the linear dependence of the brake horsepower upon the density*, for N, T and x constants, follows.

In figure 7 is represented a family of brake horsepower curves as a function of N, for different values of the density δ, but for T and x constants, with $\delta_0 > \delta_1 > \delta_2 > \delta_3$ - - - - (Compare with fig. 5.)

It follows from the foregoing that as a first approximation the power of a gasoline engine can be represented by the formula

$$L_m = mN(c\delta - c_0) \tag{43}$$

for the range of variation of N and δ that we meet in aviation practice. This last formula assumes that the engine is used in the interval of the linear variation of the power with the revolutions, and that the mechanical losses are proportional to the revolutions. For actual aviation engines the coefficients c and c_0 have for mean values

$$c = 11; \; c_0 = 0.1$$

so that for average computations, we can write

$$L_m = mN(11\delta - 0.1) \tag{44}$$

where the coefficient m is fixed by the value of the nominal power of the engine.[4]

We do not possess actually sufficient experimental data on the question of variation of the power L_m with the temperature T, the density δ being kept constant. In some tests the

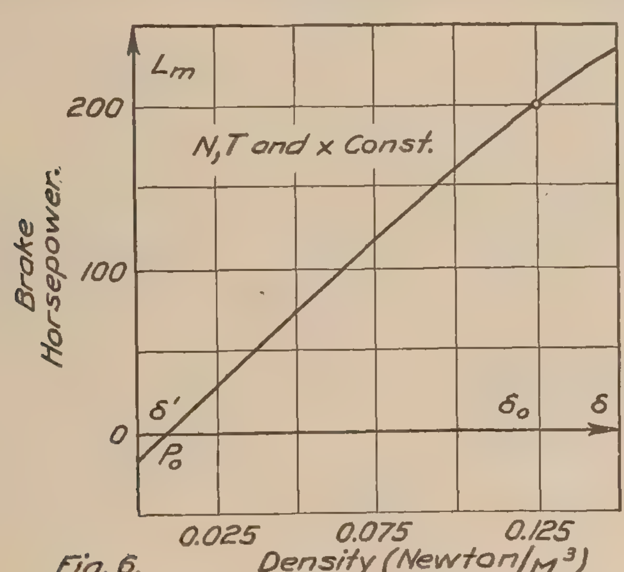


Fig. 6.

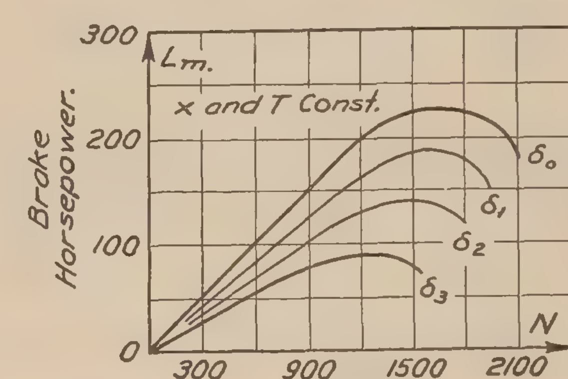


Fig. 7.

variation of the power L_m with T was observed for constant pressures. But in this case the main effect is the change of power produced by the change of density resulting from the temperature variation. It is the change of power with temperature at constant density that solely interests the aeronautical engineer for the study of altitude flight.

Sometimes engines are submitted to the following test: A propeller is fixed on the engine and this is run at different throttle openings x up to the full throttle x_0, and the curve of the power L_0 delivered by the engine is plotted as a function of the revolutions. The curve thus obtained has the general shape represented in figure 8. The main fact to be noted is that this curve is not the characteristic of the engine, but the characteristic of the propeller used, as tested at a fixed point. It is the curve that corresponds to the equation

$$L_m = L_0 = C_2'\delta N^3$$

which is a cubic parabola in N. It is possible by such tests to obtain the characteristic of the engine if it is tested either with a set of different propellers or with a variable-pitch propeller.

Suppose we run a first test with a propeller No. 1 and get the curve L_0^{I} on which we have carefully marked the different throttle openings x. Afterwards we run a test with a propeller No. 2 and get the curve L_0^{II} and so on. If we now join all the points of equal throttle openings

[4] The last formula assumes that at sea level the mechanical losses constitute around 7.5 per cent of the brake horsepower. For an engine giving at sea level 200 horsepower, at 25 revolutions per second, m turns out to be equal to 6.2.

we will get the engine characteristics L_m at constant density and different throttle openings x (See fig. 8.)

The foregoing explanations have been given only in order to recall briefly to mind those engine properties the knowledge of which is necessary for the study of our airplane steady-motion problem.

C. PROPERTIES OF THE ENGINE-PROPELLER SYSTEM.

As we have seen, the characteristics of a propeller are functions of the "advance" V/N. The characteristics of an engine are functions of the revolutions N for δ and x constant. It is easy to show that the characteristics of an engine-propeller system are functions of the flying speed V alone (for δ and x constant).

Let us consider an aviation engine with its propeller put on a railroad car and made to move along the axis of the engine-propeller system in air of density δ. Let us start by considering the car at *rest*. For a given *throttle opening*, if the engine is now set in motion it will reach a steady working condition at a certain number of revolutions N_o, at which the propeller will give a thrust Q_o. So far as we do not touch the throttle, the revolutions N_o and the thrust Q_o will remain constant. Let us now allow the car to run at a speed V. For each different value of the speed the engine will run at a different number of revolutions N and the propeller will give a different thrust Q, but if the car speed and throttle opening are kept unchanged, N and Q will remain constant. We thus see that for a given throttle opening x and air density δ, the revolutions N and the thrust Q of an engine propeller set are functions of the translation speed V alone. The main fact to be noted is that for an engine propeller set the revolutions have to adjust themselves to the speed V, which makes the thrust Q, for δ. and x constant, depend upon the speed V alone.

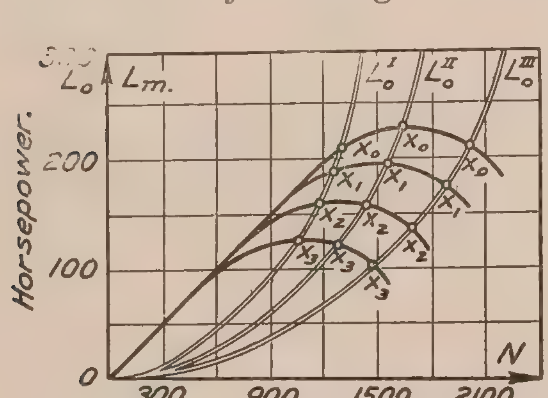


Fig. 8.

Let us now solve the following problem. A propeller is given to us by its characteristic curves:

$$\eta = F(V/N); \; L_a = \delta N^3 F_2''$$

and an engine, by its characteristic curve:

$$L_m = f(N)$$

for δ and x constant. The characteristics of the engine-propeller set have to be found.

Let us first deduce from the $L_m = f(N)$ curve, the curve of $L_m/\delta N^3$ as a function of N for the given δ and x. This last curve plotted, the following table is computed:

In column I we write selected values of $L_m/\delta N^3$.

In column II we write the corresponding values of N taken from the $L_m/\delta N^3$ curve plotted as a function of N.

I.	II.	III.	IV.	V.	VI.	VII.	VIII.
$L/_m\delta N^3$	N	V/N	η	V	$L_u/\delta N^3$	Q/δ	Q
—							
—							
—							

When a propeller is fixed to a given engine, the power absorbed by the propeller is equal to the power delivered by the engine and both run at the same number of revolutions (in case of gearing, the gearing constant has only to be introduced) that is, we have

$$\frac{L_m}{\delta N^3}=\frac{L_a}{\delta N^3}$$

Let us thus read from the $L_a/\delta N^3$ curve as a function of V/N the values of the advance V/N that correspond to the selected $L_m/\delta N^3$ values in Column I and write these values of the advance V/N in Column III.

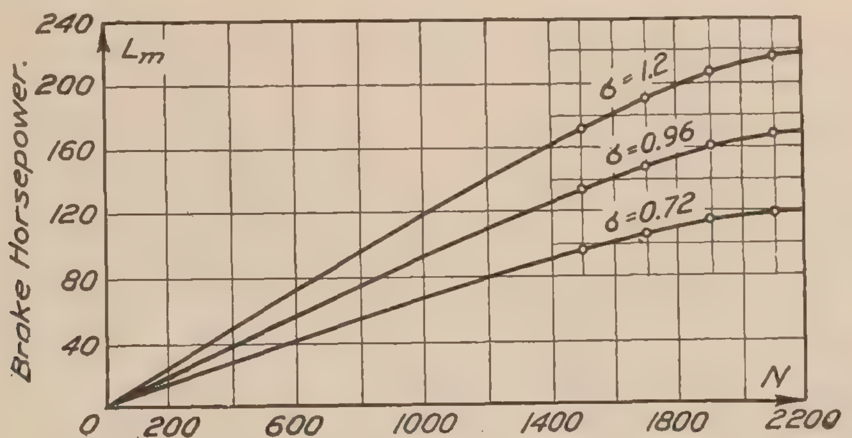


Fig. 9.

In column IV we shall write the values of the efficiencies η that correspond to these same advances V/N.

Multiplying Column II by Column III, we find the value of V (written in Column V) that correspond to the $L_m/\delta N^3$ values of Column I.

Multiplying Column I by Column IV, we find the value of $L_u/\delta N^3$ (L_u=useful power) that correspond to the values of V of Column V.

Finally, multiplying Column VI by the corresponding values of N of Column II and dividing by the corresponding values of V in Column V, we find in Column VII the values of Q/δ as a function of the V of Column V. Finally, the values of the thrust Q are given in Column VIII.

Thus we are able to plot the curve of Q/δ or Q as a function of V for given δ and x.

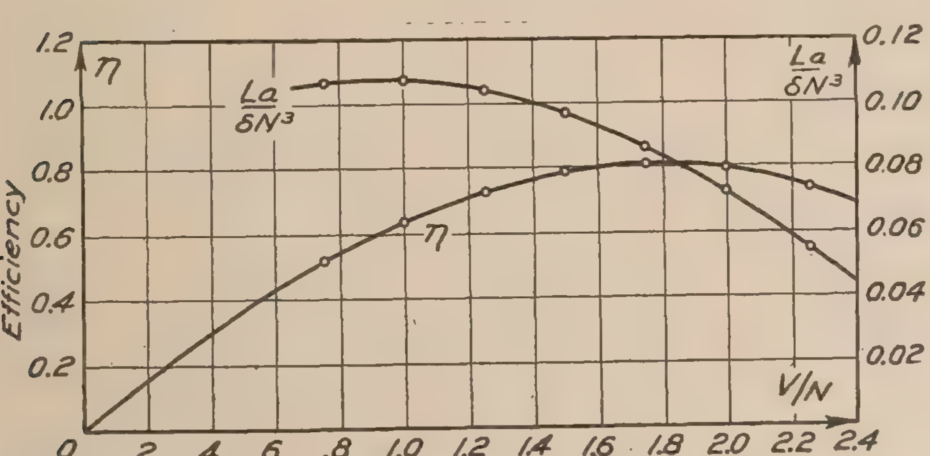


Fig. 10.

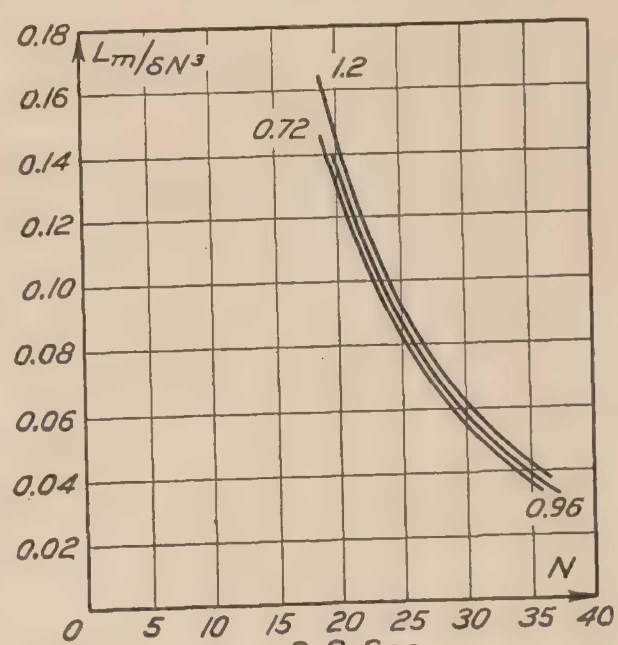


Fig. 11.

For each new values of δ and x, this computation has to be repeated. Columns I, III, IV, and VI do not change, and only Columns II, V, VII, and VIII have to be recomputed.

In such a manner, starting from the knowledge of the propeller characteristics as functions of the advance V/N and the engine characteristics as functions of N, δ, and x, the characteristics of the engine-propeller system, such as

$$Q,\ L_u,\ L_m,\ \eta, \text{ and } N$$

will be found as functions of V, δ, and x.

Following is an example of the application of this method. In figures 9 and 10 are represented the characteristics of the propeller and engine used. In figure 11 are represented the curves of $L_m/\delta N^3$ as a function of N.

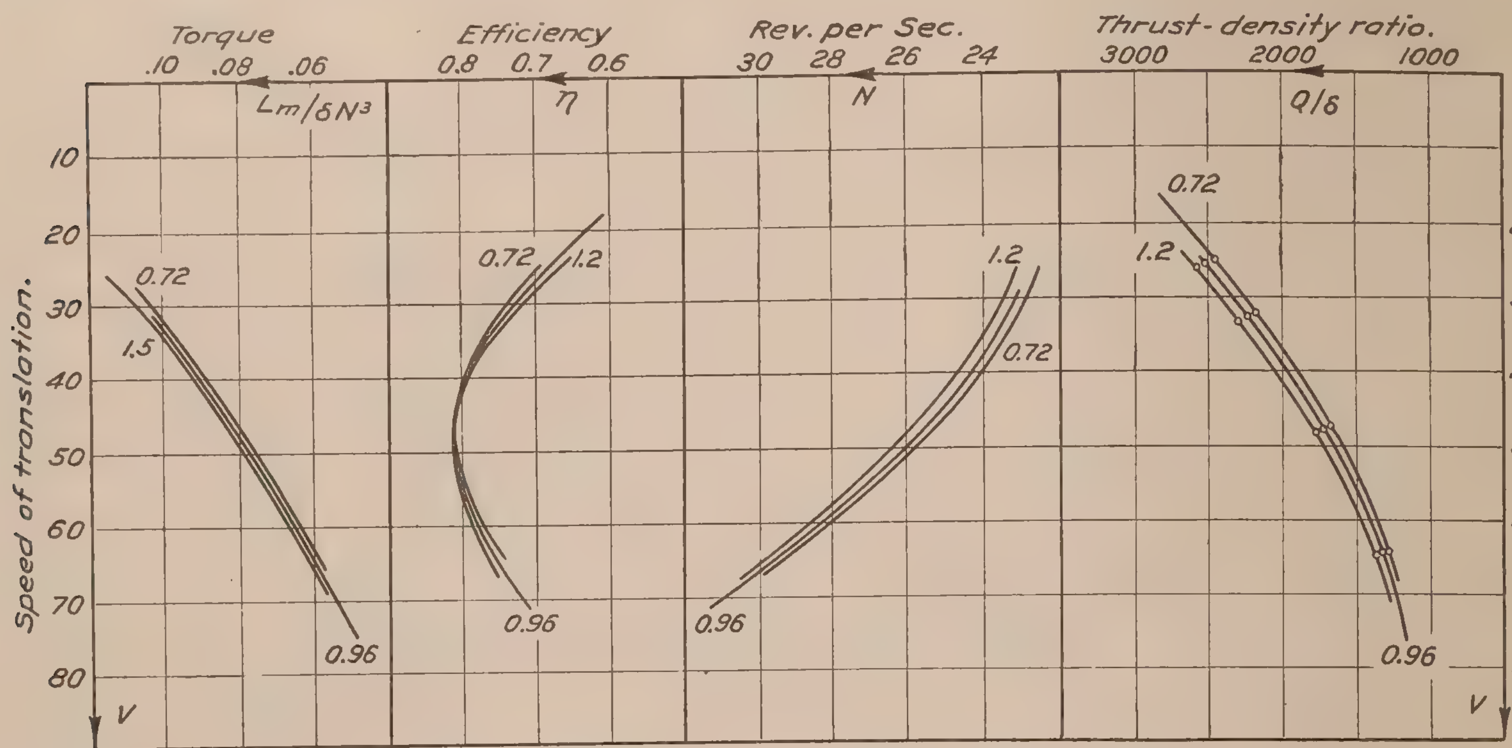


Fig. 12. — CHARACTERISTICS OF THE ENGINE-PROPELLER SET. —

In the following table are given the computations which are made for full throttle openings and for three different air densities. The characteristics obtained for the engine-propeller set are represented in figure 12, on which are plotted the curves of Q/δ, N, η, and $L_m/\delta N^3$ as functions of V for different densities. It can be easily seen that the sets of these curves can as a first approximation be reduced to one mean curve for each set, which is a consequence of the fact that the deviation of the motor power from proportionality to air density is not great.

We now see upon what factors the variation of the propeller thrust in flight depends and what laws it follows.

Computation of the characteristics of an engine-propeller system.

δ	$L_m/\delta N^3$	N	V/N	η	V	$L_m/\delta N^3$	Q	Q/δ
1.20	1.05	23.5	1.16	0.70	27.2	0.0735	325	2650
	1.00	24.0	1.40	.76	33.6	.0760	288	2350
	.80	26.5	1.87	.81	49.5	.0647	224	1830
	.60	30.0	2.20	.76	66.0	.0396	172	1400
.96	1.05	23.0	1.16	.70	26.7	.0735	247	2530
	1.00	23.5	1.40	.76	32.9	.0760	220	2250
	.80	26.0	1.87	.81	48.6	.0647	171	1750
	.60	29.6	2.20	.76	65.1	.0396	133	1360
.72	1.05	22.5	1.16	.70	26.1	.0735	177	2410
	1.00	23.0	1.40	.76	32.2	.0760	158	2150
	.80	25.5	1.87	.81	47.7	.0647	122	1660
	.60	29.2	2.20	.76	64.3	.0396	96	1315

REPORT No. 97.

GENERAL THEORY OF THE STEADY MOTION OF AN AIRPLANE.

By George de Bothezat.

PART III.

THE ATMOSPHERE.

1. SOME GENERAL PROPERTIES OF THE ATMOSPHERE.

A short review of those properties of the atmosphere that have a direct relation to the airplane steady motion will be given here.

The specific weight of air—expressed in kilograms—will be represented by σ.

The density of air—expressed in newtons will be represented by δ.

We have

$$\sigma = g\delta \text{ with } g = 9,\ 81 mt/sec^2$$

At the pressure of 1 atmosphere$\cong$10330 klg/mt^2 and absolute temperature $T = 273° + 15° = 288°$, the specific weight and density of air have for mean values

$$\sigma = 1{,}225 \text{ klg};\ \delta = 0{,}128 \text{ newton}.$$

At the same pressure of 1 atmosphere and zero degrees centigrade ($T = 273$) we find

$$\sigma_0 = 1{,}293 \text{ klg};\ \delta_0 = 0{,}132 \text{ newton}.$$

Using the former values, the gas constant R, deduced from Clapeyron's relation

$$p = \sigma R T \tag{45}$$

has for its value

$$R = \frac{p}{\sigma T} = \frac{10330}{1{,}225 \times 288} = 29,\ 27 \tag{46}$$

Let us consider the atmosphere to be in a perfect static condition (no winds). If we rise in such an atmosphere through a distance dH, the pressure p will vary by an amount $-dp$ equal to

$$-dp = \sigma dH \tag{47}$$

or on account of (45)

$$\frac{-dp}{p} = \frac{dH}{RT} \tag{48}$$

(If in this last formula we consider $dp/p = 0.01$; $R = 29,\ 27$; $T = 273$ we find $dH \cong 80\ mt$. This means that a difference of pressure of 1 per cent in the atmosphere corresponds at 0° centigrade to a change of altitude of 80 mt.)

Integrating (48) we get

$$lg \frac{p}{p_0} = \frac{1}{R} \int_H^{H_0} \frac{dH}{T} \tag{49}$$

where p_0 is the pressure at the altitude H_0 and $H > H_0$. This last formula gives the value of the altitude H from the knowledge of pressure, when the law of variation of the temperature T with altitude is known. For $T = 273$; $p/p_0 \cong 0.5$ and $H_0 = 0$, we find $H \cong 5{,}000\ mt$. In an isothermic atmosphere of zero degrees centigrade, at the altitude of around 5000 mt the pressure is one-half of the ground level pressure.

2. DISCUSSION OF THE STANDARD ATMOSPHERE.

This question of the law of variation of temperature with altitude has been lately a matter of considerable discussion. Numerous so-called *standard atmospheres* have been proposed, which are supposed to be some kind of average deduced from different sets of meteorological observations. The Paris "Peace Conference of 1919" has even considered it necessary to fix by interallied agreement, some kind of standard atmosphere.

A careful examination of all the propositions made has brought me to the conclusion that this question of the standard atmosphere has been somewhat misunderstood.

Let us consider the whole question from a general standpoint and make clear for what purpose we need the standard atmosphere in aviation.

For each geographical position, at a given hour of a given day, there exists along the vertical drawn through the place considered, a certain distribution of pressures and temperatures. This distribution of pressures and temperatures depends upon the meteorological conditions and is variable through the whole year. The variations of this pressure and temperature are very important. It is well known that the same pressures and temperatures can be met at levels where altitude differences can amount to several thousands of meters. If a certain mean distribution of pressures and temperatures is adopted, the deviation from this mean distribution can also make up actual altitude differences of the order of a thousand meters.

On the other hand for an airplane, the forces of air-resistance, the propeller thrust, and the power of its engine are all functions of the air density and decrease with this. It is a property of the airplane to be able to reach a certain limiting small value of the air density, at which the airplane can still fly level, but is unable to climb any more. This limit of density is called the *ceiling density* δ_c. The aviation engineering problem consists in finding for each airplane its ceiling density. But the question, at what altitude this density δ_c is located is purely a meteorological question. The distribution of densities in the air is greatly different and the same density can be met on different days at very different altitudes. The question of the relation of densities and altitudes stand outside the aviation engineering problem and is merely a question of public curiosity. Technically speaking, we can only say that a given airplane has the ability to reach a certain density δ_c. The smaller this density δ_c, the greater is the climbing capacity of the airplane considered. There is no reason for expressing this density in altitude figures, because density already completely specifies the question. There is only one fact that must still be taken into account. The power of the airplane engine, at a given density, depends somewhat upon the temperature. When we speak of airplane performances, they must thus be referred to a certain temperature. In the selection of this temperature, we must be guided only by convenience and simplicity. There are no reasons to adopt a temperature variable with the altitude, but there are many reasons for adopting a constant temperature at all altitudes.

It is easy to see that, exactly speaking, it is rather standard conditions for engine work that we have to select than to adopt a standard atmosphere. If we make the temperature variable with altitude, in other words, with density, this would mean that the standard conditions adopted for engine work consist of a special temperature for each density. This introduces a very troublesome element in engine-power computations, which is neither necessary nor demanded by any reason. On the contrary, a constant temperature for all densities is a natural condition, demanded for the sake of simplicity of the standard conditions adopted.

We are thus brought to the conclusion that from the standpoint of aviation engineering the only standard atmosphere that can be reasonably adopted is the *isothermic atmosphere.*

The proposition of the author is to adopt for aviation engineering, as a standard atmosphere, an atmosphere of constant temperature in its whole mass equal to zero degrees centigrade.

The advantages of such a convention are as follows:

1. In all questions of design and performance prediction all temperature corrections are totally eliminated.

2. In airplane testing the only correction to be made is the temperature correction of the engine power and reduction of the performance to this corrected power. This correction is

quite simple, since we reduce each of the quantities to the same temperature independent of the altitude.

3. If we compare our isothermic atmosphere of zero degrees centigrade with the inter-allied standard atmosphere and with the atmosphere of an average winter and summer day, the isothermic atmosphere gives a general idea of the altitude quite as good as does the inter-allied standard in the sense that the altitude departures for the winter and summer day from the isothermic atmosphere and from the inter-allied standard are of the same order of magnitude. (See fig. 13.)

It is thus evident that everything speaks in favor of the isothermic atmosphere of zero degrees centigrade.

In all aviation engineering all data, computations, and performances should be expressed exclusively in densities of the isothermic atmosphere. The altitude language has to be used for the general public only. The altitude language is of no use to the aviation engineer and only creates unnecessary complications.[1]

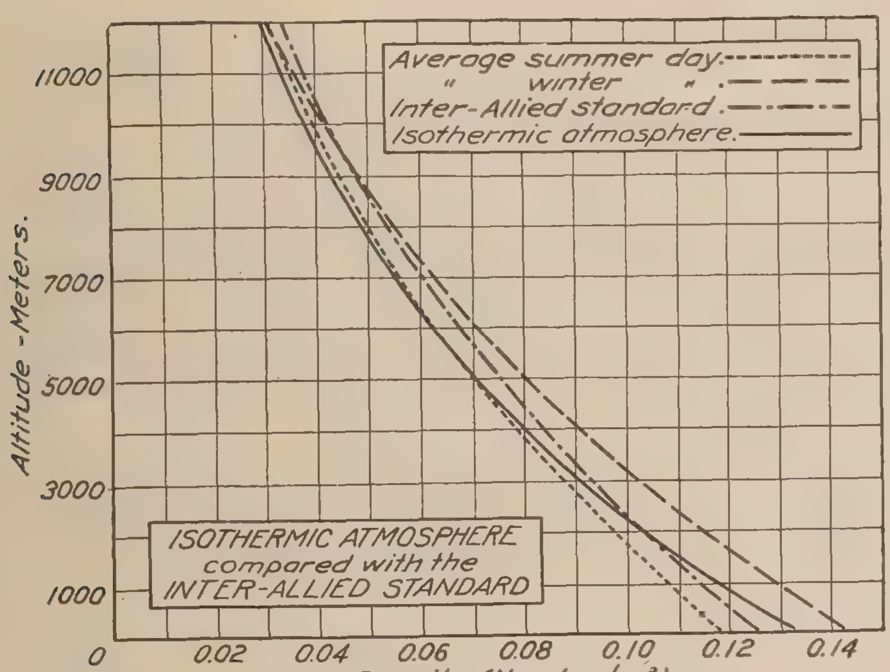


Fig. 13.

For some special problems we need to know the actual altitude of an airplane. But in such cases no standard atmosphere can be of any help to us. If we want to deduce with some accuracy the actual altitude from pressure measurements on an airplane, we have to record when climbing the laws of the actual variation of pressure and temperature with altitude.

Methods or instruments for quick computation, with a certain accuracy, of the actual altitude from such records can be developed.

As a general conclusion I will say: *It is fitted to the purpose to adopt for aviation engineering the isothermic atmosphere of zero degrees centigrade as standard atmosphere.*

3. CALCULATION OF THE RATE OF CLIMB FROM A BAROGRAM.

The vertical component U of the air speed V of a climbing airplane is generally called *rate of climb*. If in an element of time dt an airplane climbs a height dH, its rate of climb is equal to

$$U=\frac{dH}{dt} \tag{50}$$

or on account of formula (48)

$$U=\frac{RT}{p}\cdot\frac{-dp}{dt}=\sigma\frac{-dp}{dt} \tag{51}$$

By this last formula the rate of climb of an airplane can be found from the flight barogram, which gives the pressure p as a function of the time t. At each moment of time the slope of the tangent to the barogram curve at the point considered gives the value of $-dp/dt$ and the value of the specific weight follows from the corresponding values of pressure and temperature.

[1] The author is of the opinion that the scales of altimeters and barographs ought to be graduated in pressure units, pressure being the quantity that these instruments really measure. The author can not understand why it is considered "from a practical standpoint" preferable to have the pilot read the *wrong altitude* (the altitude scale of an altimeter being purely conventional) than to read the *exact pressure*. Very little practice would be required from pilots to accustom themselves to express the level reached in the atmosphere in pressure figures. This would be, physically, perfectly correct and would avoid a great deal of misunderstanding.

4. INFLUENCE OF WINDS AND SELF-SPEED ON COCKPIT PRESSURE.

Let us consider briefly what influence the atmospheric winds can have on the observed pressures. Consider two air masses at nearly the same altitude, in which the Bernouilli constant has the same value, one mass having no speed, the other having a speed of 10 meters per second which is itself a strong wind. We will find under such conditions that the pressures in these two air masses are related by the equation

$$p_1 = p_2 + \frac{\delta v^2}{2}$$

or

$$p_1 - p_2 = \frac{\delta v^2}{2} = \frac{0{,}125 \times \overline{20}^2}{2} = 6{,}25 klg/mt^2$$

At ground level with $p_1 = 10330\ klg/mt^2$ this gives

$$p_2/p_1 = 1 - \frac{6,\ 25}{10330} = 0,\ 9994$$

The difference between p_2 and p_1 thus appears to be of the order of 0.05% of the atmospheric pressure, which corresponds at ground level to a difference of altitude of around 4 meters. At an altitude where the pressure would be 10330/2 (around 5,000 meters) the difference between p_1 and p_2 would be double and this would still correspond only to a difference of altitude of 8 meters. We are thus brought to the conclusion that ordinary winds will affect only slightly the calculation of altitude from pressure distribution.

A much more marked influence is that of the variation of the airplane speed V upon the measurements of pressures as made on an airplane. The difference between the static pressure p at the level where the airplane is flying and the pressure p' in the cockpit is very closely equal to

$$p - p' = \frac{\delta V^2}{2}$$

An airplane in climbing can have its speed reduced to about half of its horizontal self-speed; that would give for the cockpit pressures p'_1 and p'_2 corresponding to the two cases, a difference of

$$p_2' - p_1' = 0,\ 75 \frac{\delta V^2}{2}$$

This is the difference between the "corrections" which are necessary in the two cases in order to determine, from the cockpit readings, the real static pressures. With $V \cong 50$ meters per second, at ground level, this can be about 1% of the atmospheric pressure, or 80 meters difference in altitude. Such differences have to be taken into account when pressure observations are made in the cockpit, at different values of the speed. The last circumstance may make it desirable to use special devices allowing the direct observation in the airplane of the static pressure, instead of the cockpit pressure.

REPORT No. 97.

GENERAL THEORY OF THE STEADY MOTION OF AN AIRPLANE.

By George de Bothezat.

PART IV.

THE THEORY OF STEADY MOTION.

1. THE BASIC EQUATIONS.

All of the properties of the airplane steady motion are a direct consequence of the fact that in *steady motion* all the forces acting on an airplane mutually balance.

In order to express the last condition let us consider an airplane in steady flight and draw from its center of mass vectors parallel to the main quantities involved in the problem. This will facilitate a determination of the angular relations. In this manner, on figure 13, have been represented:

- G_f a reference line, invariably connected with the airplane wings, from which the angle of attack is measured.
- V the self-speed.
- i the angle of attack.
- γ the angle of inclination of the flying path to the horizontal, counted positive for climbing.
- P the total weight of the airplane.
- R the total air-resistance.
- R_x the drag – component of R along the speed V.
- R_y the lift – component of R along the normal to the self-speed.
- Q the propeller thrust.

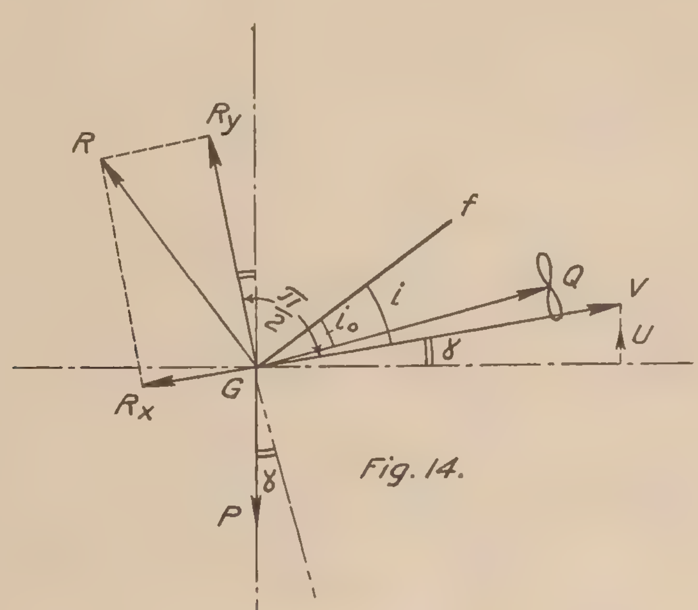


Fig. 14.

Airplanes are generally built in such a way that for horizontal flying in normal conditions, the thrust Q is directed along the speed V. The angle of attack that corresponds to those conditions will be designated by i_0.

The vertical component of V, designated by U in figure 14, is called the *rate of climb*.

$$U = V \operatorname{Sin} \gamma \tag{52}$$

Let us now project all the forces acting on the airplane on the direction of the speed V and the normal to it. The conditions of steady motion will then be expressed by

$$R_x = Q \text{ Cos } (i - i_0) - P \text{ Sin } \gamma \tag{53}$$

$$R_y = P \text{ Cos } \gamma - Q \text{ Sin } (i - i_0) \tag{54}$$

in which equations, we have:

$$R_x = k_x \, \delta \, A V^2 \tag{55}$$

$$R_y = k_y \, \delta \, A V^2 \tag{56}$$

where

k_x = drag coefficient.
k_y = lift coefficient.
δ = air density at flying level.
A = area of the airplane wings.

To these two equations (53) and (54) must be added the fundamental relation that connects the engine with the propeller.

$$L_m \, (x, \delta, N) = L_a = \delta N^3 \, T_2''\left(\frac{V}{N}\right) \tag{57}$$

which expresses the fact that in steady flight the power L_m delivered by the engine—a function of x, δ and N—is always equal to the power L_a absorbed by the propeller—a function of N, δ and V/N.

The detailed discussion of the fundamental equations (53) and (54) is greatly complicated by the complex laws, fixed by the relation (57) governing the variation of the thrust Q in flight, which we have considered in full detail in the foregoing.

In order to allow a better survey of fundamental properties of the airplane in steady motion, without complicating the question by those factors that have only a slight influence on the quantitative value of the results and do not affect at all their general meaning, we shall make the following simplifications in equations (53) and (54).

I shall first remark that, on the one hand, we do not possess any reliable information as to the laws of variation of the propeller thrust for the case when the self-speed V makes a certain angle with the propeller axis, and on the other hand, since the angle $(i - i_0)$, as we shall see later, can take only small values in normal flying conditions, we shall consider it to be a sufficient approximation to assume

$$Q \; Cos \; (i - i_0) \cong Q$$
$$Q \; Sin \; (i - i_0) \cong O$$

It must be further noted, that in normal flying conditions, the angle of the flying path to the horizontal does not usually take large values. It seldom exceeds 15°, taking larger values only in steep dives and steep glides, which must be considered separately. We thus assume

$$Sin \; \gamma \cong \gamma; \quad Cos \; \gamma \cong 1.$$

Introducing these simplifications in the equations (53) and (54) we get:

$$R_x = k_x \, \delta \, A V^2 = Q - P\gamma \tag{58}$$

$$R_y = k_y \, \delta \, A V^2 = P \tag{59}$$

The simplifications we have made affect principally the value of the self-speed V, which we shall calculate from the equation $V = \sqrt{P/k_y \, \delta \, A}$ instead of the more exact relation $V = \sqrt{Cos \; \gamma} \sqrt{P/k_y \; \delta \; A}$. But it is easy to see that for $\gamma < 5°$ we have $Cos \; \gamma > 0,96$ and the error made in the speed will be less than $1 - \sqrt{0,96} \cong 0,02$, that is less than 2 per cent. In any case,

when necessary, once the self-speed V obtained by equation (59) and the value of γ known, it is always possible to correct its value by taking it equal to $V\sqrt{Cos\ \gamma}$.

For the study of the airplane in steady motion, it is more convenient to consider, instead of the relations (58) and (59), the system of equations:

$$k_y\ V^2=\frac{P}{\delta\ A} \tag{60}$$

$$\gamma=\frac{Q}{P}-\frac{k_x}{k_y} \tag{61}$$

which follow directly from the last.

With the same approximation, the rate of climb U has for its value

$$U=V\ \gamma \tag{62}$$

It is from the system of equations (60) and (61) that we shall deduce all the properties of an airplane in steady motion. Once all of the mutual interrelations of all the quantities involved in the problem are perfectly established, all quantitative corrections, when demanded, can always be made post factum and in our statement of the problem, are only necessary in the case of steep climbing or gliding flights.

We shall start by the study of the steady motion of an airplane of constant total load weight P and constant wing area A, the engine working all the time at full throttle opening.

For this study we shall make use of a graphical interpretation of the equations (60) and (61) which consists of the following:

2. THE METHOD.

The lift coefficient k_y will be adopted as fundamental variable. As the lift coefficient k_y is a function of the angle of attack only, under the restrictions made, its value can be changed in flight only by a displacement of the control stick.

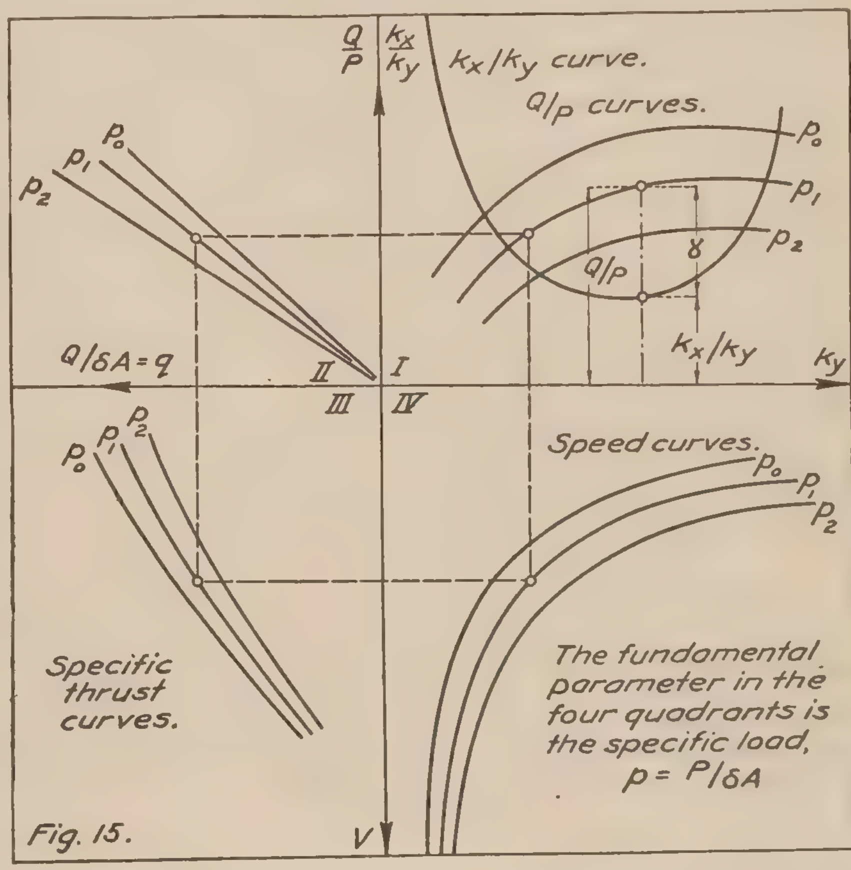


Fig. 15.

The four quadrants formed by two straight lines intersecting at right angles will be designated respectively by I, II, III, and IV. (See fig. 15.)

In quadrant IV, the horizontal axis will be adopted as k_y axis, and the downward vertical axis as V axis; and the speed V will be plotted in this quadrant as a function of the lift coefficient k_y, according to equation (60). We obtain a system of hyperbola-like curves having $P/\delta\ A$ as parameter, which allow one to read directly the value of the speed V that corresponds to each value of the lift coefficient when the value of $P/\delta\ A$ is known.

It is important to remark that when the ratio $P/\delta A$ has the same value, independently of the special values that P, δ and A have, to each k_y corresponds the same speed V. On account of this we shall use the special symbol p to represent the ratio $P/\delta A$ and shall call it *specific load*.

$$p=\frac{P}{\delta A} \tag{62}$$

It will be seen from all that follows that the specific load p is the fundamental parameter of the whole problem of steady motion.

The curves of V as a function of k_y with p as parameter will be called *speed curves.* It is easy to see that the speed curves—in the approximation used in the problem—simply represent a mathematical relation independent of the special airplane considered; that is, the same family of speed curves corresponds to any airplane. The last property of the speed curves is a consequence of our selection of variables. A set of speed curves for p having the values p_o (ground level) p_1, p_2---- is schematically represented in quadrant IV of figure 15.

We have seen, when examining in the foregoing chapters the properties of the engine-propeller system, that the thrust Q given by such set is, for a constant throttle opening, a function of the speed V and density δ. In the study of the airplane in steady motion, for the sake of uniformity, we shall consider, instead of the thrust Q, the quantity $Q/\delta A$, which we shall call *specific thrust* and designate by q, i. e.,

$$q=Q/\delta A \tag{63}$$

It is easy to see that the specific thrust q will also be, for a constant throttle opening, a function of the speed V and density δ, or, in other words, a function of V and of the specific load $p=P/\delta A$ as P and A are constants of the airplane considered. We shall plot in quadrant III the curves of the specific thrust $p=P/\delta A$ as a function of the self-speed V with the specific load $p=P/\delta A$ as parameter, the horizontal axis being the q axis. A set of such specific thrust curves is represented in figure 15, with the parameter p having the same values p_o, p_1, p_2---- as in quadrant IV. We have learned in previous chapters how to deduce this family of curves from the properties of the propeller and engine used on the airplane considered. The specific load curves allow us to deduce directly the value of the propeller thrust that corresponds to each flying speed.

Knowing the laws of variation of the thrust Q as a function of the speed V, it is easy to deduce the laws of variation of the thrust Q as a function of the lift coefficient k_y. For this purpose, let us consider the equation

$$\frac{Q}{P}=\frac{Q/\delta A}{P/\delta A}=\frac{q}{p}=y \tag{64}$$

and interpret it as a family of straight lines passing through the origin with q as abscissa, y as ordinate, and p as parameter, and plot these straight lines in quadrant II of figure 15, giving successively to p the values p_o, p_1, p_2-- and using the vertical axis as the $y=Q/P$ axis. We shall call these last lines *transfer lines.*

This system of transfer lines once plotted in quadrant II, it is easy to trace directly in quadrant I, for each given value of the specific load p, the curve of the ratio Q/P as a function of k_y that corresponds to a given curve of $q=Q/\delta A$ as a function of V. Each two corresponding points of two corresponding curves in quadrants III and I lie on the two diagonal vertices of a rectangle whose sides are parallel to the axes and whose two other vertices are located one on the speed curve in quadrant IV, the other on the transfer line in quadrant II, corresponding to the same value of the specific load p. (See fig. 15.) In such a way we can deduce in quadrant I, from the specific thrust curves of quadrant III, the curves of Q/P as a function of k_y with p as parameter, which give us the laws of variation of the thrust as a function of the lift coefficient. A set of such curves of Q/P as a function of k_y, for the same values p_o, p_1, p_2---- of the specific load, is represented in quadrant I of figure 15. We now see that the chart represented on this figure has the property that *each four corresponding points in the four quadrants I, II, III and IV lie on the vertices of a rectangle with its sides parallel to the axes.*

Let us finally plot in quadrant I, in addition to the Q/P curves as a function of k_y, the curve of the drag-lift ratio k_x/k_y as a function of k_y. As the drag and lift coefficients are functions of the angle of attack only, the ratio k_x/k_y can be considered as a function of k_y only. The general shape of the k_x/k_y curve as a function of k_y is represented in figure 15.

If we now remember equation (61), we see that the difference of the ordinates of the Q/P and k_x/k_y curves, for a given value of p, give directly the airplane path inclination γ (see figure 15).

We have thus succeeded in representing on the chart in figure 15 all the characteristics of the airplane in steady motion and their fundamental interrelations. The transfer lines of quadrant II and the speed curves of quadrant IV do not depend on the special type of airplane considered and thus are merely mathematical intermediaries. On the contrary, the curves of quadrant I and III constitute the characteristics of the airplane considered. Quadrant I with the k_x/k_y curve taken alone constitutes the characteristic of the airplane alone. Quadrant III, with the specific thrust curves, constitutes the characteristic of the engine-propeller system. Quadrant I, with the k_x/k_y and Q/P curves constitutes the characteristic of the airplane-engine-propeller system.

In Quadrant I, for a given value of k_y and p, we can read the values of k_x/k_y, Q/P and γ. In Quadrant IV we can read the corresponding values of the speed V; in Quadrant III we can read the corresponding value of the specific thrust $Q/\delta A$. We are at liberty to extend to the left Quadrant III, and plot, as had been done in figure 12, all the other characteristics of the engine-propeller set; and thus we shall obtain a complete graphical representation of the whole set of quantities involved in the steady motion of an airplane.

In order to make ourselves familiar with the above described chart, we shall discuss with its aid, in their general outline, the properties of the airplane in steady motion.

3. PROPERTIES OF STEADY MOTION.

All the curves of our basic chart represent quantities that can be measured directly, that is why all these curves can be considered as being deduced experimentally from direct tests. But for many purposes, it is convenient to have also analytical expressions, even if only to a first approximation, of all the curves of the four quadrants of the chart. That is why, deducing in the following the properties of an airplane in steady motion by the aid of the chart, we shall at the same time follow all the fundamental relations by the use of the following approximate equations.

We have already seen that the speed curves of Quadrant IV have for their equations

$$k_y V^2 = p \tag{65}$$

The shape usually obtained for the specific thrust curves of Quadrant III, allow us to use for their representation, with a sufficient approximation, an equation in the form of

$$\frac{Q}{\delta A} = q = q_0 - q_1 V^2 \tag{66}$$

the whole set of curves being represented by a single mean curve. The constant coefficients q_0 and q_1 are, as a first approximation, characteristic coefficients of the engine-propeller set. These coefficients q_0 and q_1 can be deduced from the mean specific thrust curve by the method of least squares. A justification of the last relation (66) will be found in the note at the end of this report.

By using equation (66) one sees that the ratio Q/P is equal to

$$\frac{Q}{P} = \frac{Q/\delta A}{P/\delta A} = \frac{q}{p} = \frac{q_0 - q_1 V^2}{p} \tag{67}$$

and on account of relation (65), we find for the Q/P curves of Quadrant I, the equation

$$\frac{Q}{P} = \frac{q_0}{p} - \frac{q_1}{k_y} \tag{68}$$

the specific load $p = P/\delta A$ being the parameter of this family of curves.

Rewriting equation (68) in the form

$$q_1 = k_y\left(\frac{q_0}{p} - \frac{Q}{P}\right) \tag{69}$$

it will be seen that the curves represent a family of equilateral hyperbolas having for asymptotes the Q/P axis and a line parallel to the axis, with its ordinate equal to

$$\frac{q_0}{p} = \frac{q_0 \delta}{P/A} \tag{70}$$

that is, proportional to the density δ. (See fig. 16.) When δ varies, the curves (69) merely move up or down, their ordinates changing proportionally to the density δ.

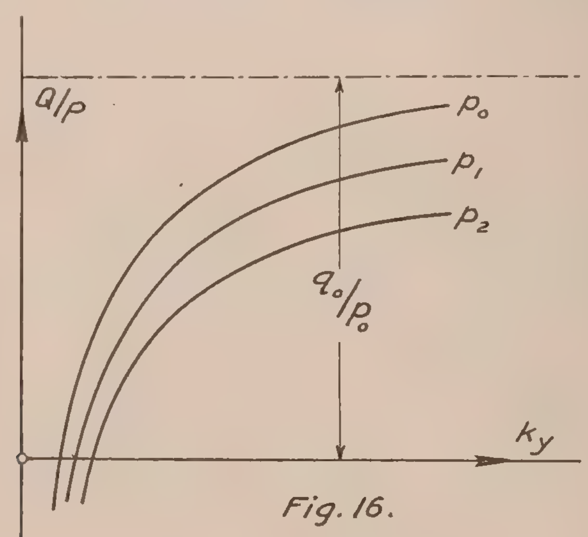


Fig. 16.

As, according to the approximate relations (8) and (9), the lift R_y is a linear function of the angle of attack; and the drag R_x is a quadratic function of the same angle, we can consider

$$R_x \cong k_y\ \delta\ AV^2\left(r\ k_y + t + \frac{\sigma}{k_y}\right); \tag{71}$$

that is, adopt for the $k_x/k_y = R_x/R_y$ curve, the approximate equation

$$\frac{k_x}{k_y} = r\ k_y + t + \frac{\sigma}{k_y} = y, \tag{72}$$

which represents a hyperbola, whose equation written in the form

$$r\ k_y^2 + (t-y)\ k_y + \sigma = f\ (y,\ k_y)$$

shows us that the center of this hyperbola has its coordinates given by

$$\frac{\partial f}{\partial k_y} = 2r\ k_y + t - y = 0$$

$$\frac{\partial f}{\partial y} = -k_y = 0$$

that is, is located at the point $k_y = 0;\ y = t$, and that the angular coefficients of the asymptotic directions are given by

$$r\ k_y^2 - y\ k_y = k_y\ (r\ k_y - y) = 0$$

that is, are equal to

$$k_y = 0 \text{ and } y/k_y = r$$

The hyperbola (72) is represented on figure 17, on which the coefficient t is assumed to be negative, as most generally is the case.

The minimum of k_x/k_y is given by

$$r\,k_y=\frac{\sigma}{k_y}$$

that is, takes place for

$$(k_y)_m=\sqrt{\frac{\sigma}{r}}$$

and is equal to

$$\left(\frac{k_x}{k_y}\right)_{min}=t+2\sqrt{r\sigma}$$

If we now remember that in the expression (71) it is the coefficient σ that depends mainly upon the parasite resistance of the airplane considered, we see that the center and the asymptotes of the hyperbola (72) are independent of the parasite resistance, and that with variable σ this hyperbola moves in and out between its invariable asymptotes.

A. HORIZONTAL FLYING.

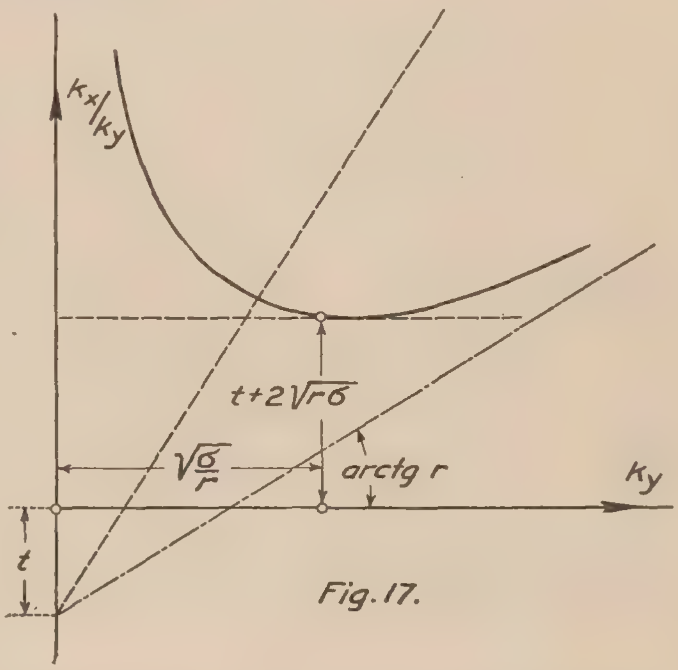


Fig. 17.

A horizontal flight of the airplane considered, at a level specified by a given value of the specific load $p=P/\delta A$ is only possible so far as the corresponding Q/P curve intersects in quadrant I with the k_x/k_y curve, because only for such points of intersection can we have $\gamma=0$. It is easy to see from figure 18 that there will be in general two such points of intersection, to which correspond two different values $(k_y)_1$ and $(k_y)_2$ of the lift coefficient. In quadrant IV we can read the two values V_1 and V_2 of the speed that correspond to these values of the lift. One of these speed values is greater than the other $(V_1>V_2)$. The greater value V_1 is called *high speed*, the smaller value V_2 *low speed*. In quadrant III we can read the corresponding values $Q_1/\delta A$ and $Q_2/\delta A$ of the specific thrust $(L_u)_1/\delta A$ and $(L_u)_2/\delta A$ of the power available; $(L_m)_1/\delta A$ and $(L_m)_2/\delta A$ of the power delivered; η_1 and η_2 of the propulsive efficiency and finally N_1 and N_2 of the revolutions. If desired, the power required can also be plotted, together with the power available.

Most airplanes can not fly at the low speed state of steady motion characterized by the values $(k_y)_2$; $(k_x/k_y)_2$; V_2; Q_2; N_2; η_2; $(L_u)_2$; $(L_m)_2$ because they do not have sufficient rudder areas to still keep control of the machine at this low speed.

When the airplane is considered flying at different increasing altitudes, the density δ decreases; that is, the value of the specific load p increases, the Q/P curve moves down and there will be a moment when it will become tangent to the k_x/k_y curve; that is, will have with the last only one point of intersection (see fig. 18). The value of the specific load that corresponds to this Q/P curve is the largest value p_c at which level flying of the airplane is still possible; and impossible for any larger value. The airplane has reached its *ceiling*, determined by the *ceiling specific load* p_c. In quadrant IV we can read the speed V_c at the ceiling, and in quadrant III, all the other ceiling characteristics.

It is easy to see and trace on the chart in quadrant IV the curve of the horizontal speeds at all altitudes. For this purpose it is sufficient to project on the corresponding speed curves the points of intersection of the k_x/k_y curve in quadrant I with the Q/P curves (see fig. 18). The speed curve to which this last curve is tangent gives again the value of the specific load p_c at the ceiling. The point at which these two curves are tangent separates the high speed states of steady motion from the low speed states.

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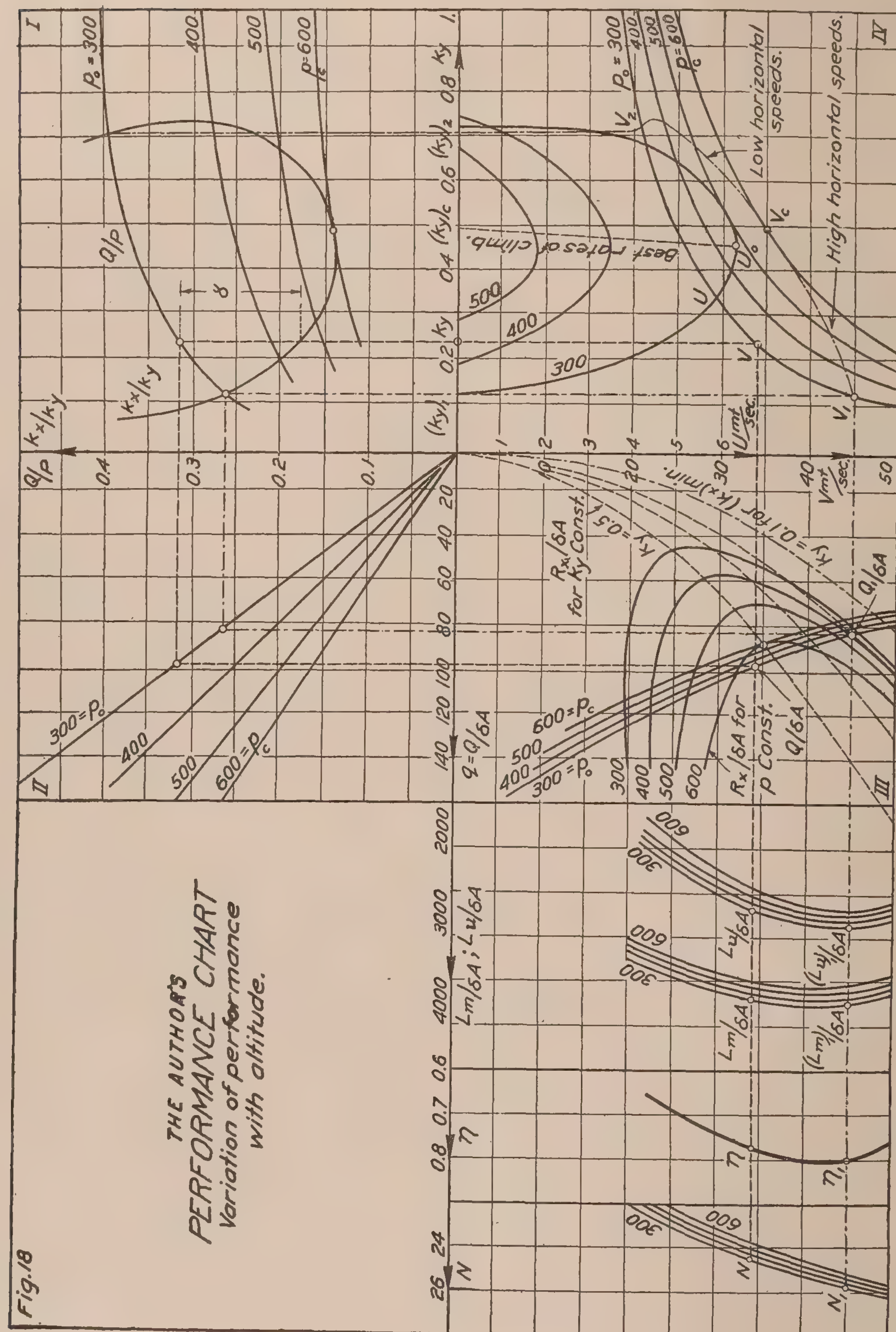


Fig. 18
THE AUTHOR'S
PERFORMANCE CHART
Variation of performance
with altitude.
Low horizontal speeds.
High horizontal speeds.
Best rates of climb.

Making use of our approximate equations (68) and (72), we may find the values of the horizontal speeds at all altitudes from the condition $Q/P = kx/ky$, that is

$$\frac{q_0}{p} - \frac{q_1}{k_y} = r\,k_y + t + \frac{\sigma}{k_y} \tag{73}$$

or

$$r\,k_y^2 + \left(t - \frac{q_0}{p}\right)k_y + \sigma + q_1 = 0 \tag{74}$$

Replacing k_y by its value from (65) we get

$$V^4(\sigma + q_1) + V^2(tp - q_0) + rp^2 = 0 \tag{75}$$

From which equation we find for the *horizontal high speeds*, the values:

$$V_1^2 = \frac{(q_0 - tp)}{2(\sigma + q_1)} + \sqrt{\frac{(q_0 - tp)^2}{4(\sigma + q_1)^2} - \frac{rp^2}{(\sigma + q_1)}} \tag{76}$$

and for the *horizontal low speeds*, the values:

$$V_2^2 = \frac{(q_0 - tp)}{2(\sigma + q_1)} - \sqrt{\frac{(q_0 - tp)^2}{4(\sigma + q_1)^2} - \frac{rp^2}{(\sigma + q_1)}} \tag{77}$$

The ceiling is given to us by the condition

$$V_1 = V_2 \tag{78}$$

that is,

$$\frac{(q_0 - tp)^2}{4(\sigma + q_1)^2} = \frac{rp^2}{(\sigma + q_1)} \tag{79}$$

or

$$(q_0 - tp)^2 = 4(\sigma + q_1)rp^2 \tag{80}$$

I shall remark here that the quantity tp is in general small (on account of t being small) in comparison with q_0, so that, as a first approximation, we can replace the condition (80) by the approximate condition

$$q_0^2 = 4(\sigma + q_1)rp^2 \tag{81}$$

from which we find:

$$p_c = \frac{P}{\delta_c A} = \frac{q_0}{2\sqrt{r(\sigma + q_1)}} \tag{82}$$

and finally for the ceiling density we get the value

$$\delta_c = \frac{2P\sqrt{r(\sigma + q_1)}}{Aq_0} \tag{83}$$

With the same approximation ($t \simeq 0$) the ceiling speed V_c has for its value

$$V_c^2 = \frac{q_0}{2(\sigma + q_1)} \tag{84}$$

and the corresponding value of the lift coefficient is equal to

$$(k_y)_c = \frac{p_c}{V_c^2} = \sqrt{\frac{\sigma + q_1}{r}} \tag{85}$$

It is easy to see that this last ceiling lift value is somewhat larger than the value of the lift

$$(k_y)_m = \sqrt{\frac{\sigma}{r}} \tag{86}$$

at which k_x/k_y is a minimum.

It is worth notice that the power required for horizontal flying

$$R_x V = k_x \delta A V^3 = P p^{1/2} \frac{k_x}{k_y{}^{3/2}} \simeq P p^{1/2} (r k_y{}^{1/2} + \sigma k_y{}^{-3/2})$$

has a minimum for a lift value $(k_y)_M$ given by

$$\frac{\partial}{\partial k_y}\left(\frac{kx}{k_y{}^{3/2}}\right) = 1/2 r k_y{}^{-1/2} - 3/2 \sigma k_y{}^{-5/2} = O$$

and equal to

$$(k_y)_M = \sqrt{\frac{3\sigma}{r}} \tag{87}$$

Usually q_1 is of the same order of magnitude as σ, thus $(k_y)_c$, being greater than $(k_y)_m$, has a value close to $(k_y)_M$.

As has been mentioned already, the ceiling of an airplane is characterized by the value p_c of the specific load, but the ceiling density δ_c depends upon the weight P, that is, upon the loading of the airplane. The loading of each airplane can be increased up to such a value that its ceiling will be dropped to the ground level. The value of this *limiting load* P_0 is given by

$$p_c = \frac{P}{\delta_c A} = \frac{P_0}{\delta_0 A} = \frac{q_0}{2\sqrt{r(\sigma + q_1)}} \tag{88}$$

where δ_0 is the ground level air density. Hence.

$$P_0 = P\frac{\delta_0}{\delta_c} = \frac{q_0 \delta_0 A}{2\sqrt{r(\sigma + q_1)}} \tag{89}$$

It is of interest to add to the specific thrust curves of quadrant III, also the two families of the following curves.

In the first place, the curves of $R_x/\delta A$ as a function of V with p as parameter. Since $R_x/R_y = k_x/k_y$, we obtain these curves directly on the chart by transferring the k_x/k_y curve of quadrant I to quadrant III by the aid of the speed curves of quadrant IV, and transfer lines of quadrant II. Each speed curve and straight line corresponding to a given value of p will allow us to get one $R_x/\delta A$ curve in quadrant IV for the same value of p (see fig. 18). Since

$$R_x \simeq k_y \delta A V^2\left(r k_y + t + \frac{\sigma}{k_y}\right) \text{ and } k_y V^2 = p,$$

these $R_x/\delta A$ curves have for their approximate equation

$$\frac{R_x}{\delta A} = V^2 (r k_y^2 + t k_y + \sigma)$$

or

$$\frac{R_x}{\delta A} = \frac{r p^2}{V^2} + p t + \sigma V^2 \tag{90}$$

With p as parameter, this equation represents a family of hyperbolas whose envelope is given by the relations

$$\frac{R_x}{\delta A} = \frac{r p^2}{V^2} + p t + \sigma V^2$$

$$\frac{\partial (R_x/\delta A)}{\partial p} = \frac{2\, r p}{V^2} + t = O$$

and has for its equation

$$\frac{R_x}{\delta A} = V^2\left(\sigma - \frac{t^2}{4r}\right) \tag{91}$$

In the second place, the curves of $R_x/\delta A$ as a function of V but with k_y as parameter. In order to plot these curves in quadrant III, it is sufficient to transfer the k_x/k_y curve of quadrant I

to quadrant III, keeping for each traced curve k_y constant, and making use of all the speed curves of quadrant IV, and transfer lines of quadrant II for each value of k_y. It is in such a way that these curves have been traced on figure 18. These curves have for their approximate valuation

$$\frac{R_x}{\delta A}=V^2(r\,k_y^2+t\,k_y+\sigma) \tag{92}$$

and represent a family of parabolas with the horizontal axis as the axis of symmetry and p as parameter. The parabola that corresponds to the minimum of $(r\,k_y^2+t\,k_y+\sigma)$, that is, to the minimum of R_x for a given V, which takes place for

$$k_y=\frac{-t}{2r}$$

has for its equation

$$\frac{R_x}{\delta A}=V^2\left(\sigma-\frac{t^2}{4r}\right) \tag{91}$$

and is the outermost of all the other parabolas of the family. This limiting parabola is the envelope of the family of hyperbolas defined by equation (90). (See fig. 18.)

The point in which a $Q/\delta A$ curve cuts a $R_x/\delta A$ curve with p as parameter gives us the horizontal speed for the corresponding value of p, and the $R_x/\delta A$ curve with k_y as parameter passing through that point gives the corresponding value of k_y.

B. CLIMBING.

If, starting from a given state of horizontal flying, the pilot by moving his stick varies k_y (see fig. 18) we can immediately see on the chart what value the path inclination γ will take. We shall be able easily to follow on the chart how γ, V, Q, L_u, L_m, η, and N will vary with variable k_y but constant p. A decrease of k_y will bring us to negative values of γ, the airplane will go down. An increase of k_y will cause the airplane to climb. The path inclination γ will pass through a maximum and decrease again until we reach the slow speed horizontal flight. Since for each value of k_y we know the corresponding values of γ and V, it is easy to compute the rate of climb

$$U=\gamma V$$

We can trace in quadrant IV the rate of climb curve as a function of k_y, plotting U on the V axis (but on a different scale). In such a way the U curves on the chart of figure 18 have been obtained for different values of p.

The maximum rate of climb decreases with increasing specific load until it becomes equal to zero at the ceiling.

Let us calculate the value of U using our approximate equations. We have (with $t\cong 0$)

$$\gamma=\frac{Q}{P}-\frac{k_x}{k_y}=\left(\frac{q_0}{p}-\frac{q_1}{k_y}\right)-\left(rk_y+\frac{\sigma}{k_y}\right) \tag{93}$$

or

$$\gamma=\frac{1}{k_y}\left[\frac{q_0}{p}k_y-rk_y^2-(\sigma+q_1)\right] \tag{94}$$

and with $k_y V^2=p$ the rate of climb is found equal to

$$\gamma=\frac{1}{k_y}\left[\frac{q_0}{p}k_y-rk_y^2-(\sigma+q_1)\right] \tag{95}$$

The rate of climb will be a maximum for

$$\frac{\partial U}{\partial k_y}=0$$

that is

$$k_y^2+\frac{q_0}{pr}k_y-\frac{3(\sigma+q_1)}{r}=0$$

which gives

$$k_y=-\frac{q_0}{2pr}+\sqrt{\frac{q_0^2}{4p^2r^2}+\frac{3(\sigma+q_1)}{r}}$$

or, on account of (82),

$$k_y=\frac{q_0}{2pr}\left[-1+\sqrt{1+3\frac{p^2}{p_c^2}}\right] \tag{96}$$

This value of k_y, when introduced in (95), will give the maximum value of the rate of climb U corresponding to each value of the specific load p.

The value of k_y at which the "best climb" takes place in general increases with the altitude and reaches its largest value at the ceiling. But the ceiling value of k_y is generally not greatly different from all the set of values given by (96); and since, on the other hand, a function does not change much near its maximum the rate of climb computed will not be greatly different from its maximum if it is assumed to take place at a constant lift equal to the ceiling value,

$$(k_y)_c=\sqrt{\frac{\sigma+q_1}{r}}$$

with which the ceiling will always be reached.

The rate of climb, with k_y having this value, is equal to

$$U=\frac{a}{p^{1/2}}-bp^{1/2} \tag{97}$$

where

$$a=\frac{q_0r^{1/4}}{(\sigma+q_1)^{1/4}}=\frac{q_0}{\sqrt{(k_y)_c}};\quad b=2r^{3/4}(\sigma+q_1)^{1/4}=2r\sqrt{(k_y)_c} \tag{98}$$

since, on account of (82) and (85), we have

$$q_0=2rp_c(k_y)_c, \tag{99}$$

we can put the expression (97) of the rate of climb in the form

$$U=\sqrt{2rq_0}(\sqrt{p_c/p}-\sqrt{p/p_c}) \tag{100}$$

The rate of climb at ground level is equal to

$$U_0=\sqrt{2rq_0}(\sqrt{p_c/p_0}-\sqrt{p_0/p_c}) \tag{101}$$

Let us calculate, under the condition of a climb with $k_y=(k_y)_c$ the time of climb t, from ground level, characterized by the value p_0 of the specific load, up to the level of specific load $p<p_c$.

According to the relations (45) and (47), we have for an isothermic atmosphere

$$dp=d\sigma RT=-\sigma dH$$

where here p is the atmospheric pressure. Thus

$$dH=-\frac{d\sigma}{\sigma}RT$$

But, since

$$U=\frac{d\Pi}{dt}\text{ and }\frac{d\sigma}{\sigma}=\frac{d\delta}{\delta}=\frac{d\left(\frac{P}{pa}\right)}{\left(\frac{P}{pa}\right)}=-\frac{dp}{p}$$

where here p is again the specific load, we find

$$dt=\frac{d\Pi}{U}=\frac{RTdp}{pU} \tag{102}$$

Making use of the value (97) for U and integrating, we get

$$t=RT\int_{p_0}^{p}\frac{dp}{ap^{1/2}-bp^{3/2}}$$

Substituting $p^{1/2}=z$ we get

$$t=2RT\int_{z_0}^{z}\frac{dz}{a-bz^2}$$

or

$$t=\frac{RT}{\sqrt{ab}}lg_e\frac{(\sqrt{ab}+bz)(\sqrt{ab}-bz_0)}{(\sqrt{ab}-bz)(\sqrt{ab}+bz_0)}$$

On account of (98) and (82) we have

$$\sqrt{ab}=bz_c=b\sqrt{p_c}=\sqrt{2rq_0}$$

and thus

$$t=\frac{RT}{\sqrt{2rq_0}}lg_e\frac{(1+\sqrt{p/p_c})(1-\sqrt{p_0/p_c})}{(1-\sqrt{p/p_c})(1+\sqrt{p_0/p_c})} \tag{103}$$

Taking account of (102) we finally have:

$$t=\frac{RT}{U_0}\left(\sqrt{p_c/p_0}-\sqrt{p_0/p_c}\right)lg_e\frac{(1+\sqrt{p/p_c})(1-\sqrt{p_0/p_c})}{(1-\sqrt{p/p_c})(1+\sqrt{p_0/p_c})} \tag{104}$$

The last two formulae give us the time of climb in an isothermic atmosphere of temperature T, from ground level up to the level of specific load p. For $p=p_c$ the time of climb turns out to be infinite. There is nothing atonishing in this last fact because the reaching of the ceiling is an asymptotic phenomenon.

Formula (104) applied to the isothermic atmosphere of zero degrees centigrade T=273° and the time t expressed in minutes, gives

$$t_{\min}=\frac{307}{U_0}(\sqrt{p_c/p_0}-\sqrt{p_0/p_c})\ lg_{10}\frac{(1+\sqrt{p/p_c})\ (1-\sqrt{p_0/p_c})}{(1-\sqrt{p/p_c})\ (1+\sqrt{p_0/p_c})} \tag{105}$$

Let us finally find the direct relation between the rate of climb U—with $k_y=(k_y)_c$—and the altitude H in the isothermic atmosphere.

According to (102) we have

$$dH=RT\frac{dp}{p} \tag{106}$$

Integrating this last relation, from ground level up to the altitude of specific load p, we get

$$H=RT\,lg\frac{p}{p_0} \tag{107}$$

The ceiling altitude is equal to

$$H_c = RT\, lg \frac{p_c}{p_o} \tag{108}$$

Subtracting (107) from (108) we find:

$$(H_c - H) = RT\, lg\; p_c/p = z \tag{109}$$

where z is the altitude below the ceiling. From this relation we get

$$\sqrt{p_c/p} = e^{\frac{z}{2RT}};\quad \sqrt{p/p_c} = e^{\frac{-z}{2RT}} \tag{110}$$

Substituting these last values in (100) we find

$$U = \sqrt{2rq_o}\left[e^{\frac{z}{2RT}} - e^{\frac{-z}{2RT}}\right] = 2\sqrt{2rq_o}\,\mathrm{Sinh}\frac{z}{2RT} \tag{111}$$

Developing Sinh in series, and keeping only the first term, which is a sufficient approximation even for the highest altitude actually reached, we find:

$$U \cong \frac{z\sqrt{2rq_o}}{RT} = \frac{(H_c - H)\sqrt{2rq_o}}{RT} \tag{112}$$

For the ground level

$$U_o = \frac{H_c\sqrt{2rq_o}}{RT} \tag{113}$$

We thus also have

$$U = U_o\left(1 - \frac{H}{H_c}\right) \tag{114}$$

The two formulae (114) and (112) show us that as a first approximation, the rate of climb is a *linear function of the altitude H.*

As a result of long experience in the measurement of rates of climb of airplanes, in free flight, it has always been observed that the rates of climb appeared to be linear functions of the latitude. This fact brings us to the conclusion that all the assumptions we have made in the foregoing really constitute an approximation practically fully sufficient and that, to the approximation with which we actually measure rates of climb, the climbing takes place as if the atmosphere was isothermic. One can thus see that the isothermic atmosphere appears to be quite as good as any other standard atmosphere, but in addition the isothermic atmosphere has all the advantages of being the simplest conventional atmosphere.

It is easy to deduce from formula (114) the time of climb as a function of the altitude. We have,

$$dt = \frac{dH}{U} = \frac{H_c}{H_o}\frac{dH}{H_c - H} \tag{115}$$

and integrating we get:

$$t = \frac{H_c}{U_o}\, lg_e \frac{H_c}{H_c - H} \tag{116}$$

The formula:

$$t_{\min} = 0,\ 0384\ \frac{H_c}{H_o}\, lg_{10} \frac{H_c}{H_c - H} \tag{117}$$

gives the time of climb in minutes, from ground level up to the altitude $H < H_c$. If we take conventionally $H = 0,\ 95\ H_c$ we find

$$t_{\min} = 0,\ 05\frac{H_c}{U_o} \tag{118}$$

We will get the exact expression for the time of climb in an isothermic atmosphere, if we use in equation (115) the value (111) of the rate of climb. We thus obtain

$$dt = \frac{dH}{2\sqrt{2rq_o}\,\text{Sinh}\,\frac{H_c - H}{2RT}}$$

and integrating we find:

$$t = \frac{-RT}{\sqrt{2rq_o}} \int_0^H \frac{d\left(\frac{H_c - H}{2RT}\right)}{\text{Sinh}\,\frac{H_c - H}{2RT}} = \frac{RT}{\sqrt{2rq_o}}\, lg_e \frac{\text{tgh}\left(\frac{H_c}{4RT}\right)}{\text{tgh}\left(\frac{H_c - H}{4RT}\right)}$$

or, since

$$U_o = 2\sqrt{2rq_o}\,\text{Sinh}\,\frac{H_c}{2RT}$$

we finally find:

$$t = \frac{2RT\,\text{Sinh}\frac{H_c}{2RT}}{U_o}\, lg \frac{\text{tgh}\left(\frac{H_c}{4RT}\right)}{\text{tgh}\left(\frac{H_c - H}{4RT}\right)} \tag{119}$$

This last formula gives the exact value of the time of climb in an isothermic atmosphere of absolute temperature T, from ground level up to the altitude $H < H_c$ where U_o is the rate of climb at the ground, and the whole climb is supposed made at a constant value of the lift coefficient k_y equal to its ceiling value $(k_y)_c$.

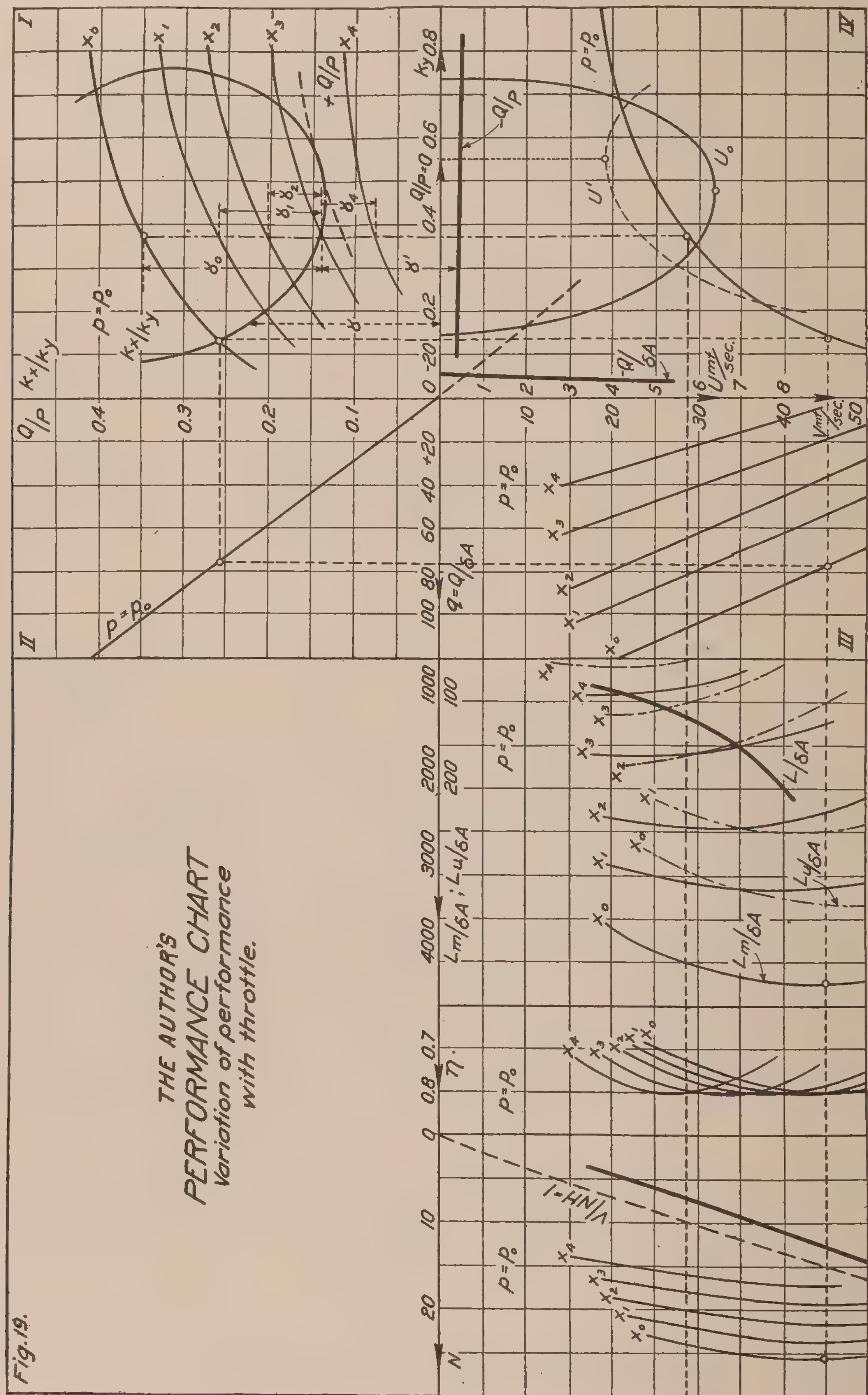
C. ENGINE THROTTLING AND GLIDING.

Until now we have considered the flight of the airplane at full throttle opening and variable altitude. Let us now consider the flight of the airplane at constant altitude, for example close to the ground level, but with variable throttle. Returning to our chart, we see that the k_x/k_y curve of quadrant I is not affected by the throttle; the speed curves of quadrant IV and transfer lines of quadrant II are also unaffected by the throttle; but the specific thrust curves of quadrant II depend directly upon the throttle opening. Proceeding as was indicated in the chapter dealing with the engine-propeller system, it will be easy to compute the curves of specific thrust as a function of the speed for different throttle openings: x_0 (full throttle), x_1, x_2, x_3 ------. A set of such specific thrust curves have been represented in figure 19. A variation of the throttle means a variation of the engine power, which brings with it a shifting of the specific thrust curves. For a given speed the specific thrust drops when the throttle is reduced. In the extension of quadrant III have been plotted the curves of $L_u/\delta A$, $L_m/\delta A$, η, and N, corresponding to the different throttle openings x_0, x_1, x_2, x_3, x_4 ------. Quadrant III and extension thus give us now a complete representation of the engine-propeller system characteristics for variable throttle openings.

Once the set of specific thrust curves with variable throttle is established, it is easy to plot in quadrant I the corresponding Q/P curves, making use of the speed curve and transfer line of the altitude considered, the throttle opening being now our parameter. The chart thus completed, the influence of the engine throttling on the flight of the airplane becomes self-evident.

Let us consider the airplane first flying at full throttle opening x_0, at a certain value of k_y, the inclination of the flying path to the horizontal having the value γ. (See fig. 19.) The corresponding value of the speed is given by the speed curve of quadrant IV, and the other flying characteristics can be read from quadrant III and extension. If we now begin to throttle the engine, the path inclination will successively take the values γ_1, γ_2; and for the throttle opening x_3, for example, the flight will already be horizontal. The speed will *not be affected*—provided the action of the slip-stream on the rudders can be neglected, as has

been already explained—because for invariable k_y and the same altitude, characterized by the value p_0 of the specific load, we read the same speed on the same speed curve. But all the other flying characteristics vary with the throttle, as can be seen directly from quadrant III and extension. For the throttle opening x_4 the path inclination γ_4 will be negative; that is, we will be descending with motor on. For each altitude there is a throttle opening, for which



the Q/P curve is tangent in quadrant I to the k_x/k_y curve (curve in dashes on fig. 19). At this throttle opening, the altitude considered is the ceiling. For any smaller throttle opening, the airplane will always be descending.

Let us now consider the engine power to be cut off, the airplane can only be descending; it is said to be *gliding*. When gliding, the propeller generally works as a windmill and thus

will give us a negative thrust. We can plot for the altitude considered the $-Q/P$ curve on the negative extension of the axis of quadrant I (see fig. 19); we shall then be able to read, for each value of k_y, the exact inclination γ' of the airplane gliding trajectory, between the k_x/k_y curve and this $-Q/P$ curve.[1] The gliding speed will be read in quadrant IV on the same speed curve. For large values of γ', corresponding to small values of k_y, we shall get a more accurate value of the speed in gliding by taking for its value $V\sqrt{\text{Cos}\ \gamma'}$, as has been already explained. The rate of descent is equal to

$$U' = V\gamma'$$

and can be computed easily for each value of k_y, as V and γ^1 are known. The curve of the rate of descent has been plotted in dashes in quadrant IV of figure 19, using for it the same scale as for the rate of climb. It is easy to see that the minimum of the rate of descent U' does not coincide with the minimum of γ', the last being a minimum for k_x/k_y minimum; the value of k_y that makes U' a minimum being larger than the one that makes γ' minimum. We now see that, if an airplane is gliding at a certain value of k_y, and if we slightly open the throttle, it is not the speed that is changed, but only the gliding path; the angle γ is decreased, and the rate of descent is reduced in proportion.

We can transfer the $-Q/P$ curve of Quadrant I as well as the $-Q/\delta A$ curve of Quadrant III, by the aid of the speed curve and transfer line, using for that purpose the negative extension of the specific thrust axis. The $-Q/\delta A$ curve thus obtained is represented by a thick line on figure 19. By the aid of this last curve we can deduce the mechanical losses of the engine when the law of variation of the engine revolutions with the speed in gliding is known. Such a curve of the revolutions in gliding as a function of V is represented by a thick line at the end of the extended Quadrant III. The fact is, that when gliding, the propeller generally rotates at a much less number of revolutions than its regular number of revolutions and thus works as a windmill, with a relatively high value of the advance V/N. But under such conditions, the efficiency of η' of the propeller, working as a windmill, will be very closely equal to

$$\eta' = \frac{NH}{V} \tag{120}$$

where H is the *effective* propeller pitch.[2] Thus, if for a given value of the gliding speed V we know the corresponding $-Q/\delta A$ and N, we have the power absorbed by the propeller working as a windmill equal to QV, and the power delivered by the propeller to the engine and absorbed by the last as mechanical losses L equal to

$$L = \eta'\, QV \tag{121}$$

or

$$\frac{L}{\delta A} = \eta'\, V\frac{Q}{\delta A} = NH\frac{Q}{\delta A} \tag{122}$$

The curve of $L/\delta A$ is represented in the extension of Quadrant III by a thick line. If in addition we assume the mechanical losses L to be proportional to the revolutions, that is, we put $L = fN$, we get

$$Q = f/H \tag{123}$$

The negative thrust given by the propeller when gliding would then appear to be constant at all speeds. Practically, the negative thrust in a glide does not appear to vary greatly.

If we wish, when gliding, the propeller thrust to be exactly equal to zero, we must adjust our throttle in such a way that the ratio V/NH is nearly equal to unity, because, as is known,[3] for $V/NH \cong 1$ the propelller thrust is equal to zero. The propeller revolutions will then vary proportionally with the speed according to the relation

$$N = V/H \tag{124}$$

[1] I owe this last remark to my assistant, Mr. W. F. Gerhardt, aeronautical engineer at McCook Field.

[2] See "General Theory of Blade Screws," by Dr. G. de Bothezat, Chapter II.

[3] For an exact discussion of this question see: "General Theory of Blade Screws," above mentioned, Chapter II.

where H is the propeller pitch. The curve of N as a function of V will be a straight line—represented in dashes at the end of Quadrant III extension of figure 19. When gliding under such conditions, our Q/P curve will be reduced to the k_y axis, as its ordinates will all be equal to zero. The inclination of our gliding path will now be measured simply by the ordinates of the curve (see fig. 19). We thus see that when gliding, the propeller acts as if the airplane drag was increased, and power has to be applied in order to eliminate this effect.

One can now see how complete a picture of the properties of an airplane in steady motion is given by the chart developed in this paper, and with what ease this chart allows us to follow the variations and mutual inter-relations of all the quantities involved in the problem.

The approximate equations we have deduced for all the curves of the chart may be used in many cases for a first checking, but attention must be paid to all the assumptions made in deducing these approximations. The approximate equations applied under carefully considered limitations give very good results for some problems, but for any general and broad discussion connected with the study of an airplane in steady motion the general method of the chart must be used.

REPORT No. 97.

GENERAL THEORY OF THE STEADY MOTION OF AN AIRPLANE.

By George de Bothezat.

PART V.

PERFORMANCE PREDICTION.

Prediction of airplane performance is at present based more or less on wind tunnel tests The purpose of this chapter is to show how such performance prediction can be based chiefly on data obtained from those free flight tests to which airplanes are usually submitted. I shall thus, in the first place, show how to deduce from regular free flight tests the data necessary for performance prediction, and, in the second place, show how to use these data in order to predict the performance.

1. COLLECTING THE NECESSARY DATA.

We have seen in the foregoing chapter that, as a first approximation, considering $t \cong 0$, the whole airplane performance depends upon the four coefficients r, σ, q_0, and q_1. Two of these coefficients, r and σ, characterize the airplane itself; the two other coefficients, q_0 and q_1 characterize the engine-propeller system. All the free flight characteristics can be expressed a, functions of these four coefficients.

Among the relations deduced in the foregoing chapter, we had:

The high horizontal speeds at all altitudes

$$V^2 = \frac{q_0}{2(\sigma+q_1)} + \sqrt{\frac{q_0^2}{4(\sigma+q_1)^2} - \frac{rp^2}{(\sigma+q_1)}} \tag{125}$$

or

$$V^2 = \frac{q_0}{2(\sigma+q_1)}\left[1+\sqrt{1-\frac{4rp^2\ (\sigma+q_1)}{q_0^2}}\right] \tag{126}$$

where $p = P/\delta A$ is the specific load which defines the altitude considered.

The ceiling value for the specific load

$$p_c = \frac{P}{\delta_c A} = \frac{q_0}{2\sqrt{r(\sigma+q_1)}} \tag{127}$$

The rate of climb

$$U = U_0\, \frac{\sqrt{p_c/p} - \sqrt{p/p_c}}{\sqrt{p_c/p_0} - \sqrt{p_0/p_c}} = U_0\left(1 - \frac{H}{H_c}\right) \tag{128}$$

where U_0 is the rate of climb at ground level, equal to

$$U_0 = \sqrt{2rq_0}\ (\sqrt{p_c/p_0} - \sqrt{p_0/p_c}) \tag{129}$$

From relation (127) we get

$$\frac{p^2}{p_c^2} = \frac{4(\sigma+q_1)rp^2}{q_0^2} \tag{130}$$

Substituting in (126), and since

$$V_c^2 = \frac{q_0}{2(\sigma+q_1)} = \frac{2rp_c^2}{q_0} \tag{131}$$

we find:

$$V^2 = V_c^2\ [1 + \sqrt{1 - (p/p_c)^2}] \tag{132}$$

For a given airplane the minimum information that we can have about it is the knowledge of its horizontal speed V_0 at ground level, its rate of climb U_0 at ground level, and its ceiling H_c.

Let us try to find the expressions for the four coefficients r, σ, q_0, and q_1 as functions of these last quantities V_0, U_0, and H_c. We shall then be able to compute the four characteristic coefficients r, σ, q_0, and q_1 from the ordinary free flight tests.

When the ceiling H_c is known, we can compute the ceiling specific load p_c from the relation

$$H_c = RTlg\frac{p_c}{p_0} \tag{133}$$

which gives

$$p_c = p_0\, C^{\frac{H_c}{RT}} \tag{134}$$

Knowing p_c, we find

$$V_c^2 = \frac{V_0^2}{1+\sqrt{1-(p_0/p_c)^2}} \tag{135}$$

Knowing V_c, we find

$$(k_y)_c = \frac{p_c}{V_c^2};\quad (k_y)_0 = \frac{p_0}{V_0^2} \tag{136}$$

Knowing p_c and U_0 from (129) and (131), we find

$$\sqrt{2rq_0} = \frac{U_0}{\sqrt{p_c/p_0}-\sqrt{p_0/p_c}};\quad \sqrt{\frac{2r}{q_0}} = \frac{V_c}{p_c}$$

and solving these two equations in r and q_0, we find

$$r = \frac{U_0 V_c}{2p_c[\sqrt{p_c/p_0}-\sqrt{p_0/p_c}]} \tag{137}$$

and

$$q_0 = \frac{U_0 p_c}{V_c[\sqrt{p_c/p_0}-\sqrt{p_0/p_c}]} \tag{138}$$

and knowing q_0 from (131), we get

$$(\sigma+q_1) = \frac{q_0}{2V_c^2} = \frac{p_c U_0}{2V_c^3[\sqrt{p_c/p_0}-\sqrt{p_0/p_c}]} \tag{139}$$

It is clear that if only three conditions are given us which we have assumed to be the values of V_0, U_0, and H_c we can find only three relations containing r, σ, q_0, and q_1 as functions of V_0, U_0, and H_c, and we have just found the expressions for r, q_0, and $(\sigma+q_1)$ in terms of V_0, U_0, and p_c. A fourth condition has thus to be put forward in order to specify fully the problem But it must be remarked that so far as we intend to predict only the horizontal self-speeds at all altitudes, the rates of climb at all altitudes, the ceiling and the time of climb, we do not need to know separately the coefficients σ and q_1 because, as can be seen from the relations (125), (128), (133), and (117), the quantities V, U, H_c, and $t_{\min}$ are functions only of r, q_0, and $(\sigma+q_1)$. But as soon as the propeller efficiency η_0 for horizontal flying at ground level is known, we shall immediately be able to find q_1 and thus know the separate values of σ and q_1. In effect we have

$$\eta_0 L_0 = (q_0 - q_1 V_0^2)\delta_0 A V_0 \tag{140}$$

where L_0 is the power delivered by the engine for horizontal flight at ground level, which in any case must be considered as a known quantity. From the last relation we get directly

$$q_1 = \frac{q_0}{V_0^2} - \frac{\eta_0 L_0}{\delta_0 A V_0^3} = \frac{1}{V_0^2}\left(q_0 - \frac{\eta_0 L_0}{PV_0}p_0\right)$$

or, since $\eta_0 L_0 = Q_0 V_0$, where Q_0 is the propeller thrust for horizontal flying at ground level, and using the notation

$$y_0 = \frac{(k_x)_0}{(k_y)_0} = \frac{Q_0}{P}$$

we finally get

$$q_1 = \frac{q_0 - y_0 p_0}{V_0^2} \tag{141}$$

The value of the coefficient q_1 once found, we can find the value of σ by the aid of relation (139).

Thus, when we know:

$$V_0,\ U_0,\ H_c,\ \eta_0, \text{ and } L_0$$

we can deduce immediately the values of the characteristic coefficients r, σ q_0 and q_1 that correspond to the airplane considered.

Unfortunately the propeller efficiency is the quantity that most generally is the least known among the quantities affecting airplane performance. That is why it may be of interest to attempt to check this efficiency, when unknown, even only to a first approximation.

We have seen in the foregoing that the value of the lift coefficient at which the ceiling is reached

$$(k_y)_c = \sqrt{\frac{\sigma + q_1}{r}}$$

is generally included between

$$(k_y)_m = \sqrt{\frac{\sigma}{r}} \text{ and } (k_y)_M = \sqrt{\frac{3\sigma}{r}}$$

that is, included between the lift value at which k_x/k_y is minimum and the lift value at which $k_x/k_y^{3/2}$ is a minimum ("power required minimum"). It is clear on the other hand that the value of q_1 depends upon the propeller of the engine-propeller system considered. There is advantage in selecting such a propeller as would give us $q_1 = 2\sigma$, because we would then have $(k_y)_c = (k_y)_M$, the power required would be a minimum at the ceiling and the highest ceiling would be reached with the power available. But it is difficult to expect that each airplane is fitted with the best climbing propeller and that is why in general $q_1 < 2\sigma$. Let us designate by n the ratio

$$\frac{q_1}{\sigma} = n \tag{142}$$

It is easy to see that for high ceiling airplanes the ratio n will be close to the value 2, and for average ceiling machines closer to 1. By making in (139) the coefficient $q_1 = n\sigma$ we get

$$\sigma = \frac{p_c U_0}{2 V_c^3 (n+1)[\sqrt{p_c/p_0} - \sqrt{p_0/p_c}]} \tag{143}$$

$$q_1 = \frac{n p_c U_0}{2 V_c^3 (n+1)[\sqrt{p_c/p_0} - \sqrt{p_0/p_c}]} \tag{144}$$

It is by estimating the value of n that one can decide to a first approximation upon the relative values of σ and q_1. It must be remarked that it is only the value of the propeller efficiencies that will be affected by the value adopted for n, because all other performance characteristics, as has already been mentioned, depend only upon $(\sigma + q_1)$ and thus are independent of the value of n.

Let us designate by N', η', and L'_m the number of revolutions, the propeller efficiency and power delivered at ground level for $k_y = (k_y)_c$—at that moment the airplane is climbing with the rate of climb U_0 and has the self-speed V; let us designate by N'', η'' and L''_m the number of revolutions, the propeller efficiency and power delivered at the ceiling; and by N_0 the number of revolutions for horizontal flying at ground level; and let us try to find the expressions for η_0, η', and η'' in terms of the characteristics of the airplane performance. We have:

$$\eta_0 L_0 = (k_x)_0\, \delta_0\, A V_0^3 = \frac{(k_x)_0}{(k_y)_0} (k_y)_0\, \delta_0\, A V_0^2 \cdot V_0 = \frac{(k_x)_0}{(k_y)_0} P V_0$$

$$\eta' L'_m = (k_x)_c\, \delta_0\, A V^3 + P \gamma V = \frac{(k_x)_c}{(k_y)_c} P V_c \frac{p_c}{p_0} \frac{V^3}{V_c^3} + P U_0$$

$$\eta'' L''_m = (k_x)_c\, \delta_c\, A V_c^3 = \frac{(k_x)_c}{(k_y)_c} P\, V_c$$

On account of

$$(k_y)_0=\frac{p_0}{V_0^2};\ (k_y)_c=\frac{p_c}{V_c^2}=\frac{p_0}{V^2};\ V^2=V_c^2\frac{p_0}{p_c};\ \frac{(k_x)_0}{(k_y)_0}=y_0;\ \frac{(k_x)_c}{(k_y)_c}=y_c$$

and

$$L'_m\cong\frac{N'}{N_0}L_0;\ L''_m\cong\frac{N''L_0\,p_0}{N_0\,p_c}$$

we find

$$\eta_0=\frac{PV_0y_0}{L_0} \tag{145a}$$

$$\frac{N'}{N_0}\eta'=\frac{PV_cy_c\sqrt{p_0/p_c}+P\ U_0}{L_0} \tag{146a}$$

$$\frac{N''}{N_0}\eta''=\frac{PV_c\,y_c}{L_0\left(\frac{p_0}{p_c}\right)} \tag{147a}$$

On account of

$$y_0=r(k_y)_0+\frac{\sigma}{(k_y)_0}=\frac{rp_0}{V_0^2}+\frac{\sigma\ V_0^2}{p_0};\ y_c=r(k_y)_c+\frac{\sigma}{(k_y)_c}=\frac{rp_c}{V_c^2}+\frac{\sigma V_c^2}{p_c}$$

and replacing r and σ by their values (137) and (143) we find:

$$\eta_0=\frac{PU_0}{L_0}\rho_0 \tag{145b}$$

$$\frac{N'}{N_0}\eta^1=\frac{PU_0}{L_0}\rho' \tag{146b}$$

$$\frac{N''}{N_0}\eta''=\frac{PU_0}{L_0}\rho'' \tag{147b}$$

where

$$\rho_0=\frac{(1+\sqrt{1-z_0^4})^{1/2}}{2z_0(1-z_0^2)}\left[\frac{z_0^4}{1+\sqrt{1-z_0^4}}+\frac{1+\sqrt{1-z_0^4}}{1+n}\right]$$

$$\rho^1=\frac{1-\frac{n}{2(1+n)}z_0^2}{1-z_0^2}$$

$$\rho''=\frac{\frac{n+2}{2(1+n)}}{z_0(1-z_0^2)}$$

and

$$z_0=\sqrt{p_0/p_c}$$

The curves of ρ_0, ρ', and ρ'' as functions of z_0^2 for $n=1$ and $n=2$ have been represented on figure 20. It is easy to see from this figure that in the foregoing formulæ the value of n has a sensible influence on the value of ρ_0, a somewhat less influence on ρ'', but that ρ' turns out to depend only very slightly upon n, especially for small values of z_0^2. that is for airplanes of high ceilings. Thus by the aid of formula (145b) it is difficult to check the efficiency η_0; by the aid of formula (147b) the efficiency η'' can be checked only to a rough approximation; but the efficiency η' can be fairly well predicted by the aid of the formula (146b).

Thus from the general knowledge of the airplane performance it is only the propeller efficiency η' at the best rate of climb that we are able to check.

Some engines show abnormal deviations for their power from the proportionality to the revolutions and density. For such engines, in the expression of the powers L'_m and L''_m as depending upon L_o, special correction factors have to be introduced in order to take account of these abnormal deviations.

We thus see that in the question of airplane performance prediction, it is the quantities connected with the power unit that will be predicted with less accuracy than all the other flying characteristics and this exclusively on account of the fact that exact information concerning the engine-propeller system is in general lacking.

A last remark has to be made concerning the coefficients r and σ. The drag of the airplane wings considered alone can, to a first approximation, be taken equal to

$$\delta A V^2(rk_y^2+s)$$

where r and s are two characteristic coefficients of the wings that can be deduced by the method of least squares from the experimental drag-lift curve. The drag of the airplane parasite

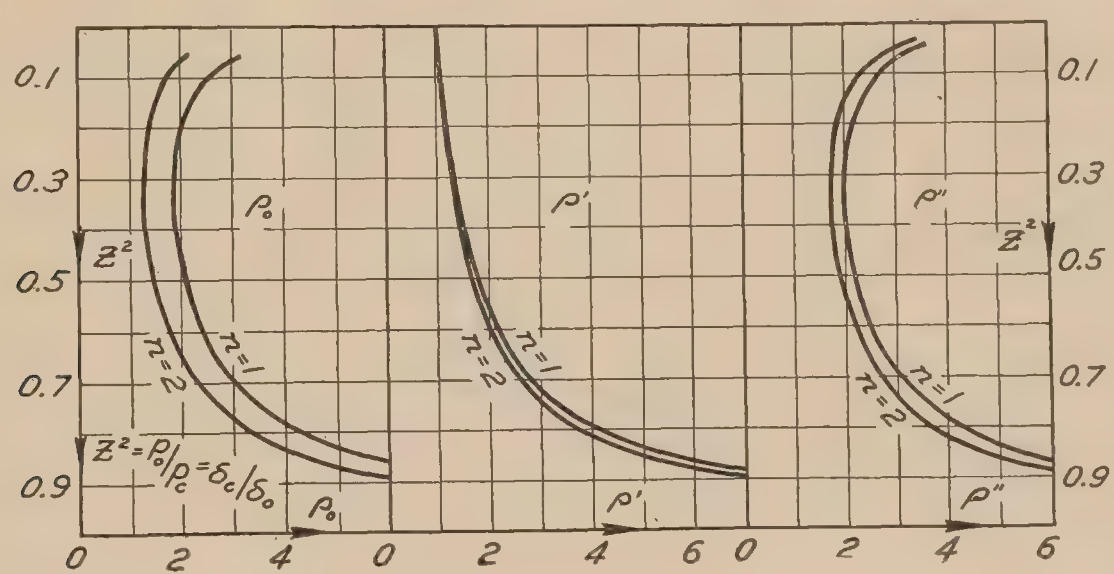


Fig. 20.

resistance considered alone can be taken equal to $k\delta aV^2$ where a is the *equivalent area* of the parasite resistance and $k=0{,}64$ the coefficient of air resistance for the orthogonal motion of a flat plate. The drag R_x of the whole airplane will then appear to us as equal to

$$R_x=\delta AV^2(rk_y^2+\sigma)=\delta AV^2(rk_y^2+s)+k\delta aV^2$$

$$=\delta AV^2\left(rk_y^2+s+k\frac{a}{A}\right)$$

We thus see that

$$\sigma=s+0{,}64\frac{a}{A} \tag{147}$$

The coefficient r thus appears to depend chiefly, to a first approximation, upon the wing shape alone, the coefficient σ to depend chiefly upon the ratio

$$\frac{a}{A}=\frac{\text{equivalent area of parasite resistance}}{\text{wing area}}$$

Knowing σ we can deduce the value of the ratio a/A, when the value of s will be known

$$\frac{a}{A}=1{,}56(\sigma-s) \tag{148}$$

Summing up the foregoing, we can say: The data obtained from average airplane tests, which usually are

The horizontal speed at ground level V_o,

The rate of climb at ground level U_o,

The ceiling H_c,

allow us to deduce the values of the four characteristic coefficients of the airplane in steady motion, when the efficiency η_o of its propeller is known. Otherwise, only by estimation can we evaluate the separate values of σ and q_1. These characteristic coefficients are equal to

$$r=\frac{U_oV_c}{2p_c[\sqrt{p_c/p_o}-\sqrt{p_o/p_c}]};\ q_o=\frac{U_op_o}{V_c[\sqrt{p_c/p_o}-\sqrt{p_o/p_c}]}$$

$$(\sigma+q_1)=\frac{p_cU_o}{2V_c^3[\sqrt{p_c/p_o}-\sqrt{p_o/p_c}]}$$

$$q_1=\frac{1}{V_o^2}(q_o-y_op_o)$$

with

$$p_o=\frac{P}{\delta_oA};\ y_o=\frac{(k_x)_o}{(k_y)_o}=\frac{\eta_oL_o}{PV_o};\ p_c=p_oe^{\frac{\mathrm{He}}{\mathrm{RT}}};\ V_c^2=\frac{V_o^2}{1+\sqrt{1-(p_o/p_c)^2}}$$

and

$$\frac{a}{A}=1{,}56(\sigma-s)$$

Making use of the notations:

$$z_o=\sqrt{p_o/p_c}=\sqrt{\delta_c/\delta_o} \tag{149}$$

$$r_1=\frac{z_o^3}{2(1-z_o^2)(1+\sqrt{1-z_o^4})^{1/2}} \tag{150}$$

$$q_{10}=\frac{(1+\sqrt{1-z_o^4})^{1/2}}{z_o(1-z_o^2)} \tag{151}$$

$$\sigma_1+q_1=\frac{(1+\sqrt{1-z_o^4})^{3/2}}{2z_o(1-z_o^2)} \tag{152}$$

we can write

$$r=\frac{V_oU_o}{p_o}r_1;\ q_o=\frac{U_op_o}{V_o}q_{10};\ (\sigma+q_1)=\frac{U_op_o}{V_o^3}(\sigma_1+q_1) \tag{153}$$

In order to simplify the computation of the coefficients r, q_o and $(\sigma+q_1)$ we have drawn on figure 21 the curves of r_1, q_{o1} and (σ_1+q_1) as functions of $z_o^2=p_o/p_c=\delta_c/\delta_o$, which allow us to read directly the values of these coefficients once the ceiling to which the airplane considered can climb is known.

Proceeding as above described, we can compute from observed performances the characteristic coefficients of the airplane's steady motions and thus collect values of these coefficients deduced from actual free flight tests.

Having deduced from tested airplanes the values that the characteristic coefficients can actually take, the prediction of performances is made as follows.

2. PREDICTING THE PERFORMANCE.

Two cases have to be distinguished:

In the first case, the airplane is considered as already built and tested, and the values of V_o, U_o and H_c experimentally found. The prediction of the complete performance is requested?

Knowing H_c we compute

$$z_o^2=p_o/p_c=e^{\frac{-\mathrm{H_c}}{\mathrm{RT}}}=e^{\frac{-\mathrm{H_c}}{8000}}$$

Having found the value of z_0^2 we compute the ceiling flying speed

$$V_c^2 = \frac{V_0^2}{1+\sqrt{1-z_0^4}}$$

The horizontal speeds at all altitudes are then found to be equal to

$$V^2 = V_c^2\,(1+\sqrt{1-z^4}) = V_0^2\,\frac{1+\sqrt{1-z^4}}{1+\sqrt{1-z_0^4}}$$

with $z^2 = p/p_c$ each altitude considered being defined by the value of the corresponding specific load

$$p = \frac{P}{\delta A}$$

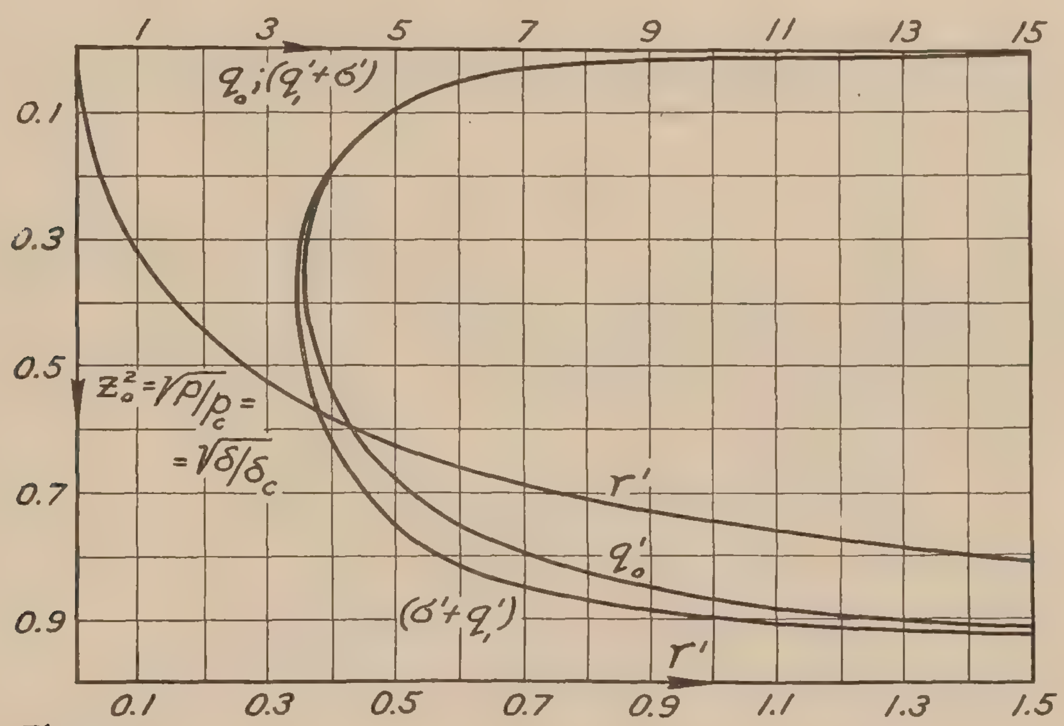


Fig. 21.

The rates of climb at all altitudes are equal to

$$U = U_0\,\frac{(1-z^2)z_0}{(1-z_0^2)z} \cong U_0\left(1-\frac{H}{H_c}\right)$$

The time of climb up to any altitude is equal to

$$t_{\min} \cong 0,\ 0384\ \frac{H_c}{H_0}\, lg_{10}\, \frac{H_c}{H_c - H}$$

Afterwards, by the aid of the formulæ (141) and (153) the four coefficients σ, r, q_0 and q_1 will be computed.

The propeller efficiencies η' and η'' are computed by the aid of the formulæ (146a) and (147a).

The propeller's efficiency for the best climb at ground level is equal to

$$\frac{N''}{N_0}\eta' = \frac{P V_c}{L_0}\left(y_c z_0 + \frac{U_0}{V_c}\right)$$

The propeller efficiency at the ceiling is equal to

$$\eta'' = \rho''\,\frac{P V_c}{L_0}$$

Finally, the whole performance chart can be traced to a first approximation. The speed curves of quadrant IV and transfer lines of quadrant II are only geometrical intermediaries and can be traced at once. The coefficients r and σ being known, the k_x/k_y curve of quadrant I can be plotted. The coefficients q_0 and q_1 being known, the specific thrust curves of quadrant III can be plotted, the whole strip of curves being replaced to a first approximation by a single curve. Further, by the aid of the transfer lines and speed curves, the Q/P curves of quadrant I can be traced. Afterwards, in extension of quadrant III, the $L_u/\delta A$ curve can be plotted. Finally, by three points (η_0, V_o), (η', V), (η'', V_c) the efficiency curve can be traced and thus the $L_m/\delta A$ curve deduced. In such a way, from the knowledge of V_o, U_o, H_c, L_o, and η_0 all the possible conclusions concerning the steady motion of the airplane considered will have been drawn.

In the second case, only drawings of the airplane considered are supposed to be available. It can be either an airplane in the process of design, or an airplane about which flying data are not available. The prediction of the complete performance is requested.

The values of the coefficients r and σ are first estimated by comparison with other similar airplanes. Two airplanes having the same wing-profile will have very closely the same values of r. The value of σ will be taken equal to

$$\sigma = s + 0,64\frac{a}{A}$$

where the equivalent area a of the parasite resistance has to be estimated and s taken from data concerning the wings used on the airplane considered.

Further, the power L_o and the airplane weight P must be known and one must decide upon the value of the efficiency η_o. As we have

$$\eta_o L_o = (k_x)_0\, \delta_0 A\, V_o^3 \text{ and } p_o = \frac{P}{\delta_0 A} = (k_y)_o V_o^2$$

we find:

$$\frac{\eta_o L_o}{P p_o^{1/2}} = \frac{(k_x)_o}{(k_y)_o^{3/2}} = (k_y)_o^{-1/2}\left[r\ (k_y)_o + \frac{\sigma}{(k_y)_o}\right] \tag{154}$$

This last relation will give us the value of $(k_y)_o$ and thus the value of the self-speed V_o that is compatible with the power available and drag offered by the airplane considered.

The easiest way to get a solution of equation (154) is to plot first the k_x/k_y curve as function of k_y—as we can do from the knowledge of the coefficients r and σ—and to plot afterwards the curve of $k_x/k_y^{3/2}$ as function of k_y, by dividing the ordinates of the k_x/k_y curve by the corresponding values of $\sqrt{k_y}$.

The smallest abscissæ of the $k_x/k_y^{3/2}$ curve corresponding to the ordinate equal to

$$\frac{\eta_o\, L_o}{P p_o^{1/2}}$$

will give us the value of $(k_y)_o$ to which corresponds the high horizontal speed V_o, which we will be able to read directly if only previously in our so-called quadrant IV, the speed curve $p_o = k_y\, V^2$ has been traced (see fig. 22).

Having found V_o by the aid of the $k_x/k_y^{3/2}$ curve, let us consider the relation

$$\eta' L'_m \cong \frac{N'}{N_o}\, \eta' L_o = P\ Vy_c + P\, U_o$$

from which we find

$$U_o = \frac{\frac{N'}{N_o}\, \eta' L_o - P\, Vy_c}{P} \tag{155}$$

As has been explained, there is advantage in taking $(k_y)_c \cong (k_y)_M$, that is $(k_y)_c$ equal to the k_y that corresponds to the minimum of the $k_x/k_y^{3/2}$ curve. Making this last selection, we have

$$V^2 = \frac{p_o}{(k_y)_M};\ y_c = y_M = \frac{(k_x)_M}{(k_y)_M}$$

and adopting a certain value for η' we can calculate U_o by the aid of (155).

When flying at the ceiling, we have

$$\eta''\,L_m'' \cong \frac{N''}{N_o}\,\eta''\,L_o\,\frac{p_o}{p_c} = (k_x)_c\,\delta_c\,A\,V_c^3 \text{ and } p_c = (k_y)_c\,V_c^2$$

and we find

$$\frac{\frac{N''}{N_o}\,\eta''\,L_o p_o}{P p_c^{3/2}} = \frac{(k_x)_c}{(k_y)_c^{3/2}} \tag{156}$$

PREDICTION OF THE FLYING SPEED

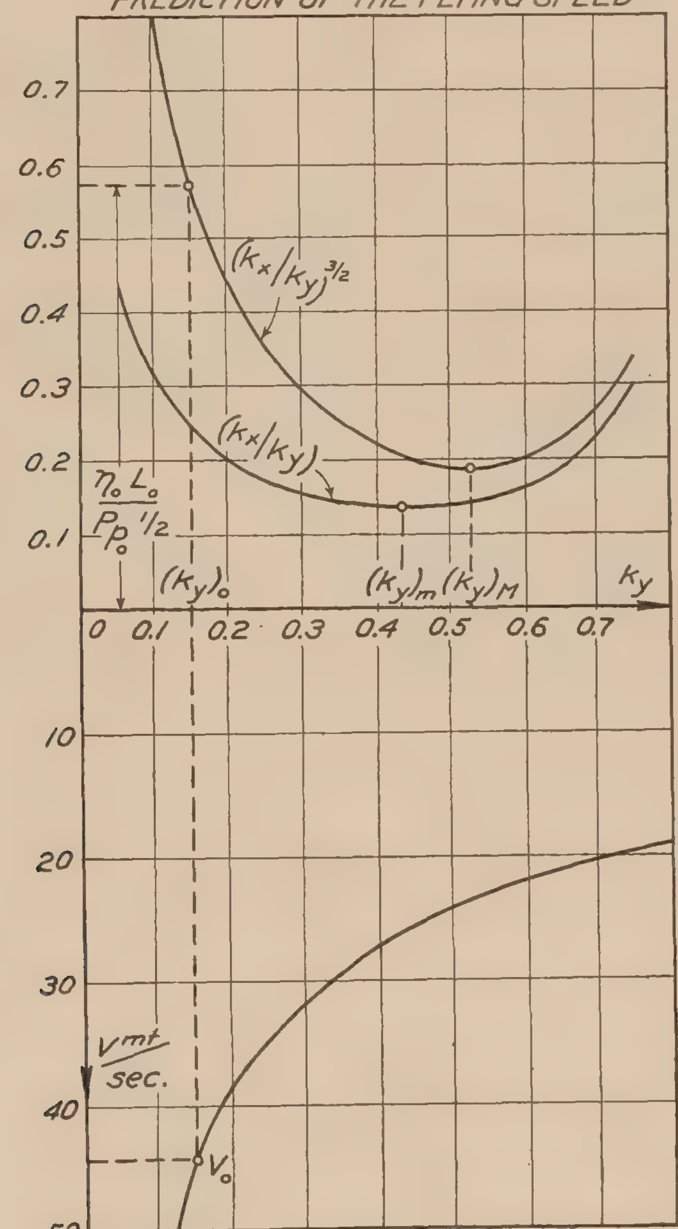


Fig. 22.

Assuming $(k_y)_c = (k_y)_M$ and deciding upon the value the efficiency η'' may reach, we get

$$\frac{p_o}{p_c} = \frac{P^{2/3}\,p_o^{1/3}}{\left(\frac{N''}{N_o}\right)^{2/3}\eta''^{2/3}\,L_o^{2/3}}\left[\frac{(k_x)_M}{(k_y)_M^{3/2}}\right]^{2/3} \tag{157}$$

which relation gives us the value of the ceiling. Knowing p_c, we find the ceiling self-speed, V_c, since

$$V_c^2 = \frac{p_c}{(k_y)_M} \tag{158}$$

It is in such a way that an estimation of the values of V_0, U_0 and p_c can be reached. The characteristic coefficient q_0 can now be easily found by the aid of formula (153), and the value of q_1 by the aid of the formula (141). Finally, we have to verify, by the aid of the formula

$$(k_y)_c=\sqrt{\frac{\sigma\times q_1}{r}}$$

how far the assumption $(k_y)_c=(k_y)_M$ holds.

The checking of the rate of climb U_0 and ceiling p_c by the aid of the last method gives good results because we have to deal with values of functions close to their minimum, where they do not vary much, the differences between y_M and y_c, and between $(k_x)_M/(k_y)_M^{3/2}$ and $(k_x)_c/(k_y)_c^{3/2}$ being in fact only very slight.

In all the preceding discussion, I had chiefly in view to point out the real nature of the problem of the performance prediction and to show by what concatenation of ideas we can be brought to its solution. Special attention must be paid to the rôle the $k_x/k_y^{3/2}$ curve plays in the finding of the self-speed V_0 from the knowledge of the power available $\eta_0\ L_0$, and the meaning of the minimum $(k_x)M/(k_y)M^{3/2}$ of the $k_x/k_y^{3/2}$ curve for the ceiling of the airplane considered.

The standpoint adopted in all this chapter was the prediction of the performance, starting with the knowledge of the smallest amount of data available concerning the airplane considered. But when for a given airplane, we know its k_x/k_y curve and possess all the data necessary in order to plot the specific thrust curve of the airplane's engine-propeller system; then the simplest way to predict the performance is just to draw, for the case considered, our performance chart, which will give the most complete performance prediction of the airplane considered. It is this question of finding from free flight tests these two fundamental curves—the k_x/k_y curve and the specific thrust curve—that we will consider in the next chapter.

REPORT No. 97.

GENERAL THEORY OF THE STEADY MOTION OF AN AIRPLANE.

By George de Bothezat.

PART VI.

FREE FLIGHT TESTING.

The performance chart we have developed in the foregoing gives a complete representation of the performance of an airplane in steady motion. A complete free flight test of an airplane must thus consist in getting all the data necessary in order to establish such a chart. The speed curves of quadrant IV and transfer lines of quadrant II being only geometrical intermediaries, it is only the curves of one of the quadrants I or III that we have to establish, because the curves of these quadrants mutually correspond to one another by the aid of quadrants II and IV. We shall show how to obtain from actual free flight tests the k_x/k_y curve and the Q/P curves of quadrant I.

Let us consider an airplane equipped with the following instruments: An air speed meter, a barograph, a strut thermometer. The barograph will be consisdered to be calibrated in pressure units, which is the only reasonable calibration of these instruments when used for free flight testing. In order to control, to a certain measure, the power of the engine, a tachometer must also be available. The test can be made either at full throttle or at any reduced throttle.

The airplane so equipped must make two or three climbs, at different indicated air speeds, but the last must be kept constant in each case all through the climb; also, on the way down, after each climb, it must make two or three glides. Each glide must also be made at different indicated air speeds, but constant for each glide. The glides will be done partly with the throttle completely closed and partly with the throttle so adjusted that

$$\frac{V}{NH}\cong 1$$

In these last glides the propeller thrust will be practically equal to zero.

The indicated air speed is proportional to the quantity

$$\frac{\delta V^2}{2}$$

But since $P\cong k_y \delta A V^2$, we have

$$k_y=\frac{2P}{A\frac{\delta V^2}{2}}$$

So that, if we keep $\delta V^2/2$ constant, that means that our glides or climbs take place at a constant value of the lift coefficient, and this is independent of our altitude.

Let us first consider the data furnished by the glides, made under the condition $V/NH=1$. The barograms obtained from those glides will give us the value of the pressure at each moment, and taking account of the corresponding temperatures, we can find the values of the densities δ for each moment of the glide and thus can deduce the actual self speeds V from the knowledge of the indicated air speeds and the calibration curve of the speed meter used. Further, the rates of descent can be deduced from the glide-barograms, which, as has been shown in Chapter III, are equal to

$$U=-\sigma\frac{dp}{dt} \tag{51}$$

where $\sigma=\delta g$ is the corresponding specific weight of the air and dp/dt is the angular coefficient of the tangent to the glide barogram curve at the point considered. Knowing V and δ, or the specific load $p=P/\delta A$, we find from the relation $k_y=P\ Cos\ \gamma/V^2$, the corresponding value of k_y. Since, for the glides under the condition $V/NH=1$, we have $Q=O$ and thus $-\gamma=k_x/k_y=-U/V$, the point in quadrant I with the coordinates k_y and $-U/V$ will be a point of the k_x/k_y curve (see figs. 18 and 19). Proceeding in the same way for glides made at different indicated air speeds, we find a set of points of the k_x/k_y curve. The author has convinced himself by the actual use of the above described method that it is easy to get points of the k_x/k_y curve for values of $k_y>(k_y)_m$ by making glides at sufficiently low self-speeds. These glides have only to be made at heights sufficient for the safety of the pilot.

When proceeding, as above described, with the glides made with the throttle completely closed, we get a certain k_x'/k'_y curve. The difference of the ordinates of this last curve and the k_x/k_y curve will give us the $-Q/P$ curve, which transferred in quadrant III by the aid of the speed curves and transfer line will give us the $-Q/\delta A$ curves, by the aid of which we can estimate the mechanical losses of the motor, as has been already shown in the foregoing. (See fig. 19.)

If now we proceed in a similar manner with the climb barogram, and recorded indicated air-speeds; that is, deducing from them the corresponding δ, V and γ we obtain from each barogram a set of values of γ corresponding to a constant value of k_y, for different values of the specific load $p=P/\delta A$ for which we can adopt a set of standard values. If now we plot these values of γ in quadrant I, starting from the k_x/k_y curve and join all the points that correspond to equal values of the specific load, we get the family of the Q/P curves, with γ as parameter, since $Q/P=k_x/k_y+\gamma$ in each climb. These Q/P curves, transferred in quadrant III by the aid of the speed curves and transfer lines will give us the set of specific thrust curves.

By tracing the rate of climb curves in quadrant IV the ceiling will be checked, and, as described in the foregoing, the whole airplane performance can be deduced with ease from the knowledge of the k_x/k_y curve and the specific thrust curves.

If when making the last tests the airplane were equipped with a torque meter, then by recording the torque and the revolutions we would know the power delivered at each moment by the engine and we then could trace in extension of the quadrant III, of our chart, the $L_m/\delta A$ curve. As the $L_u/\delta A=QV/\delta A$ curve can be directly deduced from the $Q/\delta A$ curve of quadrant III, the knowledge of the $L_m/\delta A$ curves will allow us to immediately deduce the efficiency curves. It is in such a way that from free flight tests the propulsive efficiencies can be deduced. It is easy to deduce from the efficiency curves and the $L_m/\delta A$ curves, by the aid of the revolution curves as function of the self-speed V, the efficiency curve as well as the $L_a/\delta N^3$ curve as function of V/N, and thus to get from the free flight test the complete characteristics of the propeller. It is also from the $L_m/\delta A$ curve that the engine power characteristics as function of N for different values of the density δ and throttle x can be deduced.

One can now realize how important it is to use a torque meter in free flight tests. A torque meter, giving us a continuous control of the power, will make the test perfectly reliable in the sense of knowledge of the power really developed by the engine; and besides, the torque meter will allow us to obtain, in addition to the complete characteristics of the airplane, the complete and separate characteristics of the propeller and of the engine.

The chart (fig. 26) annexed at the end of this paper gives the characteristics of a Vought airplane as actually obtained from free flight tests by the above described method. For all the details concerning such test the reader is referred to the McCook Field (Dayton, Ohio) Report No. 1242, "*A report showing the use of the de Bothezat performance chart for expressing the performance of the VE-7 airplane P-113, from data obtained in actual flights,*" by Mr. C. V. Johnson and W. F. Gerhardt.

I wish finally to call attention to one more important question connected with free-flight testing. The power of the engine is affected by the air temperature, and it is thus necessary to reduce the power of the engine, and thus the whole performance, to some standard temperature, if we wish to get results that can be compared with other tests. For reasons that have

been discussed in Chapter IV, it is the isothermic atmosphere of zero degrees centigrade that we adopt as standard, and thus the whole performance has to be reduced to zero degrees centigrade. It is evident that one has to take account of the temperature to find the value of the densities from the pressures given by the barograph, but how must we take into account the influence on the performance of the power variation due to the temperature?

At a constant density, the engine power depends upon temperature. That is, at the same density but at the temperature of zero degrees centigrade the engine would give a slightly different power from that in the actual flight. Let ΔL_m be this positive or negative increment of the power due to temperature difference at constant density. On account of the fact that the drag and lift of the airplane depend only upon density, neither k_x nor k_y nor V—because $P=k_y\delta A V^2$—will be affected by the temperature; that is, neither the k_x/k_y curve, nor the speed curves. The increment of power ΔL_m will act on the performance as a slight change of throttle and it is only the values of Q/P or $Q/\delta A$ that will have to be corrected. As δ and V remain the same, the efficiency η will remain the same, and the variation ΔL_u of the power available will be proportional to the variation of the power delivered, but as $L_u = VQ$ and V remain the same, we have

$$\Delta L_u = \eta \Delta L_m = V \Delta Q$$

The correction to be applied to the thrust thus simply turns out to be equal to

$$\Delta Q = \frac{\eta \Delta L_m}{V}$$

and the correction to be applied to the Q/P values turns out to be equal to

$$\frac{\Delta Q}{P} = \frac{\eta \Delta L_m}{PV}$$

The power correction ΔL_m due to temperature at *constant density* has to be determined by special tests of the engine.

Those who have followed carefully the methods and questions of principles discussed in this paper will not meet the slightest trouble in making the most complete and rigorous airplane free-flight tests.

REPORT No. 97.

GENERAL THEORY OF THE STEADY MOTION OF AN AIRPLANE.

By George de Bothezat.

PART VII.

SHORT DISCUSSION OF THE PROBLEM OF SOARING.

We have until now, paid exclusive attention to the airplane self-speed $\bar{V}$. This means that we have considered the airplane flight from a system of coordinates that had, relatively to the ground, a speed constant in magnitude and direction—and equal to the wind speed $\bar{v}$. Let us now follow the airplane flight from a system of coordinates invariably connected to the ground. As we have already mentioned, at the beginning of Chapter I, the ground or absolute speed $\bar{W}$ of the airplane is at each moment equal to

$$\bar{W}=\bar{V}+\bar{v} \tag{159}$$

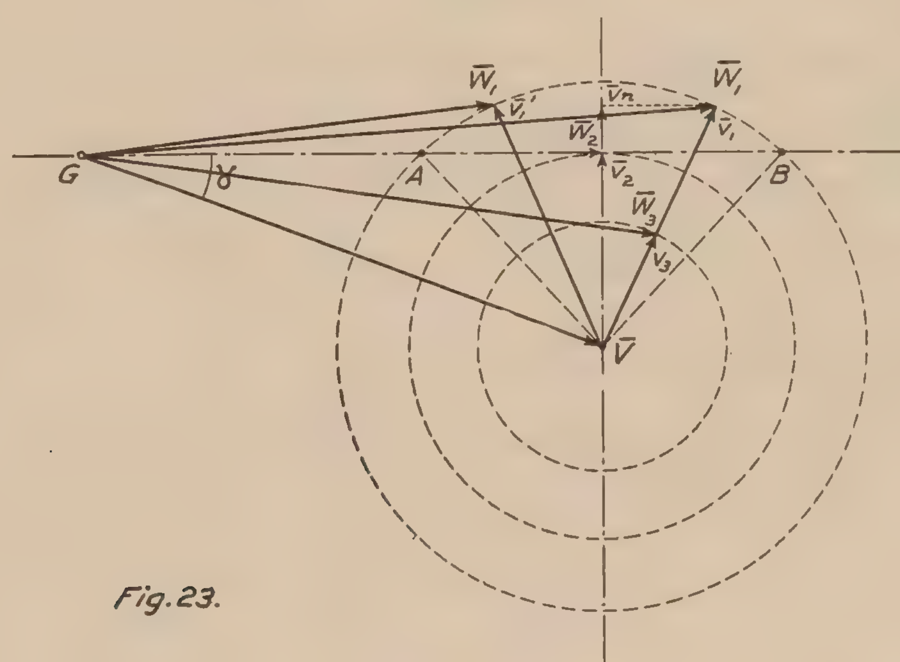


Fig. 23.

INFLUENCE OF WIND ON THE AIRPLANE GLIDING TRAJECTORIES

Let us consider a gliding airplane and for the sake of simplicity neglect the negative propeller thrust. Let us draw from the center of mass G of the airplane a vector $\bar{V}$ equal to its gliding self-speed at the moment considered and making the angle γ (the actual airplane path inclination) with the horizontal. (See fig. 23.) If the wind speed $\bar{v}$ at the moment considered and the point of the atmosphere where the airplane is actually gliding is equal to zero, then

$$\bar{W}=\bar{V} \tag{160}$$

But, since for gliding the angle γ turns out to be always negative (see equation (61), Chap. IV), the absolute speed $\bar{W}$ can under those conditions only be a descending speed. We are thus brought to the general conclusion:

In those parts of the atmosphere where there is no wind a glider can only be descending.

Let us now consider the airplane gliding in a wind having a magnitude equal to v. Let us draw from the end of the vector $\bar{V}$ (see fig. 23) a circumference having a radius equal to v. Three cases can be encountered.

In the first case the described circumference cuts the horizontal in two points A and B. In this case when the wind of magnitude v has a direction included in the angle $\widehat{AVB}$, the absolute speed $\overline{W}$ of the airplane will be either horizontal or ascending. For any other wind direction the absolute speed $\overline{W}$ will be descending.

In the second case the described circumference is tangent to the horizontal. In this case only for the wind blowing directly upwards can the absolute speed $\overline{W}$ be horizontal.

In the third case the described circumference is disposed entirely below the horizontal. In this case the absolute speed $\overline{W}$ will always be descending independent of the direction of the wind.

We are thus brought to the following fundamental conclusion:

The absolute speed $\overline{W}$ of any glider in a state of steady motion, be it an airplane or a bird, can be ascending or horizontal only in ascending wind, and provided the vertical component v_n of the last is larger than the rate of descent U.

The so-called phenomenon of soaring is thus only possible in an ascending wind, for which

$$v_n > U \tag{161}$$

The smaller the rate of descent U the smaller may be the vertical wind component v_n necessary for soaring.

Let us discuss briefly those conditions that make the rate of descent a minimum.

One can see from equation (95) of Chapter IV that for $Q=0$ the rate of descent is equal to

$$U = -p^{1/2}(rk_y^{1/2} + \sigma k_y^{-3/2}) = -p^{1/2}\frac{k_x}{k_y^{3/2}} \tag{162}$$

and has a minimum given by the condition

$$\frac{\partial U}{\partial k_y} = -p^{1/2}\frac{\partial}{\partial k_y}\left(\frac{k_x}{k_y^{3/2}}\right) = -1/2p^{1/2}(rk_y^{-1/2} - 3\sigma k_y^{-5/2}) = 0$$

which gives

$$k_y = (k_y)_M = \sqrt{\frac{3\sigma}{r}} \tag{163}$$

and

$$U_{\min} = -p^{1/2}\left[r\left(\frac{3\sigma}{r}\right)^{1/4} + \frac{\sigma}{\left(\frac{3\sigma}{r}\right)^{3/4}}\right] \tag{164}$$

The rate of descent U is thus a minimum for the same value $(k_y)_M$ of the lift coefficient for which the $k_x/k_y^{3/2}$ curve has a minimum.

We shall introduce the notation

$$S = \left[r\left(\frac{3\sigma}{r}\right)^{1/4} + \frac{\sigma}{\left(\frac{3\sigma}{r}\right)^{3/4}}\right] = 4/3r\left(\frac{3\sigma}{r}\right)^{1/4} \tag{165}$$

and call it the *soaring constant* of a glider, because it depends only upon the aerodynamical properties of the glider consider. We can thus write

$$U_M = S\sqrt{\frac{P}{\delta A}} \tag{166}$$

where U_M represents the magnitude of the rate of descent at its minimum.

The rate of descent U_M will be the smaller, the smaller P/A is, that is, the wind loading of the considered glider, and the smaller the soaring constant is.

As has already been remarked, the coefficient r depends chiefly upon the wing profile and its value is included in narrow limits. But the ratio σ/r depends upon the value of the ratio a/A and can be greatly reduced by reducing the drag of the parasite resistance and increasing the wing area. In figure 24 has been represented the curve of

$$S'=\frac{S}{r}=4/3\left(\frac{3\sigma}{r}\right)^{1/4} \tag{167}$$

as function of the ratio σ/r, which allows a quick checking of the value that the soaring constant

$$S=r\,S' \tag{168}$$

can have.

Taking for average values $\sigma/r=0,2$ and $r\simeq0,1$ from figure 24, we find $S'\simeq1,2$ and $S\simeq0,12$, and if we take $P/A\simeq8$ *klg/mt²* this would make, with $\delta=1/8$, the rate of descent equal to

$$U_M=0,12\sqrt{6\,4}\simeq0,96\ mt/sec$$

It is thus quite possible to realize gliders with a rate of descent less than 1 meter per second, especially on account of the fact that the ratio σ/r can be made still less than the average value we have adopted. A rising wind with a vertical component equal to 1 meter per second would thus be sufficient to secure the soaring of such glider.

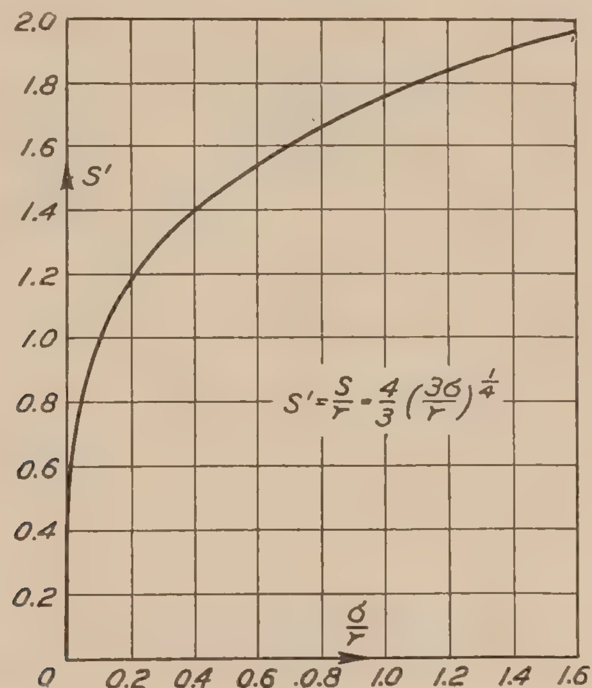


Fig. 24.

The gliding of such glider would take place at a value of

$$k_y=(k_y)_M=\sqrt{\frac{3\sigma}{r}}\simeq0,77.$$

and its self-speed would be equal to

$$V=\sqrt{\frac{P}{k_y\,\delta\,A}}=\sqrt{\frac{64}{0,77}}\simeq9,2\ mt/sec$$

This would be the low speed of the glider; its high speed could be made around 20 *mt/sec* which was the speed of the early airplanes. High cambered aerofoils can give lift values up to $(k_y)\simeq0,8$.

We are thus brought to the conclusion that it is quite possible to build gliders having a very low rate of descent. Such gliders must have a high cambered aerofoil, a low self-speed, a small drag, and a small loading per unit of area. Special attention must only be paid to secure the complete stability and maneuverability of the glider at its lowest speed, by the aid of sufficient stabilizing surfaces and rudders. Such a glider, having a low rate of descent, will soar in any ascending

wind whose vertical component is equal to or greater than the minimum rate of descent of the glider.

The fact that the soaring of birds is very often observed in some regions shows that in those regions ascending winds, whose vertical component has a sufficient value to secure soaring, must be a common phenomenon.

It is the opinion of the author that the main reason for the frequent occurrence of ascending winds is the following:

As is known, winds are generally variable with altitude, that means the different layers of the atmosphere have different velocities. It even sometimes happens that two air layers have opposite speeds; as a result of this speed difference, a vortex sheet must be formed between them.[1] But such vortex sheets being unstable, as is known, they must break into a system of vortex rows.

v. Karman[2] has shown that among all possible vortex rows, it is the system of quincunx vortex rows that constitute a stable configuration and the unstable vortex sheets seem most generally to break into such quincunx vortex rows. In figure 25 a system of such quincunx vortex rows is diagrammatically represented.

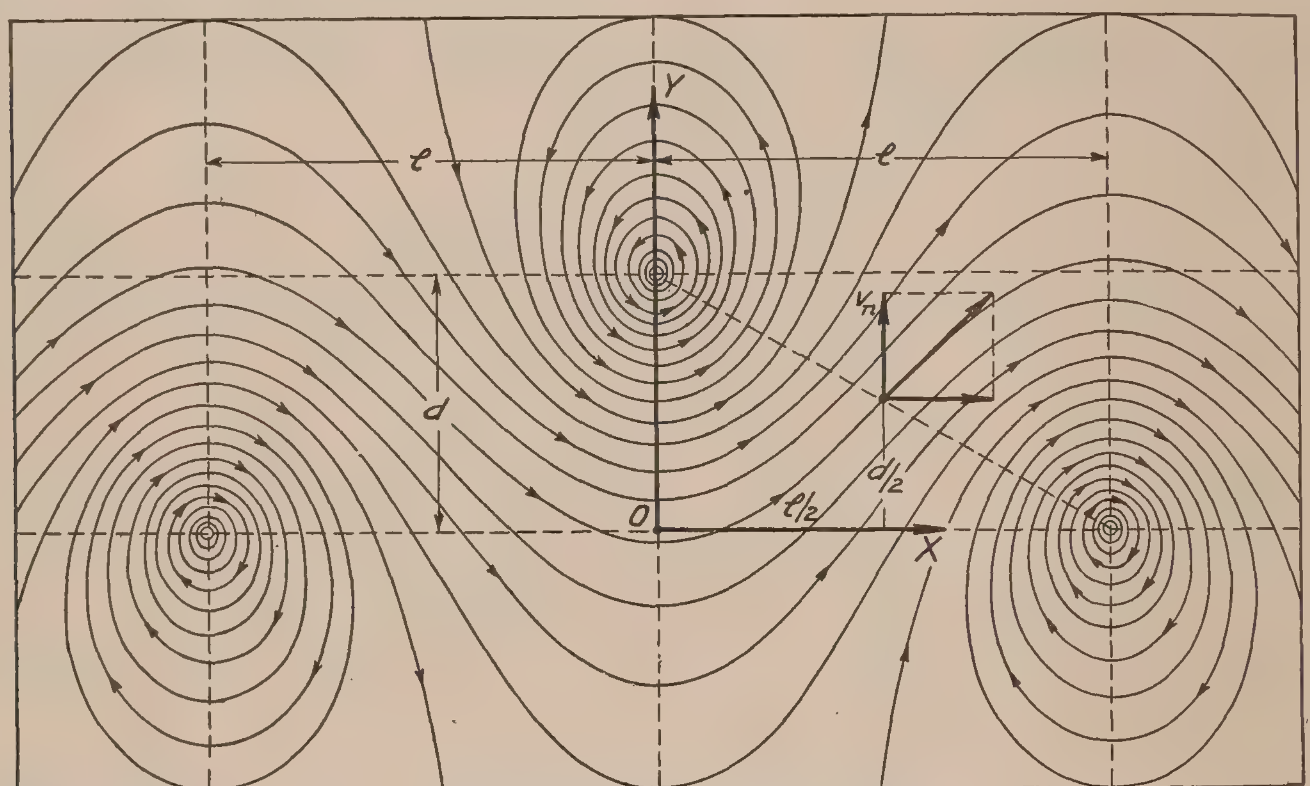


Fig. 25. THE STRUCTURE OF THE WIND
BUILDING OF QUINCUNX VORTEX ROWS
BETWEEN AIR LAYERS OF DIFFERENT VELOCITY

We are thus naturally brought to the conclusion that as a consequence of the speed difference between air-layers, a formation of vortex rows in quincunx must take place between such layers. The ordinary atmospheric wind thus appears to us in its structure to be made up of wind layers separated by quincunx vortex rows traveling between the air layers.

It seems also that the unequal heating of the ground by the sun rays acts greatly in favor of the formation of such traveling qiuncunx vortex rows.

Once the formation of such atmospheric-quincunx vortex rows is admitted, it is easy to conceive that we must meet in the atmosphere in some places ascending currents, in other places descending currents. It is the ascending waves of the atmospheric quincunx vortex rows that makes soaring possible, and it is in these waves that birds soar when they meet them. It is by remaining in the boundaries of such ascending wave, or by gliding from one ascending column into another, that birds can maintain soaring.

It is of interest to check the mean value that the vertical wind component in the ascending column produced by a system of quincunx vortex rows can have. Let us consider such a

[1] See the author's "Introduction to the Study of the Laws of Air Resistance of Aerofoils," Chapter III, Report No. 28 of the National Advisory Committee for Aeronautics, Washington, D. C.

[2] See the above-mentioned Report No. 28, p. 46, note IV.

quincunx vortex row and adopt as mean value of the vertical wind component the vertical wind value at the middle of the line joining two consecutive vortices taken one in each row (see fig. 25). If we refer one of the rows to a system of $X\ O\ Y$ axes, the origin being in the middle between two of the vortices, the X axis directed along the vortex row and the Y axis perpendicular to the last, if $2l$ is the distance between two consecutive vortices in one row and d the distance between the two rows, then the point adopted as the one having the mean vertical wind component will have the coordinates $(l/2,\ d/2)$, and the vertical wind component produced by one row will be found equal to:[3]

$$v=\frac{\pm I}{4l}\ \frac{tg\frac{\pi}{4}\left(2tgh^2\frac{\pi d}{4l}-1\right)}{1+4tg^2\frac{\pi}{4}\ tgh^2\ \frac{\pi d}{4l}} \tag{169}$$

Where I is the intensity of each of the vortices of the row. The vertical wind component produced by both rows and which we will designate by v_n will have a double value and this will be equal to

$$v_n=\frac{\pm I}{2l}\ \frac{tg\frac{\pi}{4}\left(2tgh^2\frac{\pi d}{4l}-1\right)}{1+4tg^2\frac{\pi}{4}\ tgh^2\frac{\pi d}{4l}} \tag{170}$$

But according to Karman for the stable quincunx vortex row system the ratio $d/2l$ has the value

$$\frac{d}{2l}=0.283 \tag{171}$$

Substituting this last value in (170) we get

$$|v_n|=0.24\frac{I}{2l} \tag{172}$$

Such would be the mean value of the vertical wind component produced by a system of atmospheric quincunx vortex rows. By the aid of this last relation, when two of the three quantities v_n, $2l$ or I are known, the third one can be estimated. If for example we know the values of v_n and $2l$, we can find the value of

$$I=\frac{2l}{0.24}v_n$$

of the intensity of the vortices of the rows; in other words, the value of the circulation around each vortex.

When a vortex sheet breaks into vortices, the intensity of each vortex is very closely equal to the speed differences in the two layers between which the vortex sheet was formed, multiplied by the distance between the vortices. When the vortex sheet between two atmospheric layers breaks into a quincunx vortex row, we evidently first have the formation of one row, with vortices at a distance l, but this soon goes over into the stable quincunx vortex system by the upward or downward displacement of one-half of the vortices of the row, with a distance between vortices in each row equal to $2l$. If we thus call w the original speed difference in the two vortex layers which have given rise to a system of quincunx vortex rows, the intensity I of each of such vortices will thus be equal to

$$I\cong lw$$

and substituting in (172) we find

$$v_n=0.24\frac{lw}{2l}=0.12w$$

[3] See the author's "Introduction to the Study of Laws of Air Resistance of Aerofoils," p. 51.

Another case can also happen. We can imagine the quincunx vortex rows formed by an air current appearing in an air mass, having the same velocity in the whole current. The quincunx vortex rows will then be formed from two vortex sheets. The vortices formed from each sheet will remain in the same level and we will have

$$I = 2lw$$

where w is the speed difference between the air current and the rest of the surrounding atmosphere. In such a case, we have

$$v_n = 0.24w$$

We are thus brought to the remarkable conclusion that *the mean vertical wind component produced by a system of quincunx vortex rows, resulting from the breaking of the vortex sheets between atmospheric layers, can have values from one-eighth to one-quarter of the speed difference between the atmospheric layers that have originated these quincunx vortex rows.*

As speed differences of a few meters per second are easy to conceive between atmospheric layers, ascending wind currents of somewhat smaller values must be a frequent phenomenon, as the soaring of birds undoubtedly prove.

As a general conclusion of this discussion, one can see that *the realization of gliders able to soar in average atmospheric conditions must be considered as perfectly possible and as presenting the greatest interest.*

Such a glider must be conceived, as has already been explained, as an airplane well streamlined, with high cambered wings, small wing loading and small speed and thus small power. By the aid of its engine the airplane will reach that altitude where the formation of the system of quincunx vortex rows has taken place, and once in the ascending current will soar in it and by continuously turning around will remain in it. In an airplane specially built for soaring, the pilot will very easily feel the ascending current by the upward acceleration that it will communicate to the airplane. Even in actual high-speed airplanes, pilots have a very clear feeling of the upward and downward currents. When strong enough, the pilots describe them as the so-called "bumps," and "air holes." The bumps are, exactly speaking, strong ascending currents and the holes strong descending ones. But I mention once more that the building of such soaring airplanes will be met by complete failure, if the conditions of their maneuverability and stability are not considered with sufficient attention.

REPORT No. 97.

NOTE 1.

THE APPROXIMATE EQUATIONS OF THE CHARACTERISTICS OF THE ENGINE-PROPELLER SYSTEM.

In the foregoing has been given the method of deducing the characteristics of the engine-propeller system from the empirical curves of the engine and propeller characteristics. It is desirable in several instances to possess also approximate equations of the characteristics of the engine-propeller system because they allow a better survey, even if only to a first approximation, of the relations that held between the quantities involved in the question. The approximate equations of the different characteristics of the engine-propeller system can be easily deduced when approximate equations for the characteristics of the propeller and engine have been properly selected.

For the propeller, as a very good approximation of the characteristics, for the range of the flying interval, the following equations can be adopted:

For the thrust

$$Q = h_0 \delta N^2 H^2 D^2 \, (1 - x) \tag{173}$$

For the power

$$L_a = h_0^1 \delta N^3 H^3 D^2 \, (1 - h^2 x^2) \tag{174}$$

where h_0, h_0^1 and h^2 are three constants that characterize the propeller considered; N the number of revolutions per second; H the *zero thrust* pitch; D the propeller diameter; $x = V/NH$ the *relative pitch*[1]).

We will call h_0 the *thrust coefficient*, h_0^1 the *power coefficient*, and h^2 the *pitch coefficient*, the last named being selected for reasons that will appear later.

The *zero thrust pitch* H considered above is defined by the condition that $V/NH = 1$ for $Q = 0$, that is H is taken equal to the advance V/N, for which the thrust Q disappears. The value of H has to be deduced from the $Q/\delta\, N^2$ curve plotted against V/N, which curve intersects the V/N axis at the point $V/N = H$.

The coefficients h_0, h_0^1, and h^2 must be deduced from the empirical curves of

$$\frac{Q}{\delta N^2 H^2 D^2} = h_0 \, (1 - x) \tag{175}$$

and

$$\frac{L_a}{\delta N^3 H^3 D^2} = h_0^1 \, (1 - h^2 x^2) \tag{176}$$

plotted against $x = V/NH$ by the method of least squares.

---

[1] In my general theory of blade screws I have established the following formulæ (see relation (114) p. 48):

$$\Delta Q = 2\delta \Delta \delta \Omega^2 \frac{\delta v^2}{az} r^2 tg^2 \, (\phi + \beta)$$

where ΔQ is the partial thrust. δ the air density. ΔS the annulus to which corresponds the thrust ΔQ. Ω the angular velocity of the propeller rotation$= 2\pi N$. $\rho_v{}^2/az$ a dimensionless quantity function of $x = V/NH$ only. $r^2 tg^2\,(\phi + \beta)$ a quantity nearly equal to the pitch of the blade section considered.

Integrating the above relation it will be easy to see that the result must be of the form

$$Q = h_0 \delta N^2 H^2 D^2 f(x).$$

On the other hand the function $f\,(x)$ turns out to be, in general, very closely a linear function of x, with $f\,(x) = 0$ for $x = 1$. We thus can write:

$$Q \cong h_0 \delta N^2 H^2 D^2 \, (1 - x).$$

The formula (173) is thus justified. It is from similar considerations that the formula (174) also follows.

It will be easy to convince oneself that the equations (175) and (176) will generally be able to represent the experimental curves with good accuracy.

Making use of the equations (173) and (174) we find for the efficiency η of the propeller the expression (177)

$$\eta=\frac{Q\,V}{L_a}=\frac{h_0 x\ (1-x)}{h_0^1\ (1-h^2x^2)}. \tag{177}$$

The efficiency η is equal to zero for

$$x=0,\ \text{and}\ x=1$$

and reaches its maximum η_m for

$$\frac{\partial\eta}{\partial x}=\frac{h_0\ (1-2x)\ (1-h^2x^2)+2h_0h^2x^2\ (1-x)}{h_0^1\ (1-h^2x^2)}=0$$

that is

$$h^2x^2-2x+1=0$$

which relation gives

$$h^2=\frac{2x_m-1}{x_m^2} \tag{178}$$

and

$$x_m=\frac{1-\sqrt{1-h^2}}{h^2} \tag{179}$$

where x_m is the value of x that corresponds to $\eta=\eta_m$.

It is the last relation (179) that has to be used for finding the value of x_m, the coefficient h^2 being found by the method of least squares from the empirical curves (175) and (176) of the thrust and power. One must avoid checking the value of x_m from the curve of the efficiency η, because it is always difficult to find accurately from an empirical curve the value of the abscissa that corresponds to the maximum of the relation (178) which shows that h^2 is a function of x_m alone. That is why I have called h^2 the pitch coefficient.

The maximum of $\eta=\eta_m$ is thus equal to

$$\eta_m=\frac{h_o x_m}{2\ h'_o} \tag{180}$$

and we also have

$$\frac{h_o}{h'_o}=\frac{2\ \eta_m}{x_m} \tag{181}$$

and finally

$$\eta=\frac{2\ \eta_m\ x\ (1-x)}{x_m\ (1-h^2x^2)} \tag{182}$$

Let us find the expressions of the thrust and power coefficients h_o and h'_o as function of the power absorbed by the propeller. We have

$$L_a=h'_o\delta N^3H^3D^2\ (1-h^2x^2)$$

For the propeller working at its maximum efficiency we will have

$$(L_a)_m=h'_o\delta N_m^3H^3D^2\ (1-h^2x^2_m) \tag{183}$$

where $(L_a)_m$ and N_m are the power absorbed by the propeller and its number of revolutions for $\eta=\eta_m$. It must be remembered that in general, $(L_a)_m$ as well as N_m, are functions of the translational speed of the propeller because, when $\eta=\eta_m$ we have $x=x_m$; that is, $V_m/N_m=Hx_m$ which relation fixes only the ratio of the translational speed V_m to the number of revolutions N_m corresponding to the condition of the maximum efficiency η_m. But when the propeller con-

sidered is connected to a given engine, then the characteristics of the engine-propeller system as it has been already shown in this report, are functions—for each value of the density and throttle opening—of the translational speed V of the engine-propeller system alone. In such case there will be only a single value $N=N_m$ of the revolutions and $L_a=(L_a)_m$ of the power absorbed at which—for a given density and throttle opening—we will have $\eta=\eta_m$ if only the maximum efficiency can be reached in the given working conditions of the engine-propeller system.

From (183) we find

$$h'_o=\frac{(L_a)_m}{2\ (1-x_m)\ \delta N^3_m H^3 D^2} \tag{184}$$

and on account of (181)

$$h_o=\frac{\eta_m (L_a)_m}{x_m(1-x_m)\ \delta N^3_m H^3 D^2} \tag{185}$$

We also have

$$L_a \cong (L_a)_m \frac{N^3}{N^3_m} \frac{(1-h^2x^2}{2\ (1-x_m)} \tag{186}$$

For the engine power we will adopt as a first approximation,

$$L_m \cong m\ N\ \delta \tag{187}$$

considering that the engine is used in the interval at which its power is still proportional to the revolutions and giving one the liberty of making when necessary a correction for the deviation of the power from its proportionality to the density.

When a given propeller is connected to a given engine for each state of steady conditions the power absorbed by the propeller must be equal to the power delivered by the engine; that is, we must have

$$L_a=L_m$$

or

$$\frac{L_a}{\delta N^3}=h'_o H^3 D^2 (1-h^2 X^2)=\frac{L_m}{\delta N^3}=\frac{m}{N^2}$$

From the last relations we find the law of variation of the revolutions N as function of the speed V for the engine-propeller system under consideration. We thus find—

$$N^2\left(1-h^2\frac{V^2}{N^2H^2}\right)=\frac{m}{h'_o H^3 D^2} \tag{188}$$

or

$$N^2=\frac{m}{h'_o H^3 D^2}+\frac{h^2 V^2}{H^2}=\frac{h^2 V^2}{H^2}\left(1+\frac{m}{h^2 h'_o H D^2 V^2}\right)$$

or finally putting

$$c=\frac{m}{h^2 h'_o H D^2} \tag{189}$$

we get

$$N^2=\frac{h^2 V^2}{H^2}\left(1+\frac{c}{V^2}\right) \tag{190}$$

and

$$N=\frac{hV}{H}\sqrt{1+\frac{c}{V^2}} \tag{191}$$

we also have

$$x^2=\frac{V^2}{N^2H^2}=\frac{1}{h^2\left(1+\frac{c}{V^2}\right)};\quad x=\frac{1}{h\sqrt{1+\frac{c}{V^2}}} \tag{192}$$

The expression (191) is the equation of the strip of the revolution curves of figure 12. When no allowance is made for the deviation of the engine power from its proportionality to the density the whole strip of curves is replaced by one mean curve.

Let us now find the equation of the specific thrust curve.

We have

$$Q = h_0 \delta N^2 H^2 D^2 \; (1-x)$$

or

$$Q = h_0 \delta D^2 \frac{V^2}{x^2} \; (1-x)$$

Substituting for x its value (192) we find,

$$Q = h_0 \delta D^2 h^2 V^2 \left(1 + \frac{C}{V^2}\right) \left(1 - \frac{1}{h\sqrt{1+\frac{c}{V^2}}}\right)$$

or

$$Q = h_0 \delta D^2 V^2 \left[h^2\left(1+\frac{c}{V^2}\right) - h\sqrt{1+\frac{c}{V^2}}\right] \tag{193}$$

The last relation gives us the law of variation of the thrust Q of the motor-propeller set in function of the speed V.

On account of the fact that for the flying interval the quantity $\frac{c}{V^2}$ varies between comparatively narrow limits, we can develop the radical $\sqrt{1+\frac{c}{V^2}}$ in serie neglecting the terms of higher order and thus simplify the relation (193). We can thus take

$$\sqrt{1+\frac{c}{V^2}} \cong \alpha + \beta \frac{c}{V^2}$$

where α and β are two constants. On account of (187); (178) and (184) the value of (189) of c can be written

$$c = \frac{2(1-x_m)}{2x_m - 1} V^2_m$$

and thus

$$\frac{c}{V^2} = \frac{2(1-x_m)}{2x_m - 1} \frac{V^2_m}{V^2}$$

For most propellers x_m is included between 0.7 and 0.8, and in the flying range the ratio $\frac{V_m}{V}$ can hardly come out of the limits

$$1/2 < \frac{V_m}{V} < 2$$

Consequently the ratio c/V^2 will be usually included between the limits

$$1/2 < \frac{c}{V^2} < 8$$

For the last interval of variation of c/V^2 one can take with a good approximation $\alpha = 1.3$; $\beta = 0.21$ and thus

$$\sqrt{1+\frac{c}{V^2}} \cong 1.3 + 0.21 \frac{c}{V^2}$$

Making use of the approximate expression of the radical $\sqrt{1+\frac{c}{V^2}}$ the relation (193) can be written

$$Q \cong h_0 \delta D^2 V^2 \left[h^2 \left(1+\frac{C}{V^2}\right) - h\left(\alpha+\beta\frac{c}{V^2}\right)\right]$$

or

$$Q \cong h_0 \delta D^2 [ch(h-\beta) - h\ (\alpha - h) V^2] \tag{194}$$

We thus find for the specific thrust the expression

$$q = \frac{Q}{\delta A} = \frac{h_0 D^2}{A} [ch(h-\beta) - h\ (\alpha-h) V^2] \tag{195}$$

The approximation, adopted in this report for the specific thrust as being of the form

$$q = q_0 - q_1 V^2 \tag{196}$$

is fully justified, and we find

$$q_0 = ch_0 D^2 \frac{h(h-\beta)}{A} \tag{197}$$

$$q_1 = h_0 D^2 \frac{h(\alpha - h)}{A} \tag{198}$$

since the relations (175) and (176) constitute a good approximation for the propeller thrust and power characteristics, the possible deviation of the specific thrust curve—for a given density and throttle opening—from the law (196) must be chiefly due to the deviation of the engine power from its proportionality to the revolutions.

Substituting in the last expressions of q_0 and q_1 for the constants C, h_0 and h their values (189), (185) and (178) we find

$$q_0 = \frac{C_0}{\delta A} \frac{\eta_m (L_a)_m}{V_m}; \quad q_1 = \frac{C_1}{\delta A} \frac{\eta_m (L_a)_m}{V_m^3} \tag{199}$$

with

$$C_0 = \frac{2\sqrt{2x_m - 1} - \beta x_m}{\sqrt{2x_m - 1}}; \quad C_1 = \frac{(\alpha x_m - \sqrt{2x_m - 1})\ \sqrt{2x_m - 1}}{(1-x_m)} \tag{200}$$

In order to easily check c_0 and c_1 curves of these coefficients as functions of x_m can be traced.

We thus find for the specific thrust curve the general equation

$$q = \frac{Q}{\delta A} = q_0 - q_1 V^2 = \frac{\eta_m (L_a)_m}{\delta A V_m} \left(c_0 - c_1 \frac{V^2}{V_m^2} \right) \tag{201}$$

The thrust curve of the engine-propeller, represented in figure 12, for the approximation of a single strip, has for equation

$$\frac{Q}{\delta} = \frac{Q_m}{\delta} \left(c_0 - c_1 \frac{V^2}{V_m^2} \right) \tag{202}$$

with

$$Q_m = \frac{\eta_m (L_a)_m}{V_m} \tag{203}$$

Let us further find the equation of the efficiency curve of the engine-propeller set. We have:

$$\eta=\frac{QV}{L_a}=\frac{h_0 x\ (1-x)}{h_0^1\ (1-h^2x^2)}$$

Substituting for x its value (192) we get

$$\eta=\frac{h_0 V^2}{h^1_0 c}\left[\frac{1}{h}\sqrt{1+\frac{c}{V^2}}-\frac{1}{h^2}\right]$$

and on account of $\frac{c}{V^2}$ being small we find

$$\eta=\frac{h_0}{h^1_0}\left[\frac{\beta}{h}-\frac{1-\alpha h}{ch^2}V^2\right] \tag{204}$$

Substituting for h_0, h_0^1, h and c their values (185), (184), (178) and (189) we find

$$\eta=\frac{\eta_m}{\sqrt{2x_m-1}}\left[2\beta-\frac{(x_m-\alpha\sqrt{2x_m-1})\ \sqrt{2x_m-1}}{1-x_m}\cdot\frac{V^2}{V_m^2}\right] \tag{205}$$

setting

$$\frac{(x_m-\alpha\ \sqrt{2x_m-1})\sqrt{2x_m-1}}{1-x_m}=c_1'. \tag{206}$$

we finally get

$$\eta=\frac{\eta_m}{\sqrt{2x_m-1}}\left(2\beta-c_1'\frac{V^2}{V_m^2}\right)$$

The last equation gives the important law of variation of the efficiency η of the engine-propeller system as function of the speed V.

Let us finally find the equation of the power of the engine-propeller system. We have

$$L_a=\frac{N^3\ (1-h^2x^2)\ (L_a)_m}{2N_m^3\ (1-x_m)}$$

After corresponding substitutions we find

$$L_a=(L_a)_m\left[\alpha\ \sqrt{2x_m-1}\cdot\ \frac{V}{V_m}+\frac{2\beta\ (1-x_m)}{\sqrt{2x_m-1}}\cdot\ \frac{V_m}{V}\right] \tag{207}$$

The relations (191), (202), (206) and (207) are to first approximations, the equations of the main characteristics of the engine-propeller system.

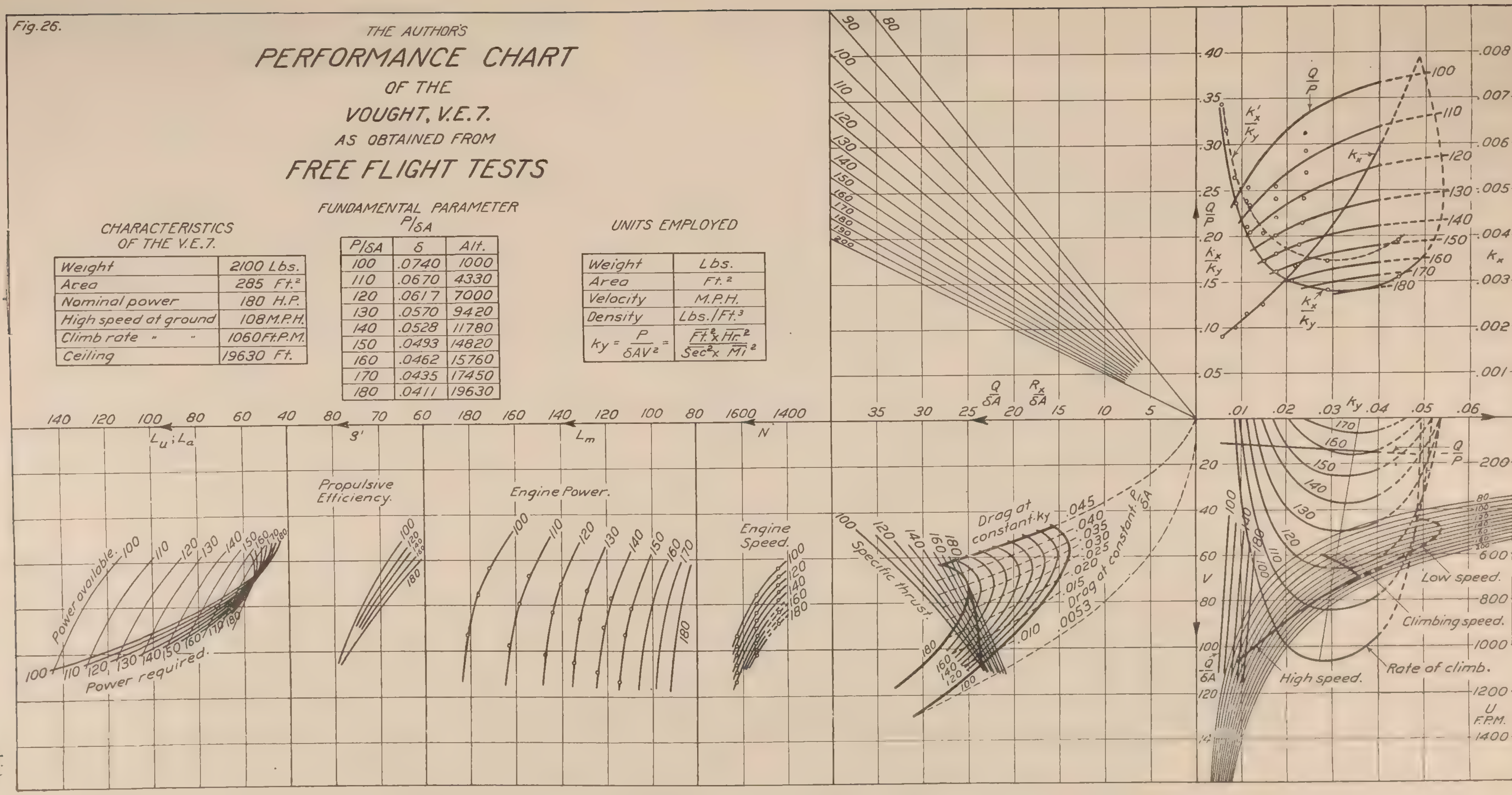


Fig. 26.
THE AUTHOR'S
PERFORMANCE CHART
OF THE
VOUGHT, V.E.7.
AS OBTAINED FROM
FREE FLIGHT TESTS
CHARACTERISTICS OF THE V.E.7.

Weight	2100 Lbs.
Area	285 Ft.²
Nominal power	180 H.P.
High speed at ground	108 M.P.H.
Climb rate " "	1060 Ft.P.M.
Ceiling	19630 Ft.

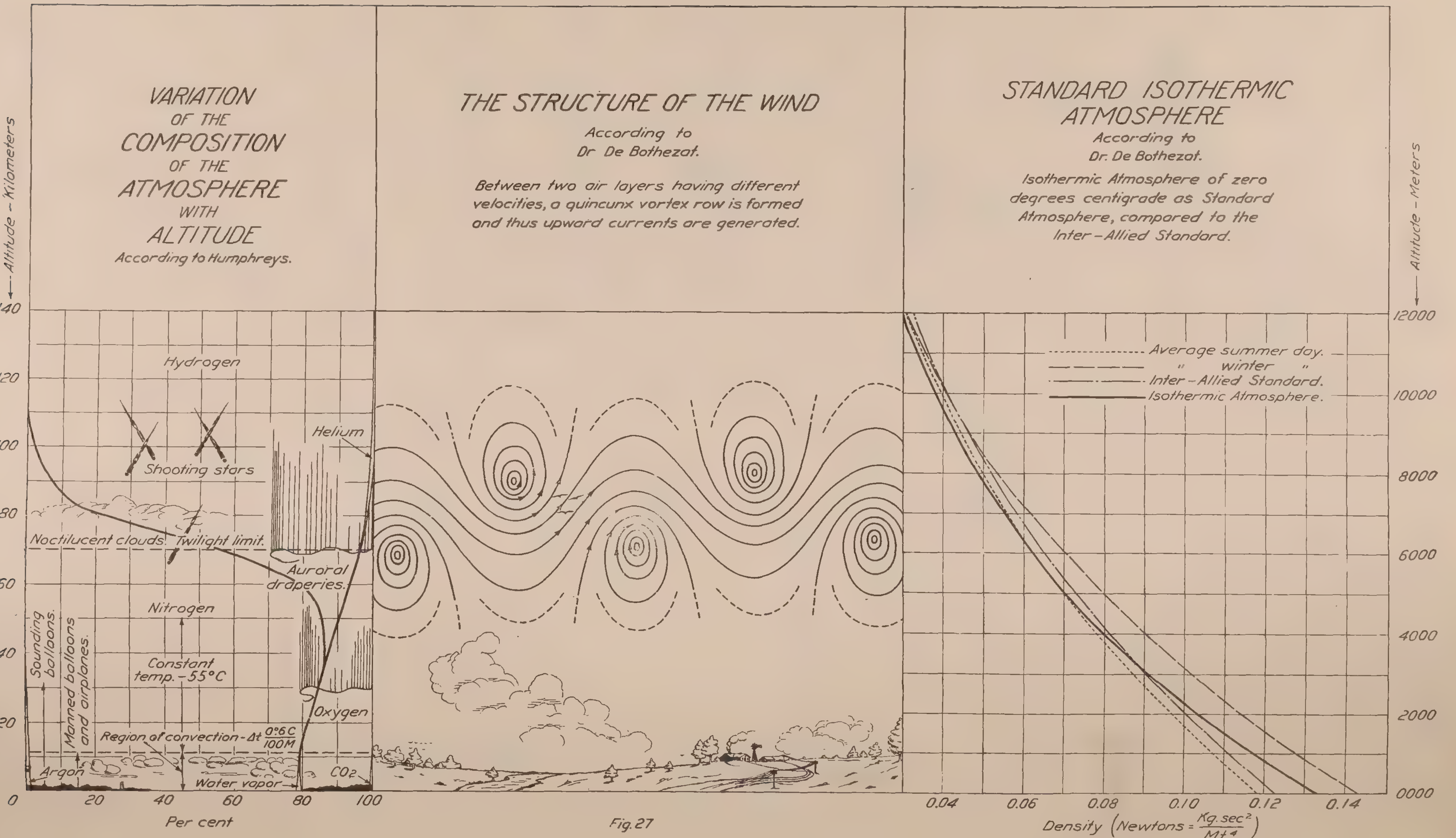
FUNDAMENTAL PARAMETER $P/\delta A$

$P/\delta A$	δ	Alt.
100	.0740	1000
110	.0670	4330
120	.0617	7000
130	.0570	9420
140	.0528	11780
150	.0493	14820
160	.0462	15760
170	.0435	17450
180	.0411	19630

UNITS EMPLOYED

Weight	Lbs.
Area	Ft.²
Velocity	M.P.H.
Density	Lbs./Ft.³
$k_y = \frac{P}{\delta A V^2} =$	$\frac{Ft.^2 \times Hr.^2}{Sec^2 \times Mi^2}$

L_u; L_a
δ'
L_m
N
Propulsive Efficiency.
Engine Power.
Engine Speed.
Power available.
Power required.
Specific thrust.
Drag at constant k_y
Drag at constant $\frac{P}{\delta A}$
$\frac{Q}{\delta A}$
$\frac{R_x}{\delta A}$
$\frac{Q}{P}$
$\frac{k_x'}{k_y}$
$\frac{k_x}{k_y}$
k_x
k_y
V
U F.P.M.
Low speed.
Climbing speed.
Rate of climb.
High speed.



VARIATION OF THE COMPOSITION OF THE ATMOSPHERE WITH ALTITUDE
According to Humphreys.
Altitude - Kilometers
140
120
100
80
60
40
20
0
20
40
60
80
100
Per cent
Hydrogen
Shooting stars
Helium
Noctilucent clouds.
Twilight limit.
Auroral draperies.
Nitrogen
Sounding balloons.
Manned balloons and airplanes.
Constant temp. -55°C
Oxygen
Region of convection - Δt 0°.6 C / 100 M
Argon
Water vapor
CO2
THE STRUCTURE OF THE WIND
According to Dr De Bothezat.
Between two air layers having different velocities, a quincunx vortex row is formed and thus upward currents are generated.
STANDARD ISOTHERMIC ATMOSPHERE
According to Dr. De Bothezat.
Isothermic Atmosphere of zero degrees centigrade as Standard Atmosphere, compared to the Inter-Allied Standard.
Altitude - Meters
12000
10000
8000
6000
4000
2000
0000
Average summer day.
" winter "
Inter-Allied Standard.
Isothermic Atmosphere.
0.04
0.06
0.08
0.10
0.12
0.14
Density (Newtons = Kg.sec² / Mt⁴)

Fig. 27

REPORT No. 98

DESIGN OF WIND TUNNELS AND WIND TUNNEL PROPELLERS, II

By F. H. NORTON and EDWARD P. WARNER
Langley Memorial Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Field, Va.

REPORT No. 98.

THE DESIGN OF WIND TUNNELS AND WIND TUNNEL PROPELLERS, II.

By F. H. Norton and Edward P. Warner, National Advisory Committee for Aeronautics.

SUMMARY.

This report is a continuation of National Advisory Committee for Aeronautics Report No. 73, and was undertaken at the Langley Memorial Aeronautical Laboratory for the purpose of supplying further data to the designer of wind tunnels. Particular emphasis was placed on the study of directional variation in the wind stream. For this purpose a recording yaw-meter, which could also be used as an air speed meter, was developed, and gave very satisfactory results. It is regrettable that the voltage supplied to the driving motor was not very constant, due to varying loads on the line, but as this motor was of a lightly loaded induction type, the variation in speed was not as large as the variation in voltage. The work was carried on both in a 1-foot model and the 5-foot full-sized tunnel, and wherever possible a comparison was made between them. It was found that placing radial vanes directly before the propeller actually increased the efficiency of the tunnel to a considerable extent. The placing of a honeycomb at the mouth of the experimental portion was of the greatest aid in improving the flow, but, of course, somewhat reduced the efficiency. Several types of diffusers were tried in the return air, but only slight improvement resulted in the steadiness of flow, they not being nearly as effective as the honeycomb.

APPARATUS.

The efficiency of the tunnel and the slip of the propeller were determined by the same method as described in Report No. 73, but to better record the fluctuations in velocity and direction a recording instrument was constructed. This instrument, as shown in Figs. 1 and 2, consists of a thin mica diaphragm whose movement rotates a very light spindle containing a small silvered mirror. Light from an illuminated slit is transmitted by a lens to this mirror and the reflected beam is then focused on a moving photographic film so that any movement of the mica diaphragm is recorded as a continuous curve. By this method any small and rapid variation in the air flow of the tunnel is indicated and recorded by means of a Pitot-static tube which is connected to the two compartments separated by the diaphragm, and any change in direction is recorded in the same way by connecting the sides of a yawhead to the compartments on opposite sides of the mica diaphragm. The Pitot and the connecting tubes are made comparatively large so that any rapid fluctuation in velocity can be immediately transmitted to the diaphragm without damping or lag. Over 50 records were taken but only a few typical ones are reproduced here. Numerous experiments on the efficiency of tunnels and on speed fluctuation have previously been made in England.[1], [2], [3]

EFFICIENCY AND SLIP WITH NEW PROPELLER.

In order to give a more even flow of air in the exit cone a new propeller was designed for the model tunnel having a larger pitch at the tip so that the air in this portion would be drawn through with a relatively greater velocity. In every other respect this propeller is very similar to the propeller used in the test described in Report No. 73, which, owing to a piece of wood being dropped in the running tunnel, was completely destroyed. In Fig. 3 is shown the efficiency of this propeller when working in a parabolic cone and in a straight cone. It will be noted that in the same way as with the first propeller the straight cone is considerably more

[1] An Investigation into the Steadiness of Wind Channels, by L. Bairstow and Harris Booth: British Advisory Committee for Aeronautics, R. & M. 67, September, 1912.

[2] Experiments on Models of a "Duplex" Wind Channel, by T. E. Stanton and J. H. Hyde: Brit. A. C. A., R. & M. 522, November, 1917.

[3] Reports on Tests of a Model of the Proposed 7-foot Wind Channel at the R. A. E., by C. G. Sandison and W. K. Alford: Brit. A. C. A., R. & M. 574, December, 1918.

efficient at high speed than the parabolic cone. In Fig. 4 is shown the slip of this propeller in the parabolic cone and in the straight cone and it is noted that the slip is less at high speeds for the straight cone. It is then evident that the straight cone is aerodynamically superior to the parabolic cone, in addition to being easier to build.

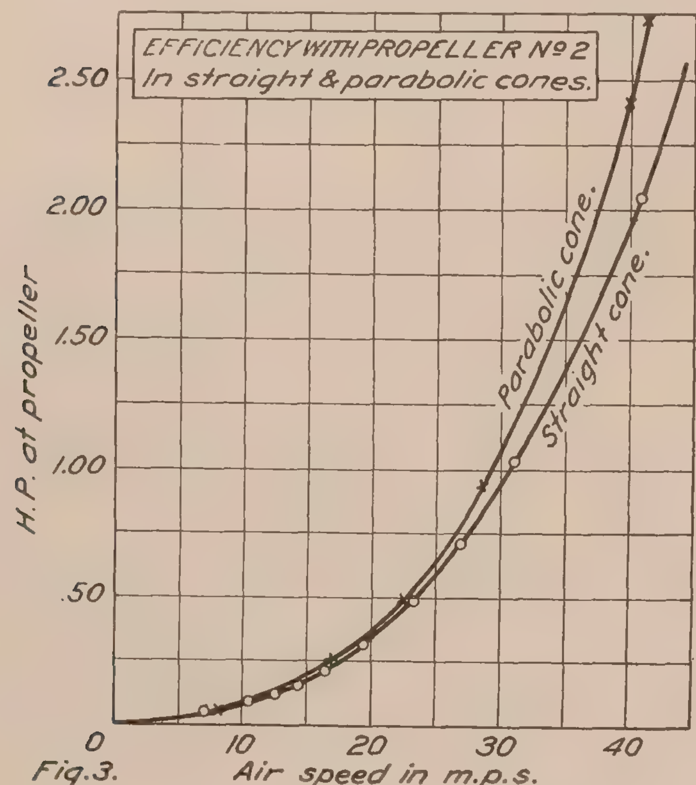


Fig. 3.

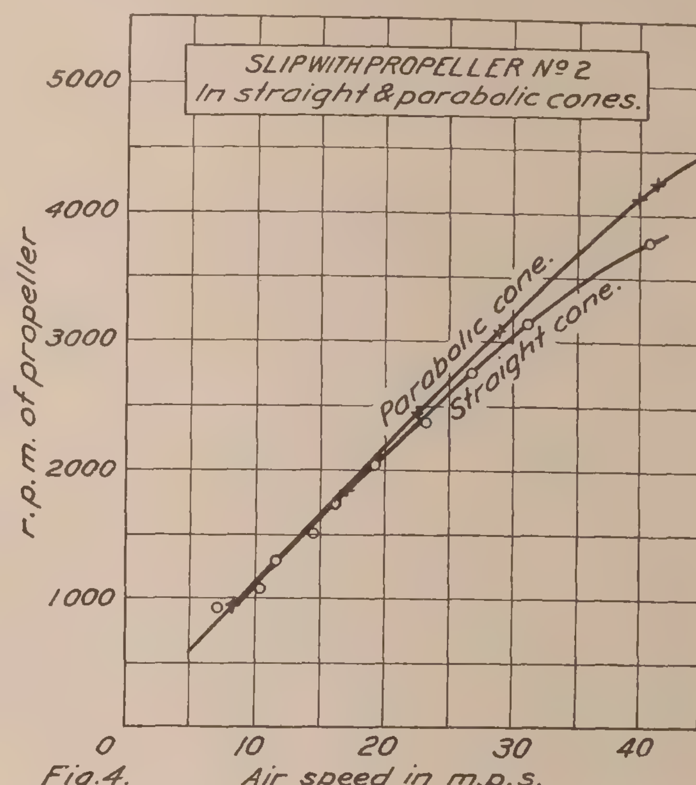


Fig. 4.

EFFECT OF SIZE OF THE ROOM.

All the test runs described in Report No. 73 were conducted in a large room, approximating to free-air conditions. In the tests described in this report a temporary room was built around the tunnel, representing to scale the building provided for the 5-foot N. A. C. A. wind tunnel; and all runs except those shown in Figs. 3 and 4 were made in this model room. The cross section of the model room and the wind tunnel are shown in Fig. 5. For the same power this room decreased the air speed from 69 to 59 miles per hour or a decrease of 14.5 per cent. In the small room the maximum variation of speed was ±7 per cent and the maximum variation in direction was ±10°. The air speed records show that for the first 20 seconds after starting, in the large room, and for the first 10 seconds in the small room, the air speed is very steady, and that the fluctuations suddenly appear at a definite time and will be indicated on the record. This appearance of sudden fluctuations seems to indicate that the large part of the speed fluctuations are due to the disturbed air from the propeller as it returns through the room to the entrance cone.

EFFECT OF RADIAL VANES.

Eight radial vanes 3 mm. thick and 450 mm. deep were placed symmetrically in the exit cone immediately before the propeller. These vanes joined in the center in a stationery spinner which was of the same diameter as the propeller base. (Fig. 6.) These vanes actually increased the speed of the air in the tunnel for the same power by 5 per cent, but the fluctuations in direction and velocity remained unchanged. In order to determine what part of the vane gave the increased efficiency, 25 mm. was cut off of the outer end of each vane and the run repeated which gave a 3 per cent increase in speed for the same power over the tunnel with no vanes. Again the vanes were cut off on the end 75 mm. and in this case the same speed was obtained as with the tunnel without vanes. This seems to show that it is the whole area of the vane which acts as a straightener for the air flow and that no particular part is especially valuable in increasing the efficiency of the tunnel. Eight additional vanes 3 mm. thick were then placed along the inner surface of exit cone, each vane being 75 mm. wide. This distribution of vanes decreased the speed by 12 per cent for the same power and the variation in speed was ±6 per cent

FIG. 1.—RECORDING AIR SPEED METER.

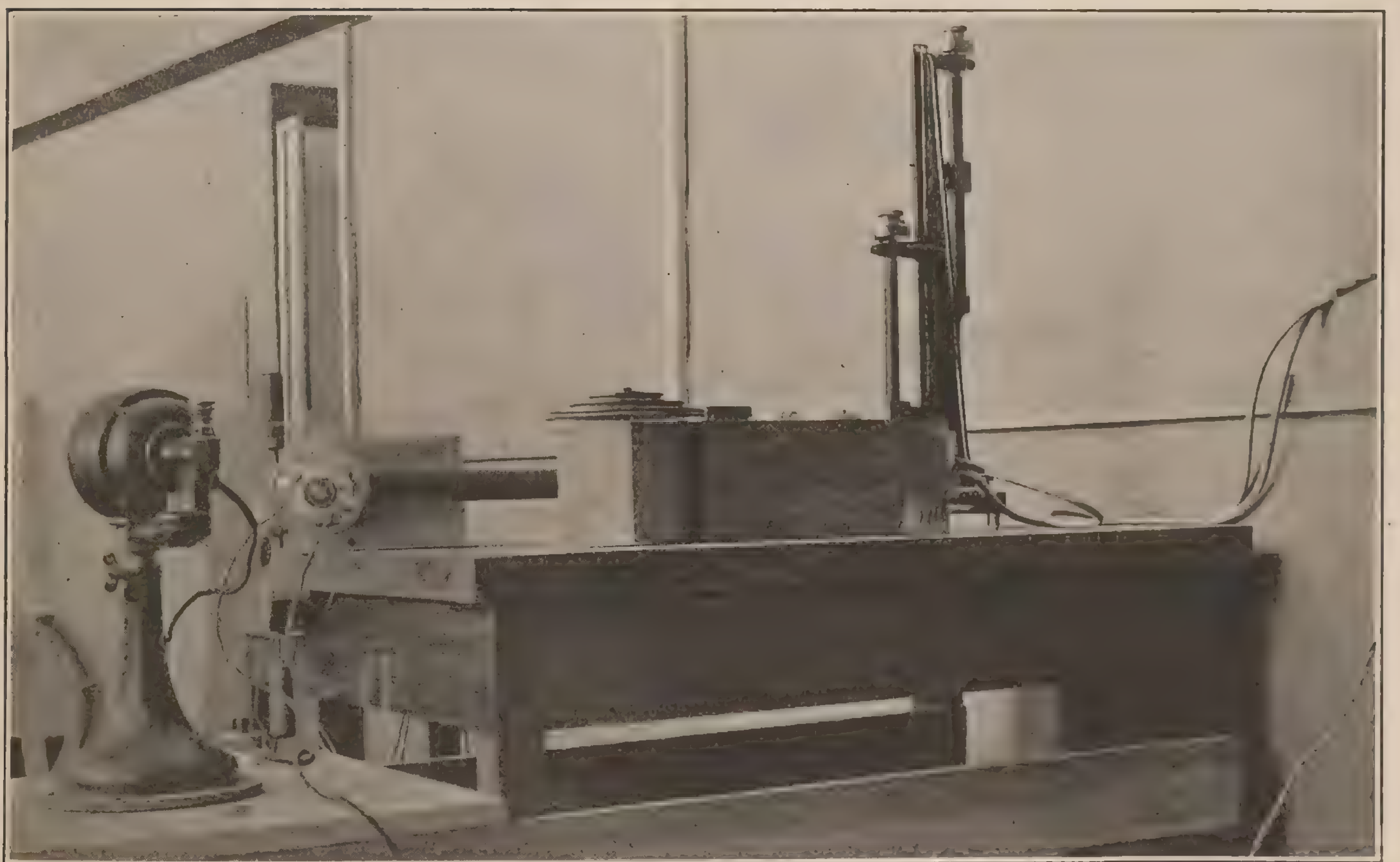
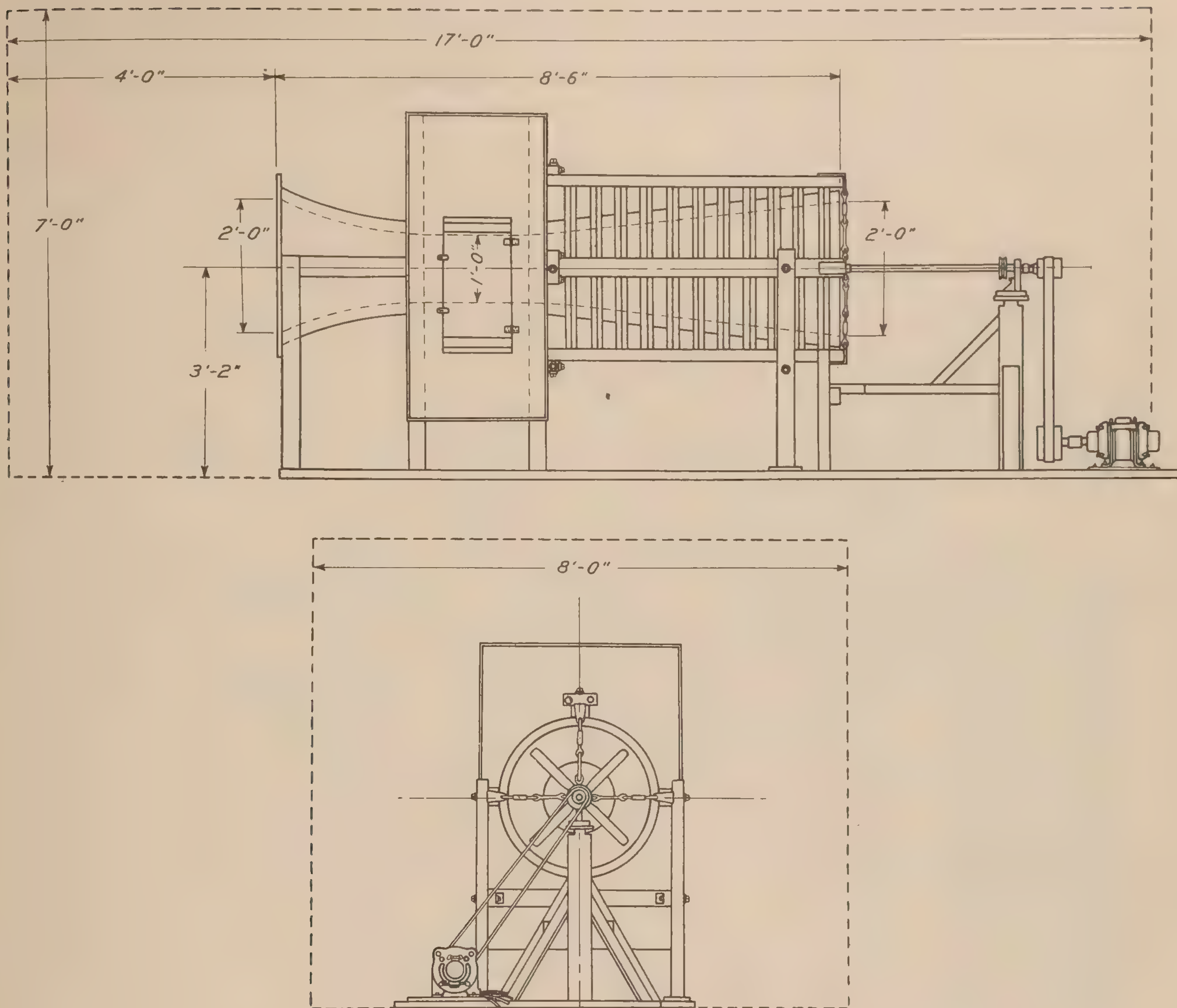
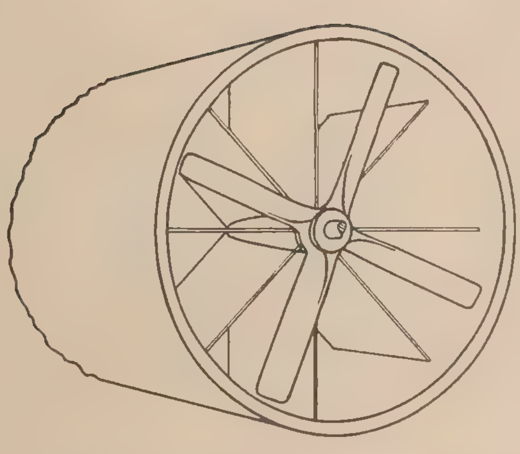


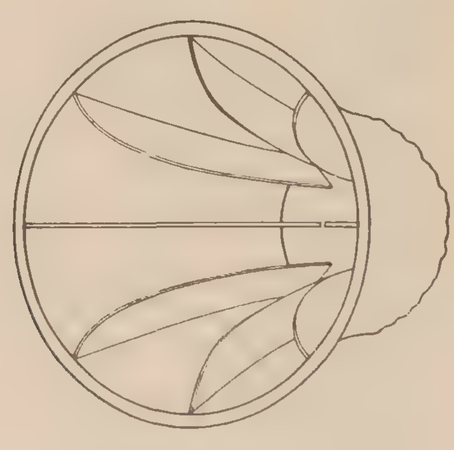
FIG 2.—RECORDING AIR SPEED METER WITH ILLUMINATING AND RECORDING APPARATUS.



MODEL WIND TUNNEL AND ROOM
Fig. 5



RADIAL VANES IN EXIT CONE
Fig. 6.



VANES IN ENTRANCE CONE
Fig. 7.

and the variation in direction was ±10°. The same vanes were then placed in the entrance cone, as shown in Fig. 7, and in this case the speed was decreased by 8 per cent and the variation in direction was ±8°. With this type of vane in both the exit and entrance cone the speed was decreased by 20 per cent for the same power and the variation in direction was ±8°. It is evident from these tests that the narrow vanes in either the exit or entrance cones are of little value in any way.

EFFECT OF PLACING SCREEN ACROSS THE TUNNEL.

A section of chicken wire of 25 mm. mesh was placed across the exit cone 45 centimeters ahead of the propeller. The use of the chicken wire decreased the speed by only 3 per cent, so it does not seem that this distribution of screen would be of any great harm to the efficiency of the tunnel and it is of great use in preventing small objects from being drawn into the propeller. A piece of window screen placed at the beginning of the straight portion of the tunnel decreased the speed by 14 per cent and the fluctuation in speed was −12 per cent and was −10 per cent in direction, showing that the screen in no way helps the steadiness of flow for the particular condition of this test. Screens have been used to advantage in other tunnels. With window screen at the mouth of the entrance cone the speed was decreased by only 7 per cent.

EFFECT OF PLACING SPINNERS BEFORE THE PROPELLER.

A spinner 75 mm. in diameter and 450 mm. long was supported by steel wires before the propeller, as shown in Fig. 8. The use of this spinner seemed to have no material effect on the air flow.

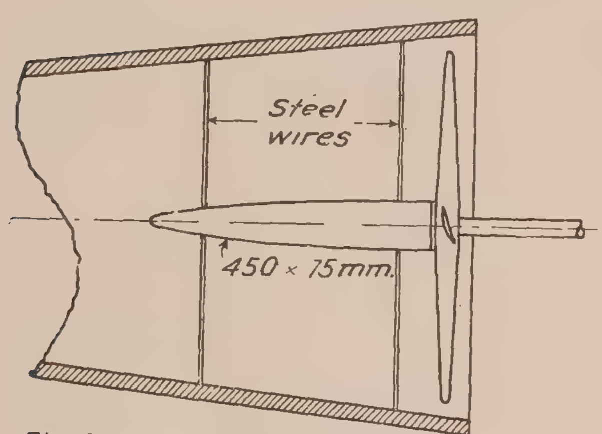


Fig. 8.

SPINNER

THE EFFECT OF EXTENDING THE EXIT CONE BEYOND THE PROPELLER.

By extending the exit cone as shown in Fig. 9, there was no change of the air flow inside the tunnel, but the tangential flow, which had been noticed before with the propeller, was somewhat straightened out, and the air flow was more directly to the rear through the extension of the cone. A cylinder was then attached to the propeller end of the tunnel as shown in Fig. 10, which decreased the air speed about 5 per cent, the air issuing from the tunnel at a considerably

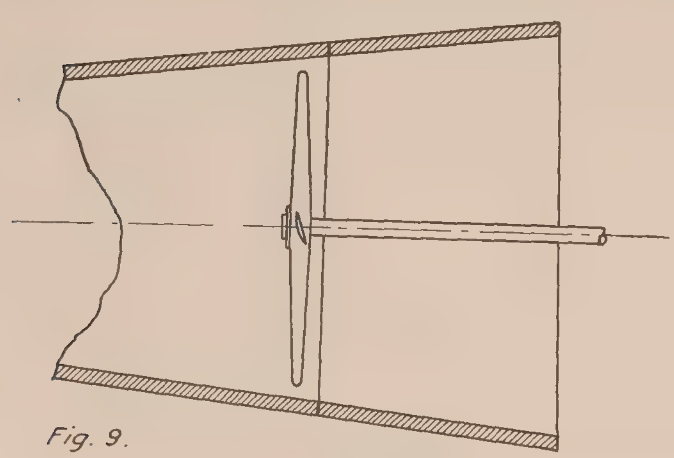


Fig. 9.

CONICAL EXTENSION

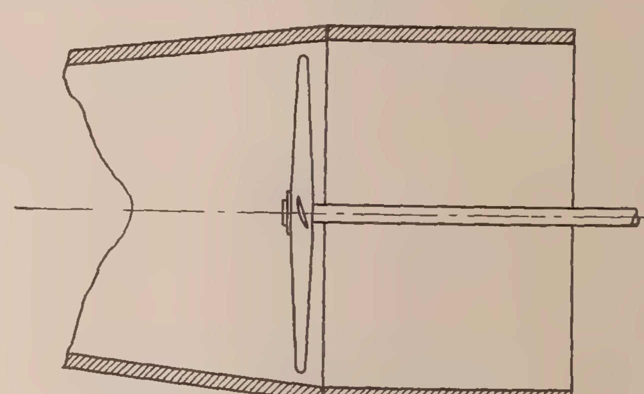


Fig. 10.

CYLINDRICAL EXTENSION

higher velocity and in a more compact stream, the borders of the stream still being sharply defined at a distance of 20 feet. As extensions of this kind mean a larger and longer building for the wind tunnel there would certainly be no advantage in using them.

EFFECT OF HONEYCOMBS.

A honeycomb was constructed as shown in Fig. 11 and was placed at the entrance to the straight portion of the tunnel. Owing to the difficulty in obtaining thin-walled metal tubing and to the expense of constructing honeycombs of this type, only this one was tried. It is

quite evident, however, even from this one test that the honeycomb is of the greatest importance in straightening out the flow. The speed is reduced 18 per cent and the energy ratio 45 per cent by this honeycomb, but the maximum speed variation was only ±2 per cent and the variation in direction was reduced to ±0.5°. In order to show more clearly the great increase in steadiness of flow, a curve taken with a recording yawmeter is shown for the open tunnel and for the tunnel containing the honeycomb. (Figs. 12 and 13.) It is evident from these how great is the advantage of the honeycomb. As the length diameter ratio in the tubes of this honeycomb are only 2½ it is quite possible that by using longer tubes the flow would be even better and the reduction in speed should not be appreciable. There seems to be no doubt from these tests that the honeycomb is absolutely essential in most wind tunnels.

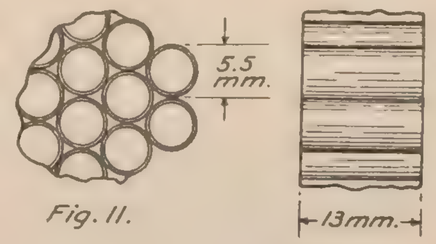


Fig. 11.
SECTION OF HONEYCOMB

EFFECT OF DIFFUSERS.

The first diffuser tried is shown in Fig. 14 and consists essentially of a cubical box of which both sides are perforated with small holes, whose diameter is equal to the thickness of the wall of the box and whose spacing between centers is about twice that of the diameter of the hole. This

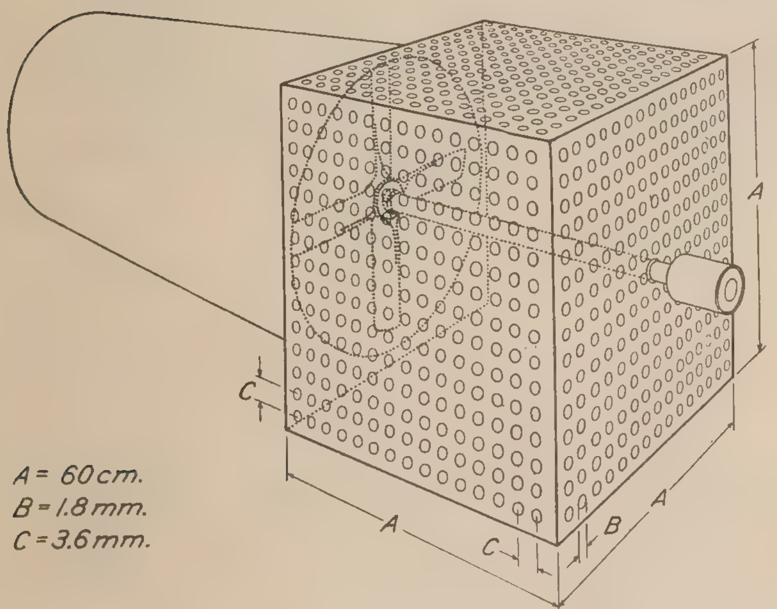


Fig. 14. CUBICAL DIFFUSER

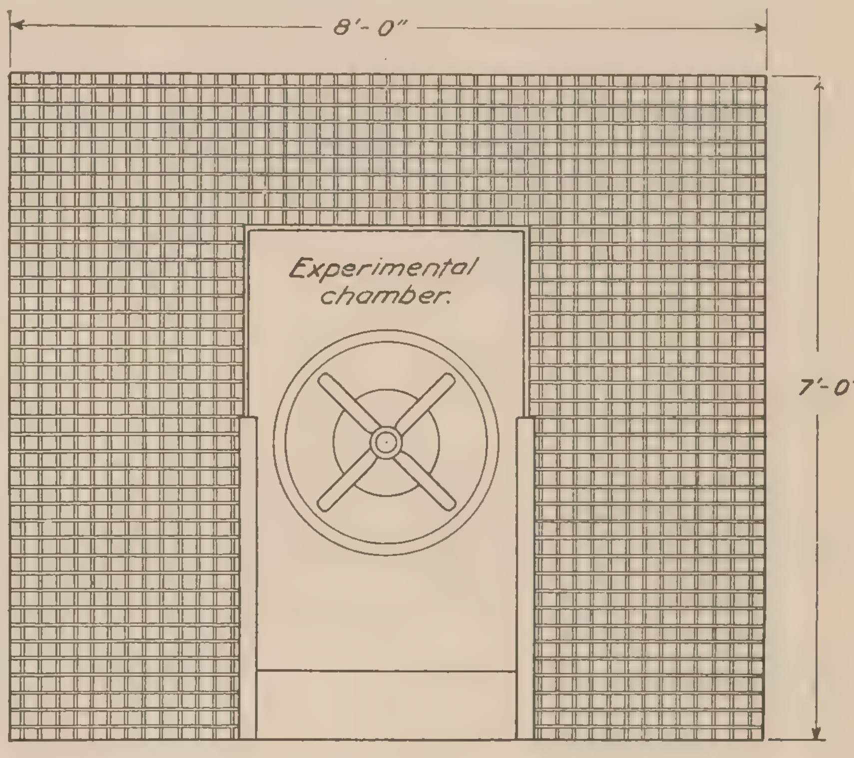
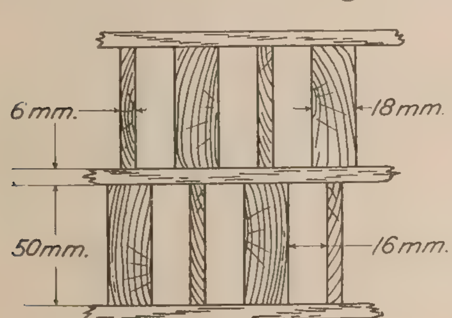


Fig. 15 DIFFUSER.

box was connected rigidly to the rear of the exit cone so that all the air passing through the propeller must escape through these small holes. It was hoped in this way to break up any pulsations which would originate from the propeller. This arrangement decreased the speed of the tunnel by 7 per cent and the maximum variation of speed was ±6 per cent and the direction variation was ±5°, so that it would seem that the flow is slightly straightened, but nowhere near as much as with the honeycomb. A second diffuser was tried as shown in Fig. 15, which consists of a latticework across the tunnel room at the experimental chamber consisting of 50 mm. square cells having a 6 mm. wall with a length 2¼ times their diameter. This diffuser only reduced the speed of the tunnel by 2 per cent, and the maximum variation was ±7 per cent, and the variation in direction was ±5°. Although this diffuser has very little effect on the efficiency of the tunnel, at the same time it does not much improve the steadiness of flow. A third diffuser was constructed as shown in Fig. 16 and placed in the same position as the last. This diffuser decreased the air speed for the same power about 5 per cent, the variation in velocity was ±5 per cent, and the variation in direction was ±4°, showing only a slight



SECTION OF DIFFUSER.
Fig. 16.

improvement over the open room. It seems strange that these diffusers did not improve the air flow more, as the British have found that diffusers greatly improve the flow in their tunnels. The results of these tests would not, however, justify the use of a diffuser in a full-sized tunnel because of the rather large expense of construction of such a piece of apparatus.

EFFECT OF PERFORATING THE STRAIGHT PORTION OF THE TUNNEL.

In order to determine the effect on air flow of opening the doors in the cylindrical portion of the tunnel and in using small holes for the introduction of apparatus, various tests were made on the model in order to see how this would effect the efficiency and steadiness of flow. Also the velocity of the air in the experimental chamber was determined by a small anemometer. A slot was first cut in the cylinder parallel to its axis and one-fifteenth of the diameter wide, running the whole length of the experimental chamber. The air flow extended out about the width of the slot from the walls of the cylinder, and beyond this there was no flow in the chamber and the efficiency of the tunnel was not appreciably affected. This slot was then increased in width to one-sixth of the diameter of the tunnel, thus decreasing the efficiency of the tunnel very slightly, and the flow of air extended about one-sixth of the tunnel diameter into the experimental chamber nearest the exit cone, but this air flow was less marked as the distance to the entrance cone was decreased. When the width of the slot was increased to three-eighths of the tunnel diameter the efficiency was decreased about 15 per cent and the air flow extended two-thirds of the width of the slot into the experimental chamber, near the exit cone, but there was no flow elsewhere in the experimental chamber.

TESTS IN FULL-SIZED TUNNEL.

A few tests were made in the large tunnel in order to afford a comparison with the model. In Fig. 17 is shown the slip in the large tunnel. In comparing this with a similar condition in the model tunnel (Fig. 4) it is seen that for the same air speed the revolutions per minute is

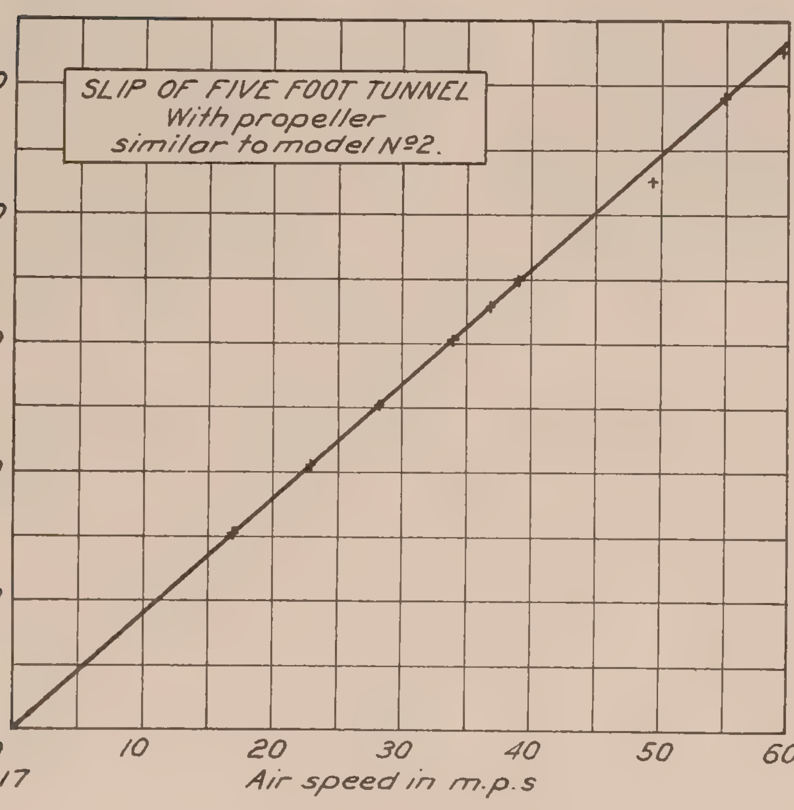


Fig. 17

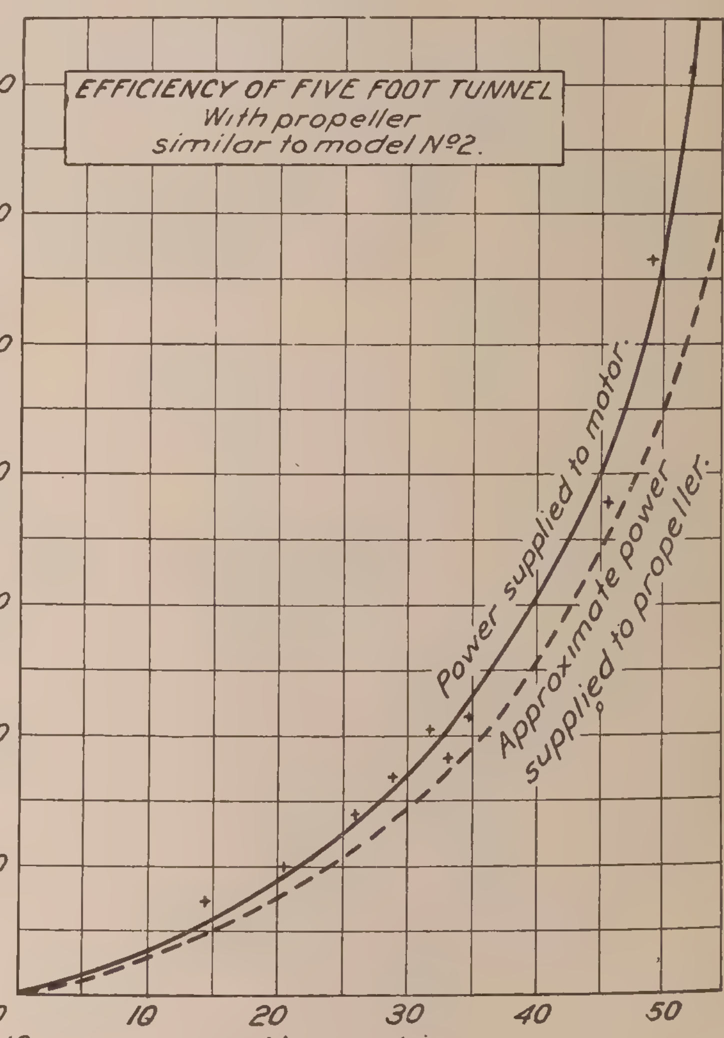


Fig. 18.

5.7 times as large in the small tunnel as in the large one. Theoretically, the ratio should be exactly 5, but the fact that the model test was run in a proportionately larger room would account for this difference.

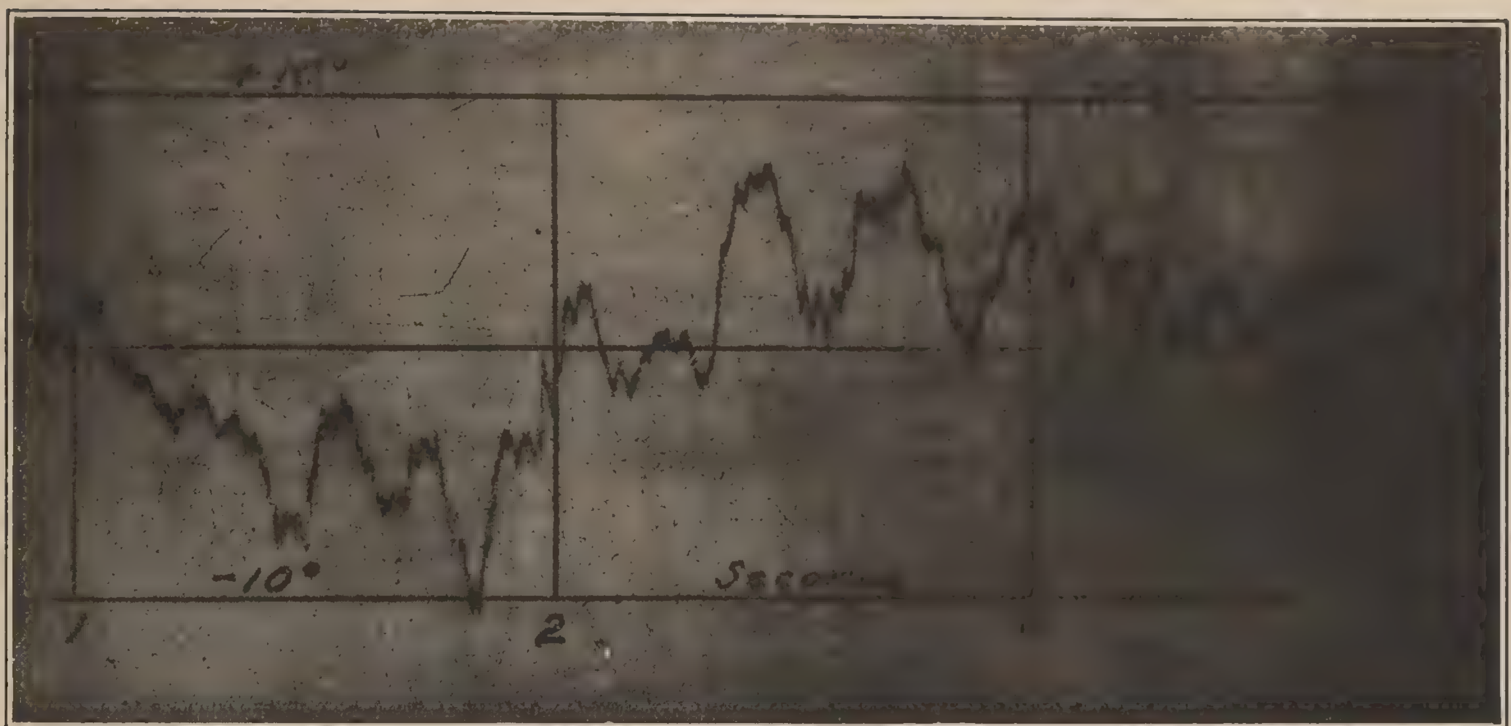


FIG. 12.—VARIATION IN DIRECTION IN THE MODEL TUNNEL WITH NO HONEYCOMB.

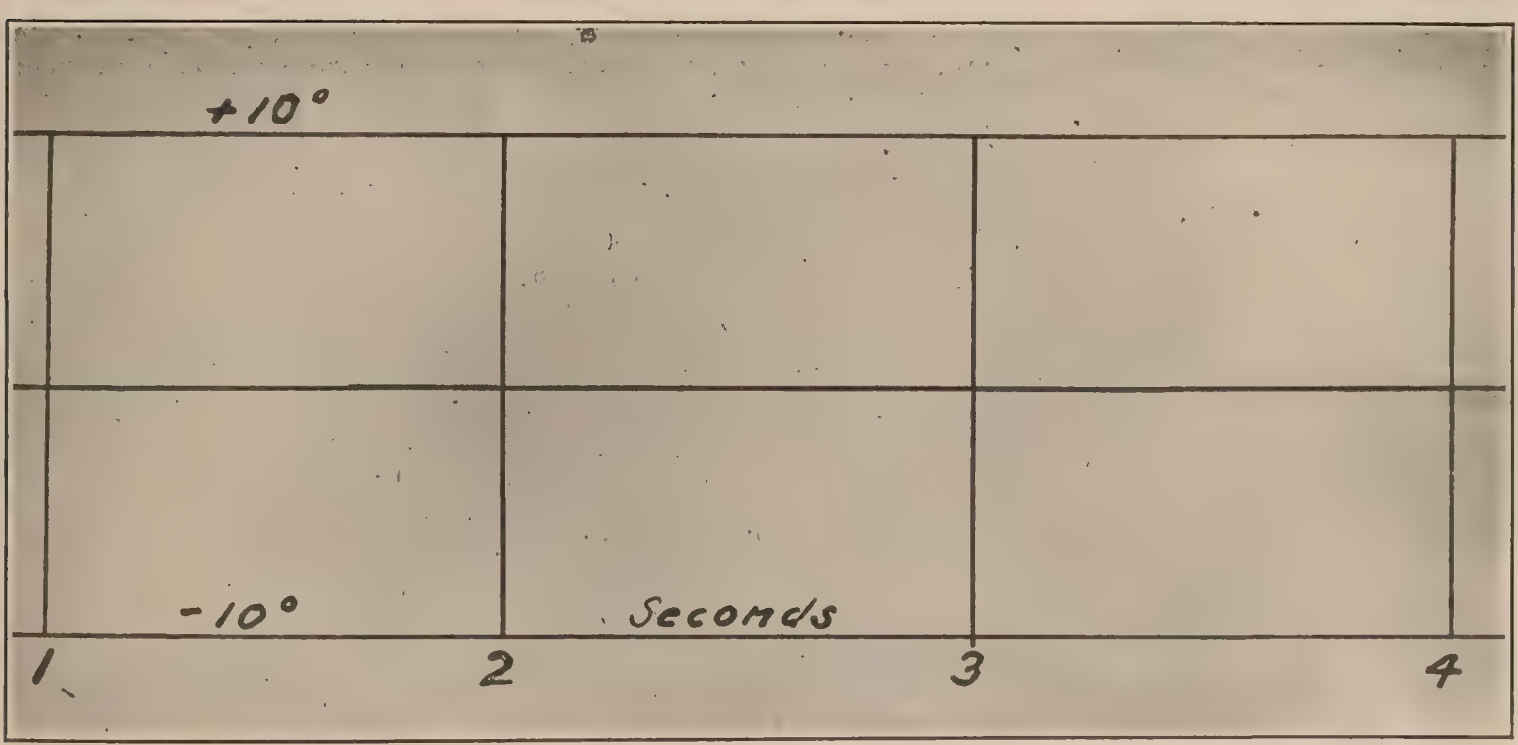


FIG. 13.—VARIATION IN DIRECTION IN THE MODEL TUNNEL WITH A HONEYCOMB.

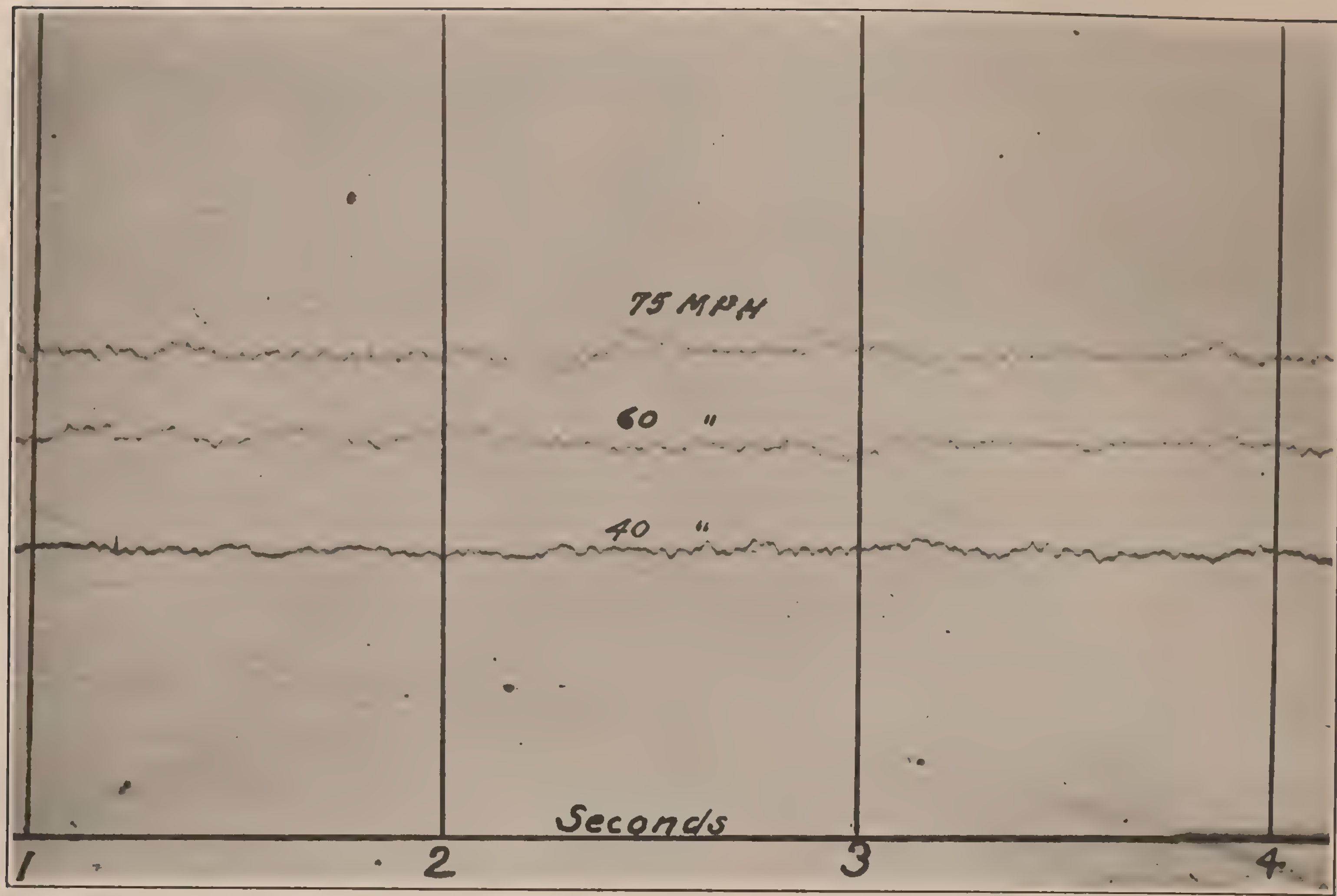


FIG. 19.—VELOCITY VARIATIONS IN LARGE TUNNEL WHEN THE DRIVING MOTOR WAS CONNECTED TO A 25 K. W. GASOLINE GENERATING SET.

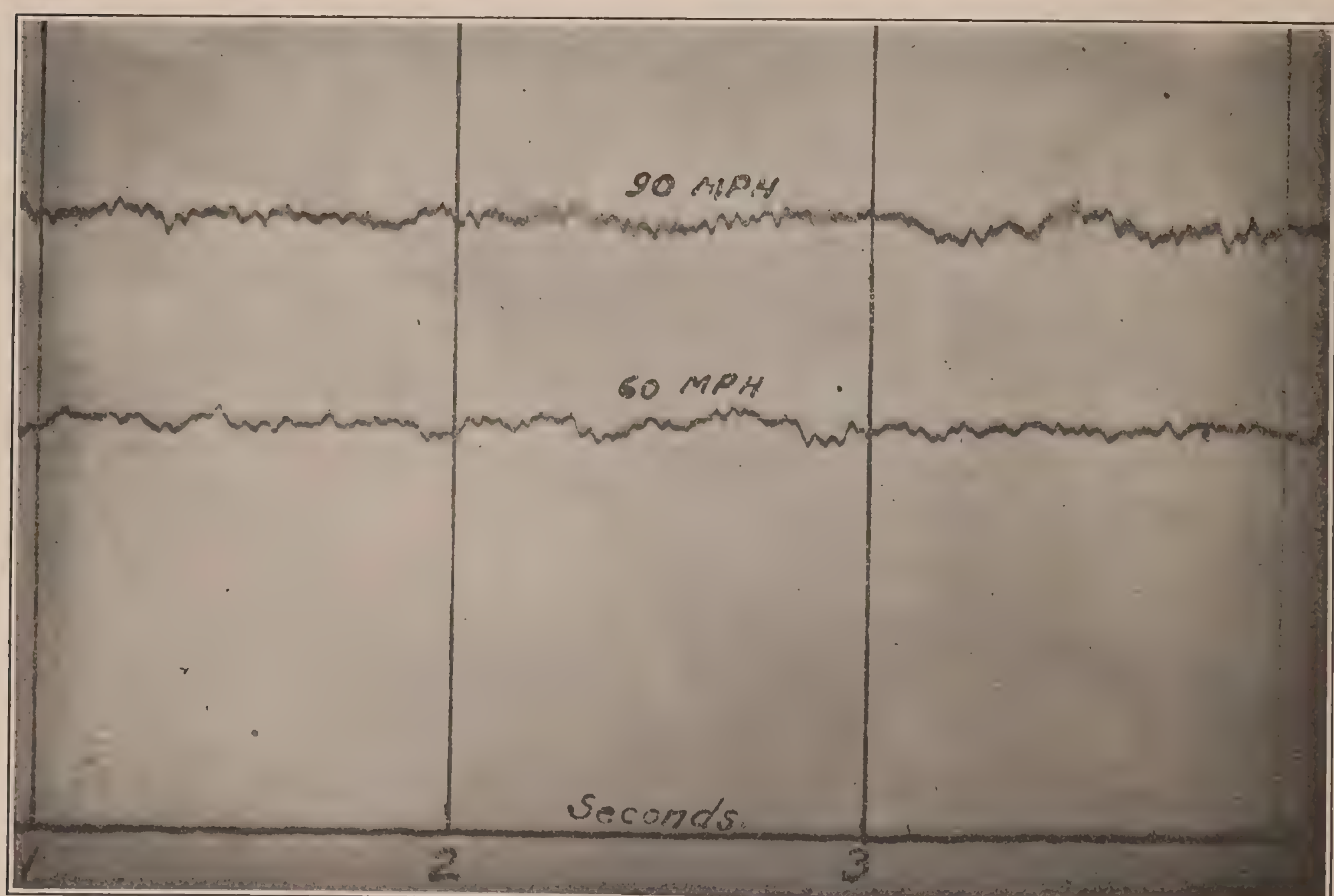


FIG. 20.—VELOCITY VARIATIONS IN LARGE TUNNEL WITH DRIVING MOTOR CONNECTED TO A 300 K. W. LIBERTY GENERATING SET.

As the exact efficiency of the driving motor in the large tunnel is unknown, a curve of horsepower supplied to the motor is plotted against air speed, but to give some idea of the power supplied to the propeller a dotted curve is drawn from the estimated motor efficiency. (Fig. 18.)

In comparing this curve with the one obtained in the model, it is seen that the full-sized tunnel is slightly more efficient, so that results may be taken from models to safely predict the performance of the full-sized tunnels. It is also interesting to notice that the power does not increase as rapidly as the cube of the speed but more nearly as $V^{2.5}$, although, as the efficiency of the motor is not exactly known, the value of the exponent can not be determined very closely.

Records were taken in the full-sized tunnel of variations in velocity, and these are reproduced in Figs. 19 and 20. In the first figure the wind-tunnel motor was connected to a gasoline-driven generator of 25 kilowatts and records taken at several speeds. In Fig. 20 the motor was connected to a 300-kilowatt generator driven by a Liberty motor. The most important characteristic of these records is that the magnitudes of the fluctuations do not increase as rapidly as the air speed, so that at the higher speeds, quite contrary to expectations, the velocity is relatively steadier. The maximum variation in air speed at 90 miles per hour was about $\pm$ 1.5 per cent, whereas in the model it was about $\pm$ 2 per cent, so that it would seem that the steadiness was about the same in any size of tunnel.

Yawmeter records were also taken in the large tunnel, but were not reproduced, as they show practically a straight line, indicating that the honeycomb was satisfactorily straightening out the flow.

NATURAL PERIODS OF TUNNEL.

A wind tunnel acts as an open organ pipe and its natural period will be given by:

$$P=\frac{4l}{V}$$

where l is the length of the tunnel in feet,
and V is the velocity of sound, or 1,040 ft./sec.

The model tunnel would then have a period of 0.03 seconds and the large tunnel a period of 0.15 seconds. Vibrations of this nature are very evident audibly in the tunnels at certain speeds, but do not seem to be present on the records, as the pitot tube is very nearly at the node of the vibration. The honeycomb has a considerable influence in damping these vibrations, which are more of a curiosity than of any practical interest.

AUTOMATIC REGULATORS.

As it is not practical to supply a constant voltage to a wind tunnel, although some tests have been made with storage batteries where an extremely constant speed was required, it is either necessary to keep the voltage constant as nearly as possible by hand regulations or use some type of automatic regulator. In small tunnels it is quite easy to regulate the wind by hand, but in larger tunnels the inertia of the moving parts is so great that there is considerable amount of lag between the change in regulation and the response of the air speed, making hand regulation very difficult. A very complicated regulator has been constructed at Göttingen (N. A. C. A. File No. 5346–10) and seems to hold the velocity quite constant. There are also numerous electrical devices for maintaining a constant motor speed, and some of these regulators will hold the speed within 0.1 per cent. It seems probable, however, that even if the revolutions per minute of the propeller is constant that there will still be fluctuations in the air speed, so that a successful regulator must be actuated by the air flow. There is a great deal of work to be done on such regulators, and the N. A. C. A. intends to carry on work of this kind in the near future.

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CONCLUSIONS.

The qualities that should be aimed at in wind-tunnel design in order of their importance are:

1. Constant direction of flow.
2. Constant velocity of flow.
3. Uniform velocity across section.
4. Efficiency.
5. Ease of working around tunnels.
6. Simplicity and cheapness of construction.

A good many of these qualities are contradictory, and the best compromise must be made between them and the type of work that is to be undertaken. For example, a tunnel for testing instruments should have a high efficiency, but need not have a very steady flow. On the other hand, a tunnel for testing wings should have its efficiency somewhat lowered in order to obtain a steady flow. It is quite possible to so arrange the honeycomb and diffusers that they may be removed when it is desired to obtain the highest speed. It would also be of value to make it possible to open the ends of the building, as there are many days when the wind would have little effect on the steadiness, and the efficiency would apparently thus be considerably increased. This arrangement would also make it possible to cool off the air in the building in a very short time, an advantage that would be greatly appreciated in hot weather.

This work seems to show conclusively that a straight exit cone is more efficient than a curved one, and it is certainly cheaper to construct. Diffusers affect the air flow very little, and they do not seem to warrant the expense of construction. Honeycombs, however, are of the greatest value and should be placed in every tunnel.

REPORT No. 99

ACCELERATIONS IN FLIGHT

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National Advisory Committee for Aeronautics
Langley Field, Va.

NOTE.—This report was prepared for the Bureau of Construction and Repair, Navy Department, and is published by permission of the Chief Constructor, U. S. Navy

REPORT No. 99.

ACCELERATIONS IN FLIGHT.

F. H. NORTON and E. T. ALLEN, National Advisory Committee for Aeronautics.

INTRODUCTION.

This work on accelerometry was carried out at the Langley Field Station of the National Advisory Committee for Aeronautics at the request of the Bureau of Construction and Repair, United States Navy, for the purpose of obtaining the magnitude of the load factors in flight and to procure information on the behavior of an airplane in various maneuvers.

When an airplane is flying on a straight and level course a spring scale with a 1-pound weight attached to it would record 1 pound. If, however, the plane is put into a turn or a zoom the scale will no longer record 1 pound, but may record 2 or even 3 pounds—that is, the apparent weight of objects on the airplane have increased two or three times. Should the control be suddenly pushed forward to nose the plane over, the spring scale may read zero—that is, an object on the plane would have no weight. When a spring scale is used in this way the pound graduations on the scale represent accelerations in terms of the acceleration of gravity g, which is in English units about 32 feet per second per second.

If the average loading of the wings is 10 pounds per square foot in level flight, during a maneuver in which the spring scale reads 3 the wings would then be carrying a load of 30 pounds per square foot. The readings of the accelerometer therefore give the loads that the airplane structure must undergo during a maneuver and also the load that the pilot and passengers experience. Every flier knows that he is pushed down into his seat during a tight spiral, for instance, and it is almost impossible to stand up or lift the feet from the floor. During violent stunts a 180-pound man may increase in weight to as much as 800 pounds.

The accelerometer records are of value to the designer, as they show him what stresses the airplane structure undergoes and how long these stresses last. The records also show clearly the pilot's ability, especially in stunts and in landings, so that an accelerometer should be an excellent means of examining a flier, as it gives a clear and unbiased record of his handling of a machine.

DESCRIPTION OF INSTRUMENT.

As the spring scale and weight described above would be undamped and would have such a long period that the shorter vibrations would not be recorded, an instrument working on the same principle was built, but having a much higher period and means for recording the accelerations on a moving film. The instrument consists mainly of a flat steel spring supported rigidly at one end, so that the free end may be deflected by its own weight from its neutral position by any acceleration acting at right angles to the plane of the spring. This deflection is measured by a very light tilting mirror, caused to rotate by the deflection of the spring, and thus reflecting a beam of light onto a moving film, giving an accuracy of about 0.01 g. The essential portions of this instrument are shown in Fig. 1 and photographs of it in Figs. 2 and 3.

The motion of the spring is damped by a thin aluminum vane, which vibrates with the spring between the poles of an electric magnet, and the amount of damping can be varied by altering the current passing through the magnet. The source of light consists of a low-voltage tungsten lamp very similar to a flash-light bulb, the image of its filament being focused after reflection from the mirror onto the moving film. This film is driven at constant speed by a

governor-controlled clock having an electric brake for starting and stopping at a distance. In order to determine the constancy of the clock under accelerations it was mounted on a whirling arm and its rate was measured at several speeds of the whirling arm. In Fig. 4 is shown a curve of speed variation plotted against acceleration. It will be noticed that the point at zero g is shown considerably above the curve, this being due to the fact that the clock was tested on its side, giving zero g along the vertical axis but not along the horizontal axis, as it should be to make all parts weightless. By careful design of the governor it should be possible to keep the speed of the clock constant within 1 per cent under any acceleration that would occur in an airplane. A photograph of the clock is shown in Fig. 5. The natural period of the instrument is 70 vibrations per second, which is high enough to be above any motor vibrations that could occur, and yet a deflection of $\frac{3}{4}$ inch is obtained on the film for an acceleration of 1 g. In order to give a reference line from which to measure accelerations a second mirror is fixed in such a way that it reflects a steady beam of light on the film at the position of zero g.

In order to test the performance of the instrument it was mounted on a whirling arm having a horizontal axis. Upon rotating the arm the accelerations on the instrument changed through a range of 2 g. for each revolution, thus tracing a sine curve, the height of its medium line depend-

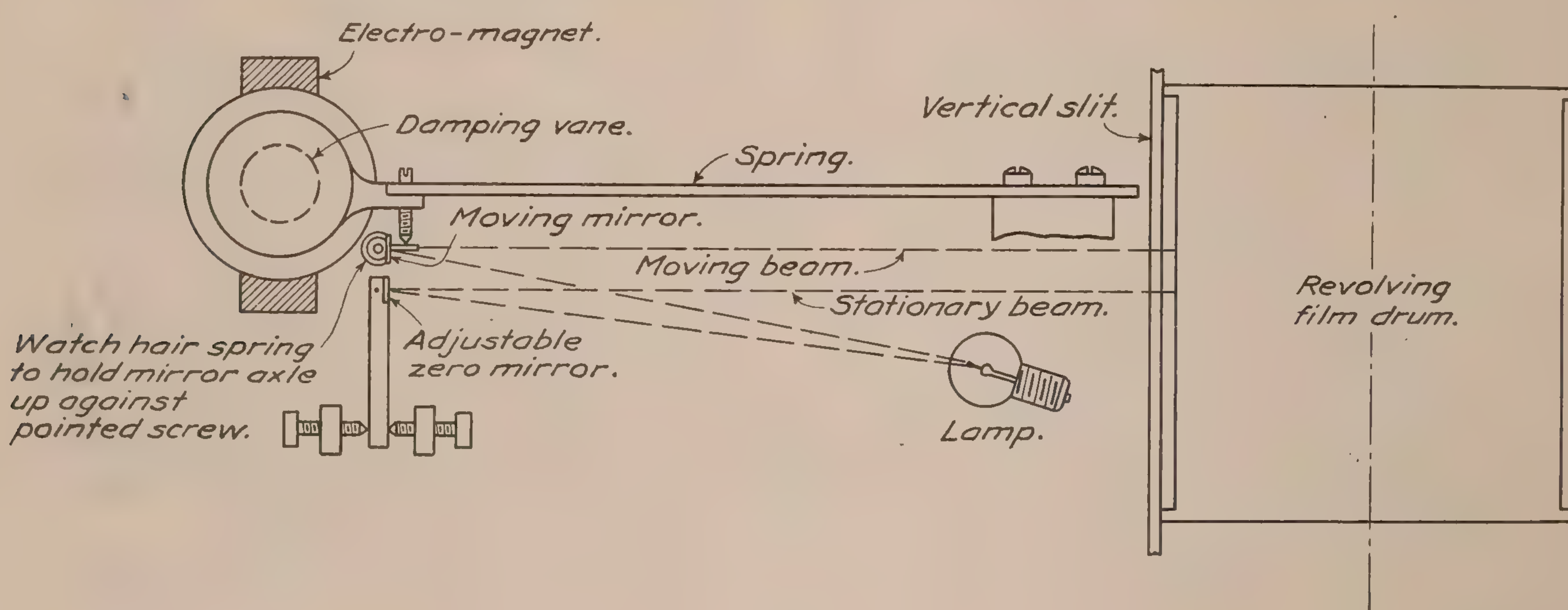

FIG. 1.

ing on the rate at which the whirling arm was revolved. A record taken in this way is shown in Fig. 26. It will be noticed that the curve is very smooth and with practically no instrumental vibrations. In order to test the instrument for vibrations of a high period, a rocking platform was constructed that could be oscillated at any frequency or amplitude by means of cams. The accelerometer was fastened to the table of this oscillator and a record taken as shown in Fig. 28. As one cam had a slight lead on the other there was a sharp knock experienced once every revolution, thus causing excessive vibration in the accelerometer record. The instrument was then mounted on about 1-inch of sponge rubber and another record taken in the same way, except that the film was run at a slightly greater speed, in order to separate the vibrations more clearly. This record is shown in Fig. 29, and it will be seen that the smaller vibrations are almost completely damped out by the rubber.

After the laboratory tests on the instrument were satisfactorily completed, it was mounted in the JN-4H airplane, on a sponge rubber mounting, which isolated it from the vibrations of the fuselage. The instrument was mounted in the front cockpit and was within a few inches of the center of gravity of the machine, and two switches were wired back to the pilot so that he could start the clock or turn on the light at any time. In the other airplanes on which records were taken it was necessary to carry the instrument in the rear cockpit, which is several feet behind the center of gravity, so that the accelerations recorded on these machines were

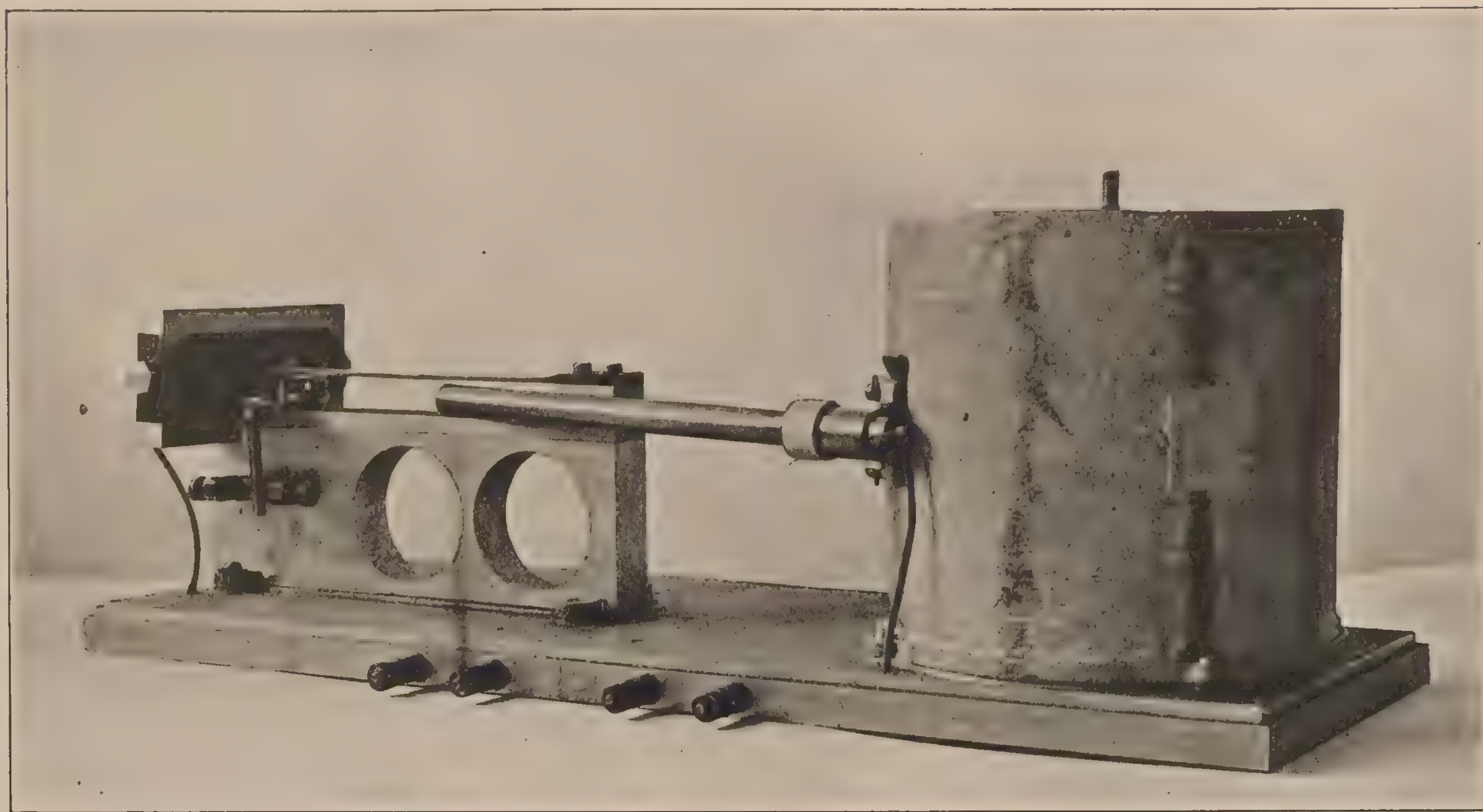

FIG. 2.—N. A. C. A. ACCELEROMETER WITH COVER REMOVED.

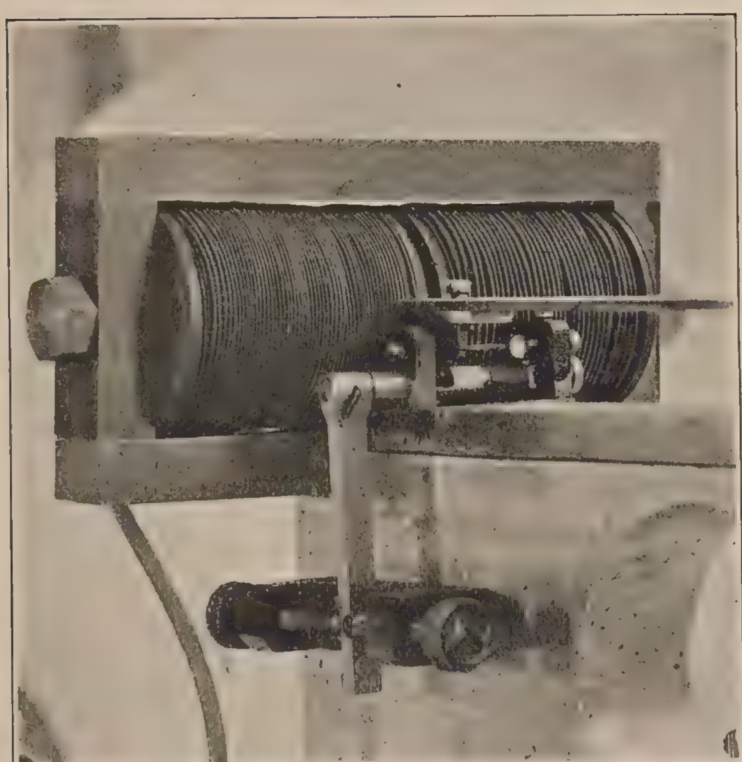

FIG. 3.—DETAIL OF MIRRORS AND DAMPING MAGNETS.

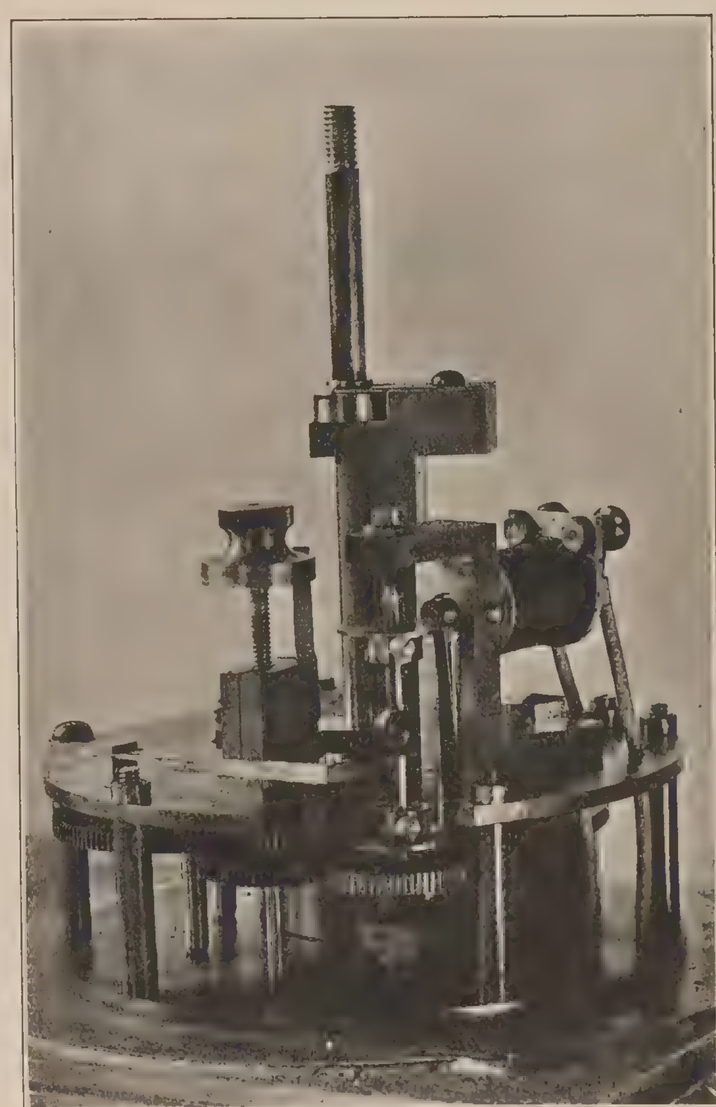

FIG. 5.—ACCELEROMETER CLOCK.

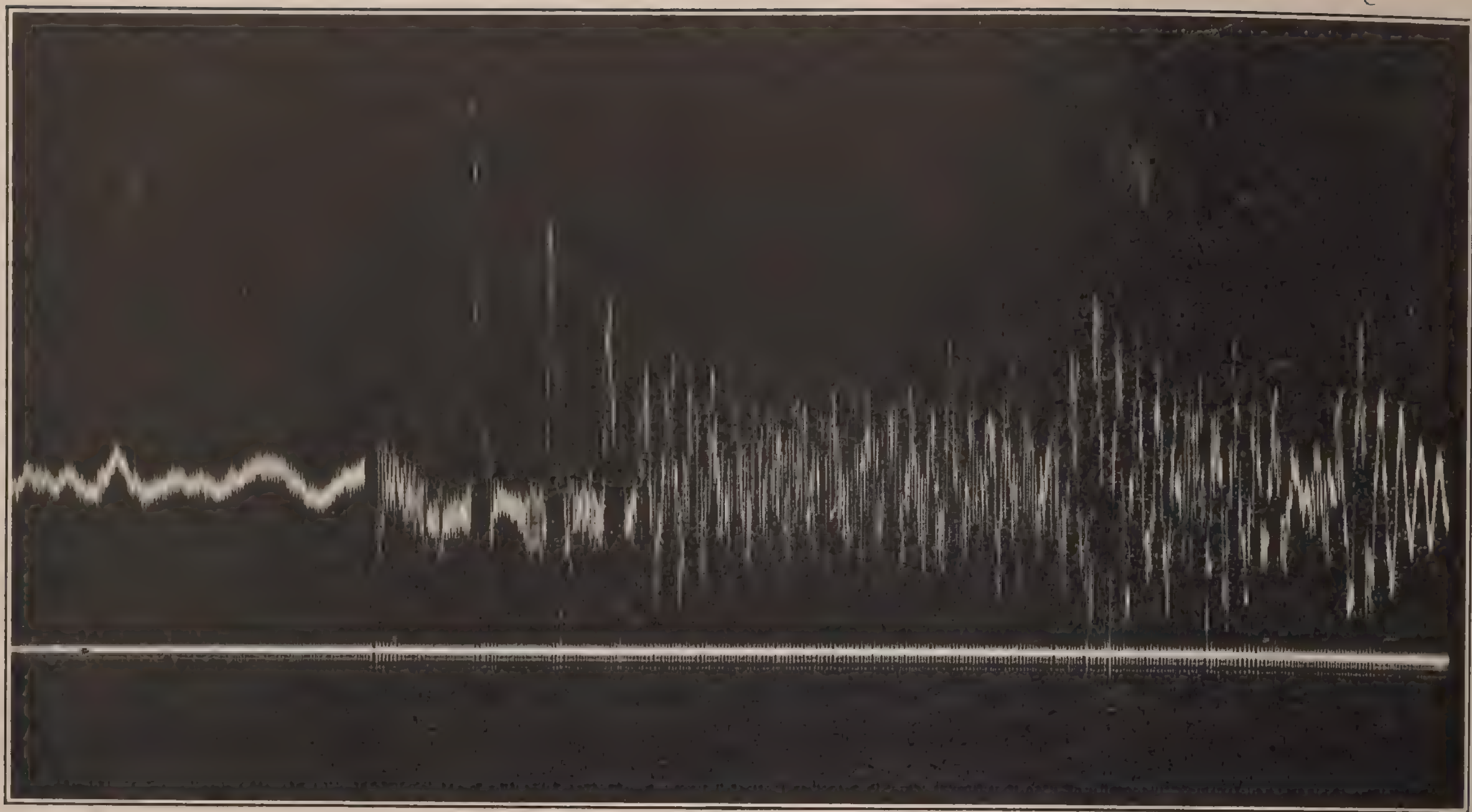

FIG. 6.—LANDING, JN4H, TAIL HIGH AND LITTLE LEVELING OFF. THE PROPELLER WAS NOT REVOLVING. THE MACHINE HAD PRACTICALLY STOPPED ROLLING AT END OF THE RECORD. MAXIMUM ACCELERATION IS 5.25 G.

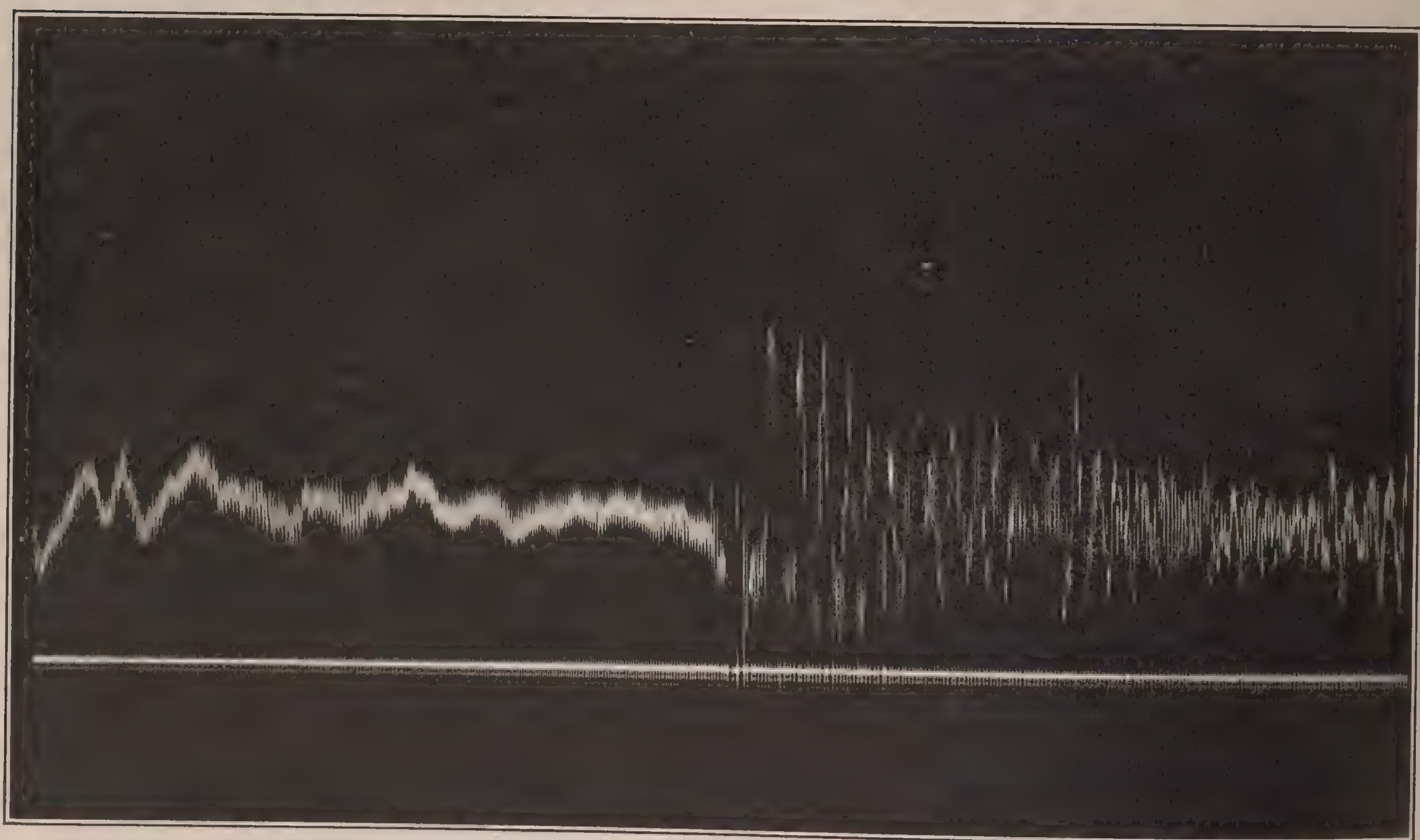

FIG. 7.—LANDING, JN4H, PANCAKED ABOUT 4 FEET. QUITE ROUGH. MAXIMUM ACCELERATION IS 4.95 G.

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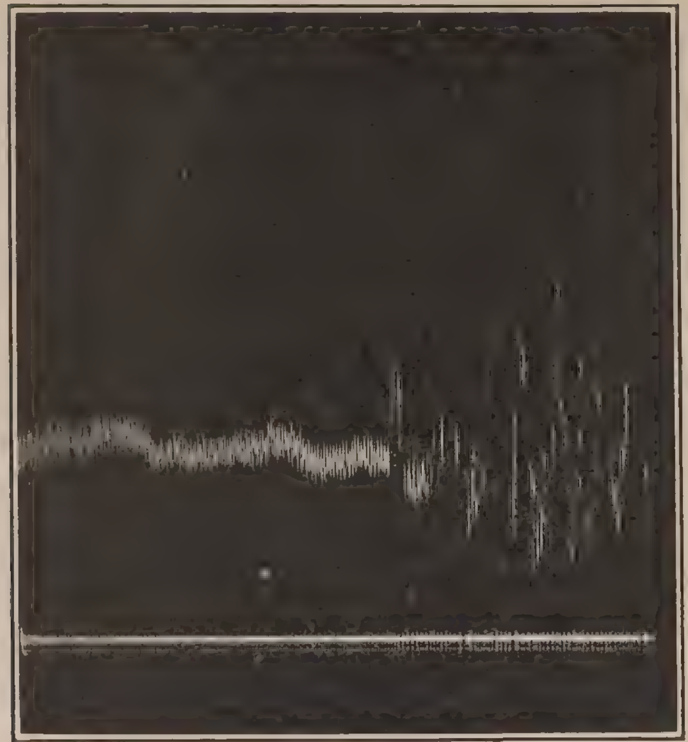

FIG. 8.—LANDING, JN4H, THREE POINT, QUITE SMOOTH. MAXIMUM ACCELERATION IS 2.20 G.

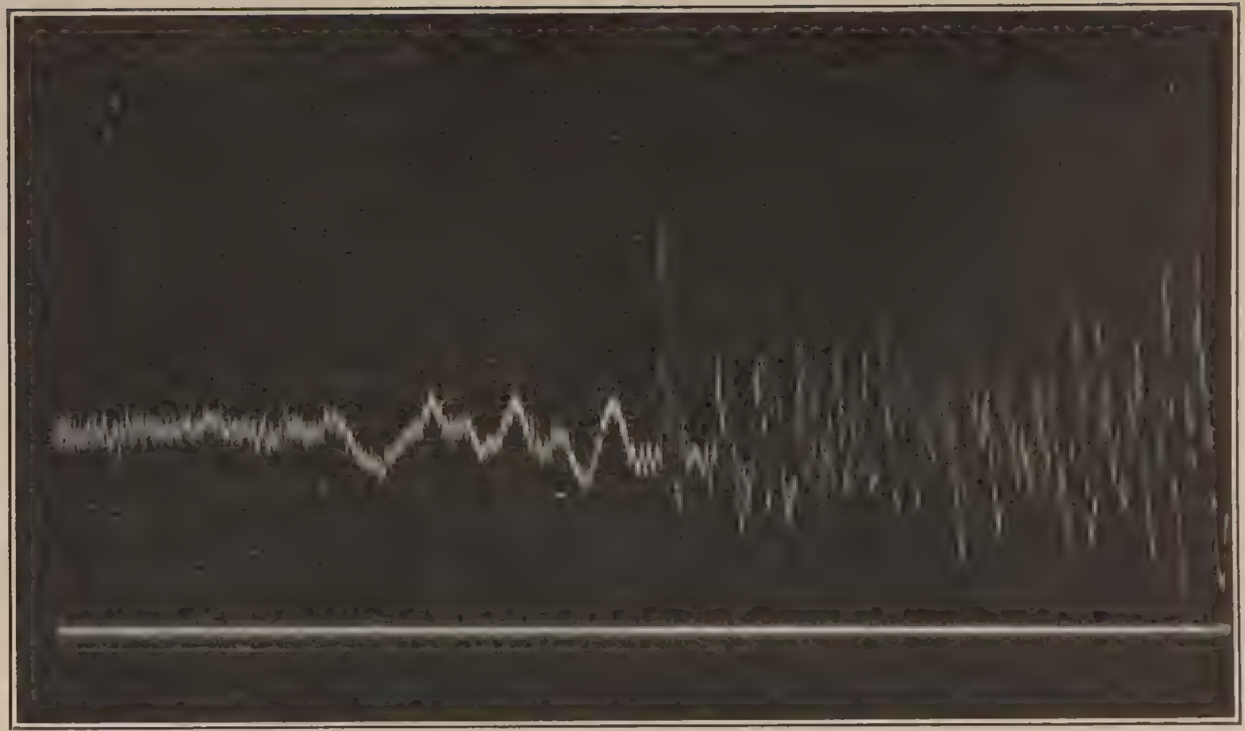

FIG. 9.—LANDING, DH4B, SMOOTH. MAXIMUM ACCELERATION IS 2.56 G.

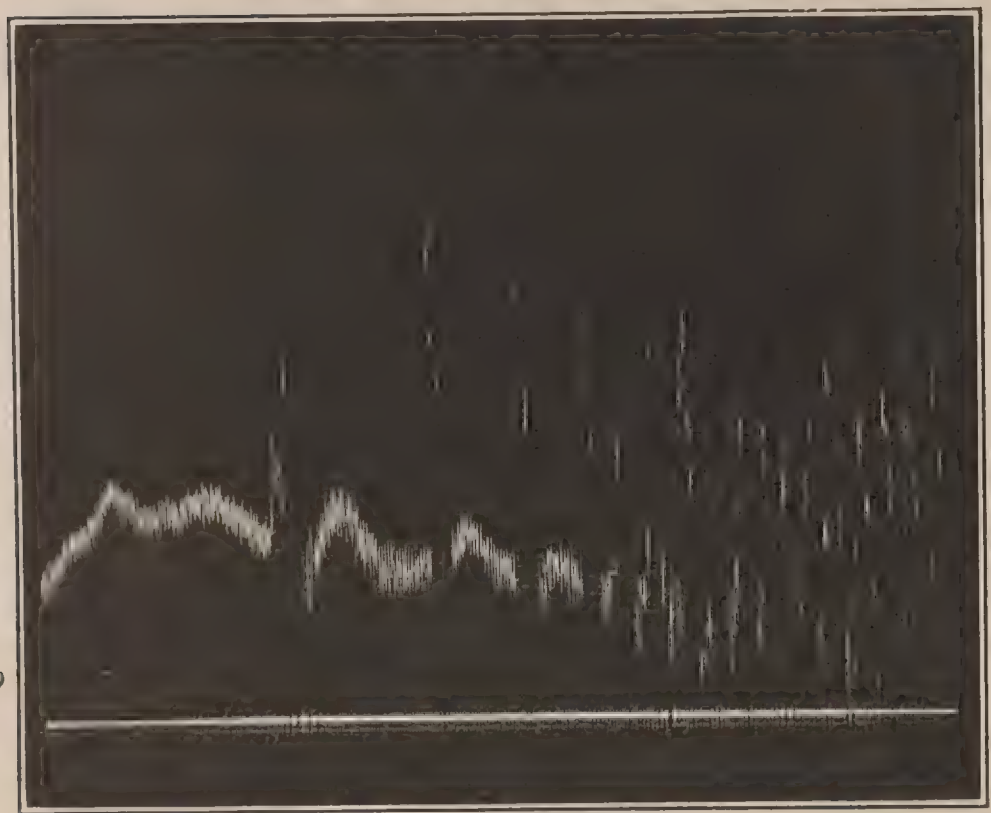

FIG. 10.—LANDING. JN4H, TAIL HIGH, BUT NOT VERY ROUGH. MAXIMUM ACCELERATION IS 3.14 G.

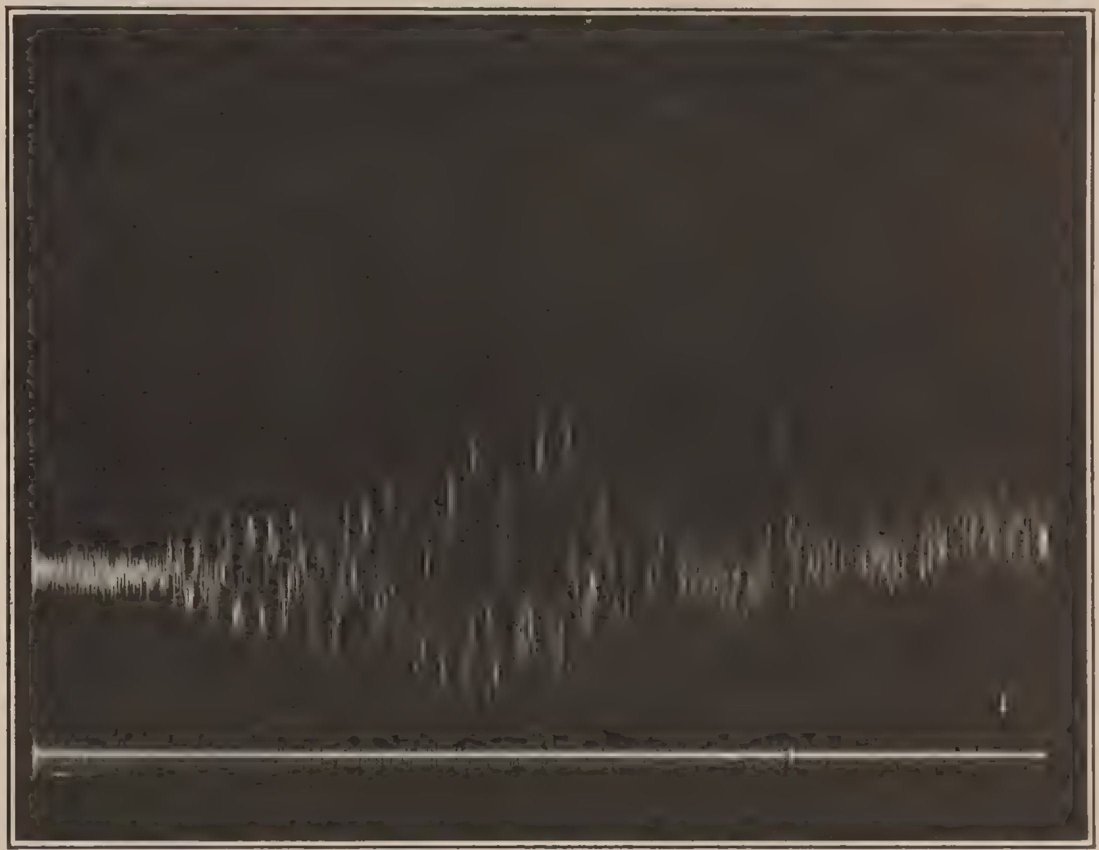

FIG. 11.—TAKE OFF, JN4H. FAIRLY SMOOTH GROUND, LAST BOUNCE MADE PURPOSELY, GIVING AN ACCELERATION OF 3.78 G.

FIG. 12.—LOOP, JN4H. AIR SPEED AT START WAS 72 M. P. H. QUICK PULL UP, STALLED AT THE TOP AND FELL OUT SIDEWAYS. FIRST MAXIMUM IS 3.21 G. AND SECOND IS 2.75 G.

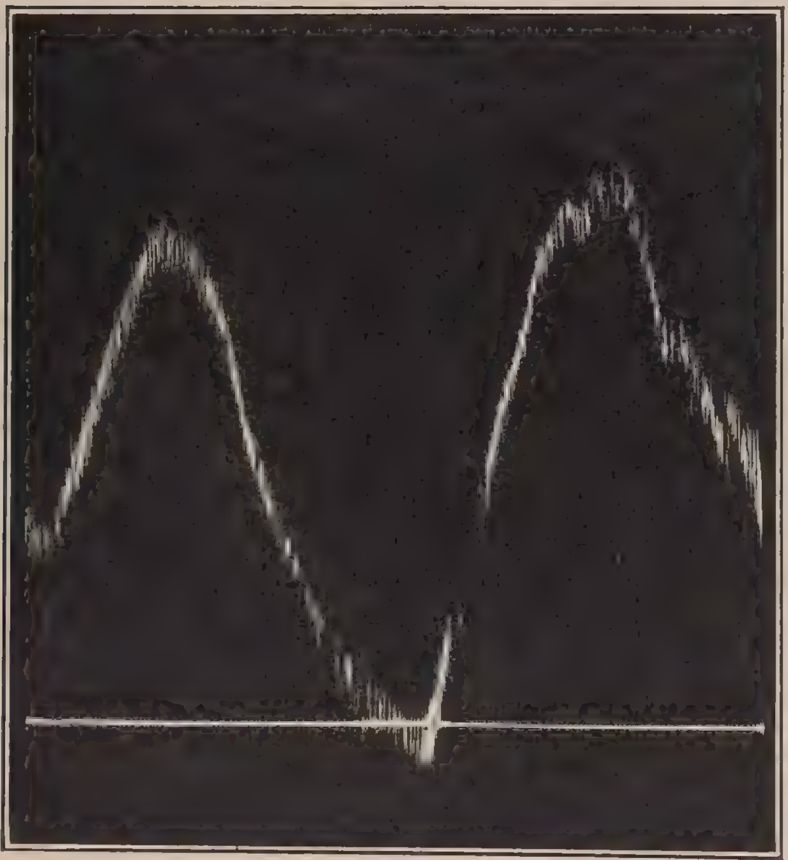

FIG. 13.—LOOP, JN4H. THE AIR SPEED AT START WAS 75 M. P. H., HUNG AT TOP. FIRST MAXIMUM IS 2.85 G. AND SECOND IS 3.22 G.

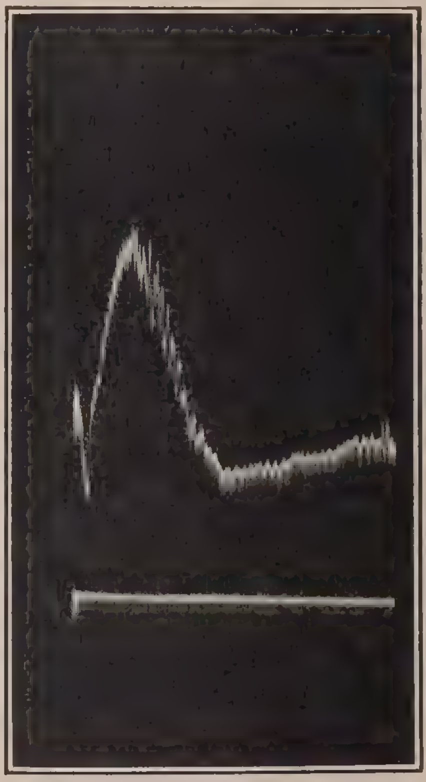

FIG. 14.—STICK PULLED BACK AS SUDDENLY AS POSSIBLE ON THE DH4B AT 75 M. P. H. MAXIMUM ACCELERATION IS 2.05 G.

somewhat in error, due to the effect of angular rotation of the whole machine. The records obtained begin at the left hand end of the film and run toward the right, and the film moves at the rate of 0.215 inch per second, and in all cases an acceleration of 1 g. corresponds to 0.70 inch, measured from the zero line. It will be noticed that even with the shock-absorbing rubber there is transmitted to the instrument a certain definite period of vibration from the fuselage of the plane, and that the period of this vibration is constant for any plane, no matter what the engine speed, but as the amplitude of this vibration is small it is quite easy to estimate a mean line which will represent the true accelerations on the machine as a whole. All accelerations are taken normal to the wing chord.

RECORDS OBTAINED IN FLIGHT.

Landings and take-offs.—In Fig. 6 is shown a record of a landing made in a JN–4H, with the tail high and with little or no leveling off from the glide, and it will be noticed that the first time the wheels struck the ground an acceleration of 5.25 g. was reached, then the machine bounced into the air for about two seconds, struck again with slightly less shock, and then rebounded twice more, each bounce being of shorter period and each shock of less magnitude, until the machine finally rested permanently on the ground. It is also evident that while taxi-ing even on a fairly smooth field, accelerations of as much as 2.5 g. are experienced, and a good landing will always give less acceleration than will the subsequent taxi-ing on anything except the smoothest of fields. In Fig. 7 the same machine was pancaked with a drop of about 4 feet, giving an acceleration at the time of striking of 4.95 g. and the rebound was very slight, and although this landing was intentionally made with a considerable drop it did not feel rougher than many routine landings. Fig. 8 is a record of a smooth three-point landing, where the acceleration on first striking is only 2.2 g., which is considerably less than some of the accelerations experienced shortly afterwards while taxi-ing. Fig. 9 shows a smooth landing in a DH–4B, the acceleration when first touching being only 2.56 g., representing an average landing for this type of machine. In Fig. 10 is shown a tail-high landing in a JN–4H, when the machine porpoised considerably, and three well-marked bounces are evident in the record. The maximum acceleration was 3.14 g. and the landing did not seem exceptionally heavy. Fig. 11 shows a take-off in a JN–4H over fairly smooth ground, and the last bounce was purposely made, which gave an acceleration of 3.78 g. It is quite noticeable that the vibrations of the plane when the motor is wide open on a climb are of greater amplitude, but of the same period, than when the machine is gliding.

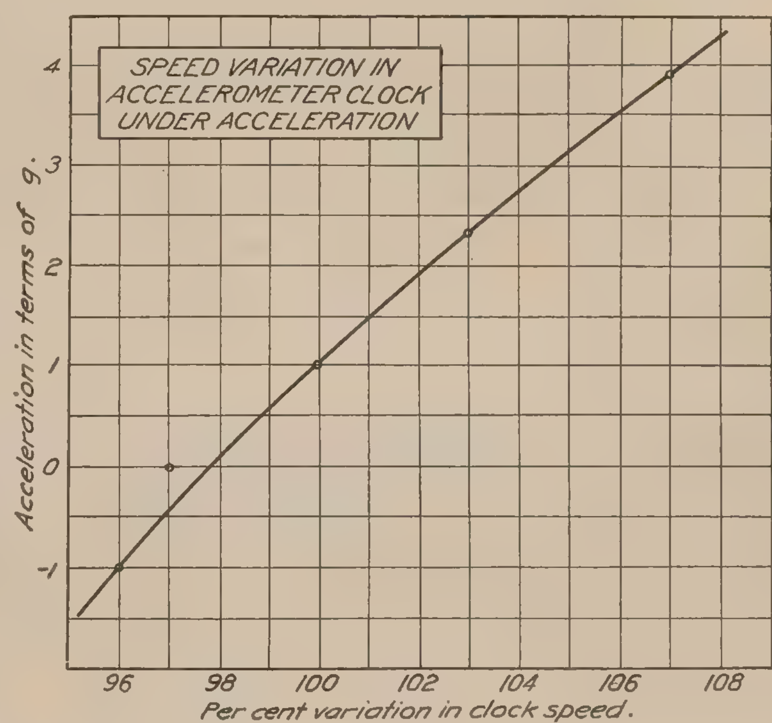

FIG. 4.—Speed variation in accelerometer clock under acceleration.

Looping.—Fig. 12 shows a slow loop with a JN–4H in which the machine fell out at the top, a slow loop usually being characterized by sharp peaks in the acceleration curve. Fig. 13 is another slow loop where the machine hung at the top, but the maximum acceleration on pulling out reached 3.22 g., due to a rather long dive at the end. Fig. 15 shows a loop started at 100 miles per hour, the machine being pulled up slowly and going over very smoothly, and the record approaches very nearly a sine curve except for the small irregularities in the last end of the record, which are believed to be due to the machine entering its own wake, as this fact is often noticed in vertical banks as well as loops. Even with the high initial speed the maximum acceleration was only 3 g. in this loop. Fig. 16 is another loop taken at 92 miles an hour. Fig. 17 is a loop made at 105 miles an hour and shows a very smooth record, although the

machine was pulled around rather rapidly and a maximum acceleration of 3.68 g. was reached. The length of time taken to complete a loop in a JN–4H varied between 14 and 17 seconds, depending not so much on the air speed as on the rate at which the machine was pulled around.

Spins.—Fig. 24 is a spin in a JN–4H, which was stalled into it suddenly in order to set up the maximum amount of oscillation. These oscillations evidently tend to damp out and if the spin was continued long enough would probably disappear. At the beginning of the record the acceleration dropped below 0 g. as the machine fell off from the stall, and after the record that follows very closely a sine curve, but the period of the oscillation is not dependent on the rate of spinning—that is, one oscillation does not necessarily come in one turn of the spin. Another spin on the same machine is shown in Fig. 25, which was a much smoother spin, but the damped oscillations are still evident and an acceleration of 3.12 g. was experienced in pulling out of this spin. The oscillations are practically absent in the spin in a DH–4B (Fig. 22), damping out after the first oscillation. In the same way (Fig. 23) in a record taken of a Bristol fighter there appears to be only one oscillation, and after that the curve is quite smooth and continuous. These records show that there are no large accelerations experienced in the spin itself and unless the machine is dived rapidly in pulling out accelerations should never exceed 3 g. on any type of machine in this maneuver.

Tight Spirals.—A record of a tight spiral is shown in Fig. 21, giving an acceleration of 2.06 g. The accelerations experienced in a tight spiral seem to the pilot greater than they really are because the acceleration is continuous rather than lasting only a few seconds, as in other maneuvers.

Rolls.—In Figs. 18 and 19 is shown the record of a roll of the JN–4H with the motor throttled and with the motor on. The acceleration rises very rapidly to a sharp peak, then falls to about 1 g. and again rises more slowly to a lower peak. The acceleration in the first peak is very high, reaching a value of 4.20 g., which is the highest acceleration experienced in any maneuver. It would seem that rolling would place an exceedingly high stress on the machine. Strange as it may seem the roll is executed with the rudder, and no aileron is used, so that, it is hard to say at just what part of the maneuver the maximum acceleration comes, but it is believed to take place after about 90° of roll. The time for a complete roll on this machine was about 11 seconds.

Top Loading.—In Fig. 20 is shown a record where the stick was pushed forward as rapidly as possible at 70 miles an hour, giving an acceleration of −0.53 g. This maneuver is a very uncomfortable one but it does not seem to place much load on the plane.

SUMMARY OF RECORDS.

The following table gives the maximum acceleration found in each maneuver:

Maneuver.	Machine.	Maximum acceleration.
Porpoise landing	JN–4H	5.25 g.
Pancake, 4-foot drop	JN–4H	4.95 g.
Loop	JN–4H	3.68 g.
Roll	JN–4H	4.20 g.
Spin, maximum in pulling out	JN–4H	3.12 g.
Spin	DH–4B	2.78 g.
Do	Bristol	2.72 g.

From these figures it would seem that in no reasonable stunt would the air load ever exceed 4.5 g. A normal landing should not give more than 3 g., and a very rough landing will seldom exceed 5.5 g. It is quite possible that on a high-speed scout machine, higher loadings than these would be experienced in stunting, but the accelerometer records taken by the Bristol in mock fights show no loads in excess of 4.5 g. An attempt was made to obtain records on an S. E. 5, but the present instrument was so bulky that it had to be attached to the outside of

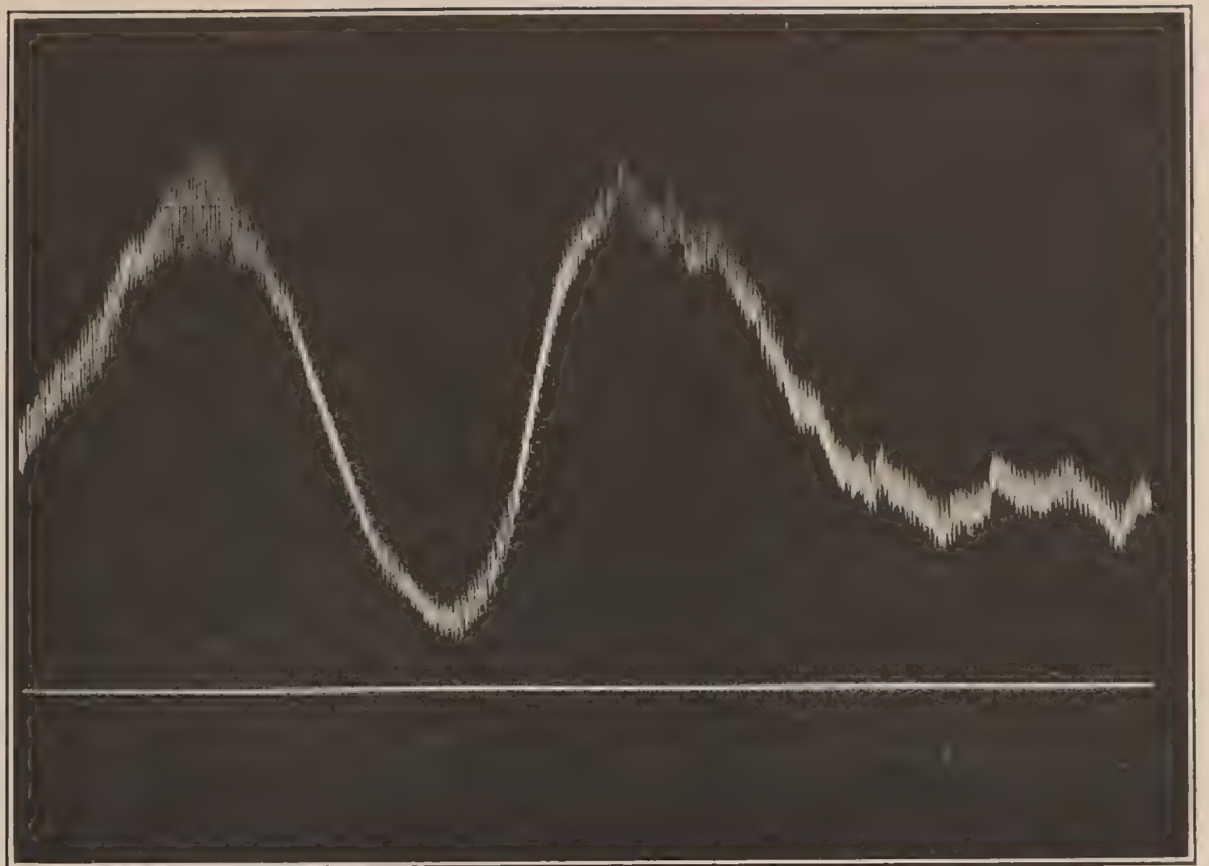

FIG. 15.—LOOP, JN4H. AIR SPEED AT START 100 M. P. H. SLOW, SMOOTH PULL UP NOTICEABLE BUMP WHEN PULLING OUT, DUE TO ENTERING OWN WAKE. FIRST MAXIMUM IS 3.00 G. AND SECOND IS 2.88 G.

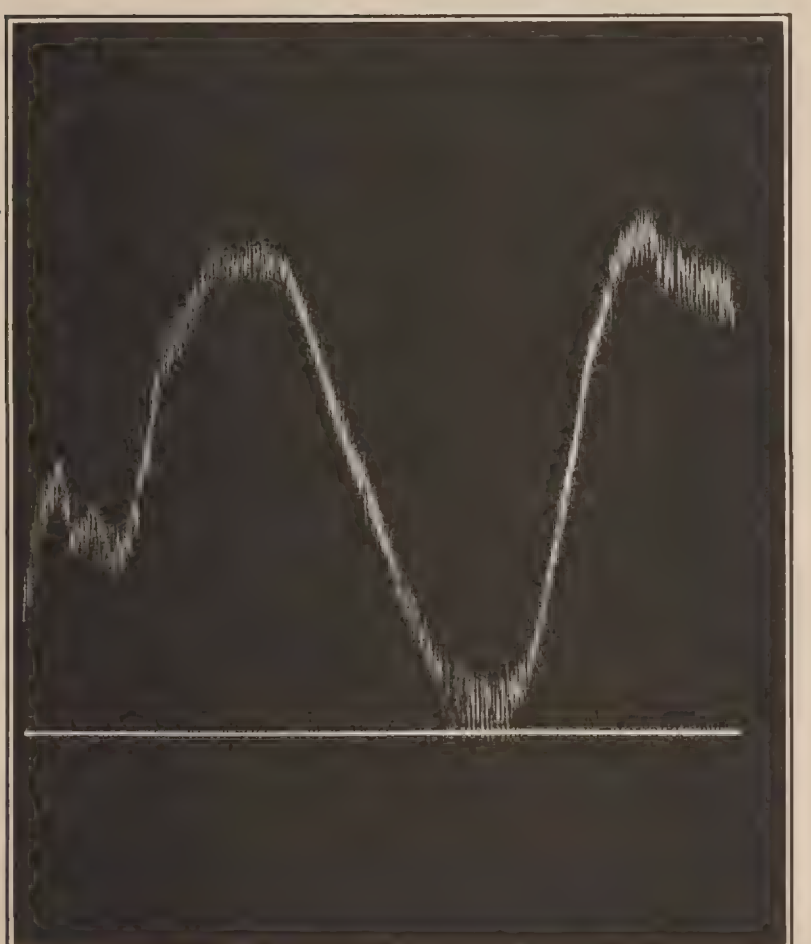

FIG 16.—LOOP, JN4H. AIR SPEED AT START 92 M. P. H. SMOOTH ALL THE WAY AROUND. FIRST MAXIMUM IS 2.85 G. AND SECOND IS 2.97 G.

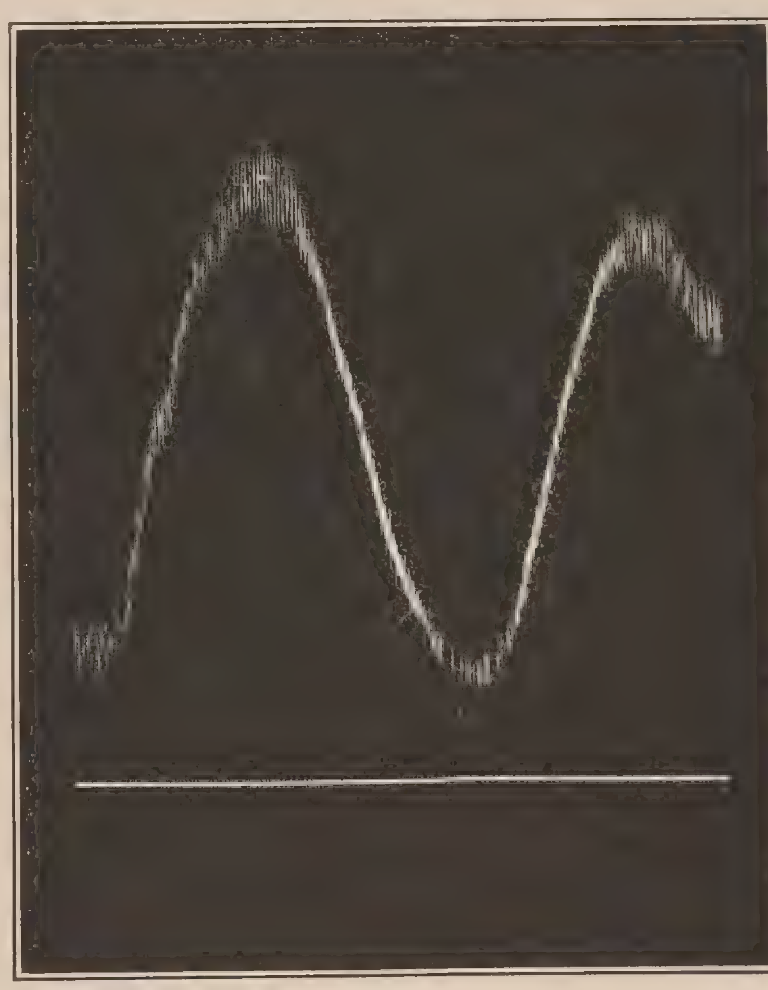

FIG. 17.—LOOP, JN4H. AIR SPEED AT START WAS 105 M. P. H. SLOW PULL UP. FIRST MAXIMUM IS 3.68 G. AND THE SECOND IS 3.26 G.

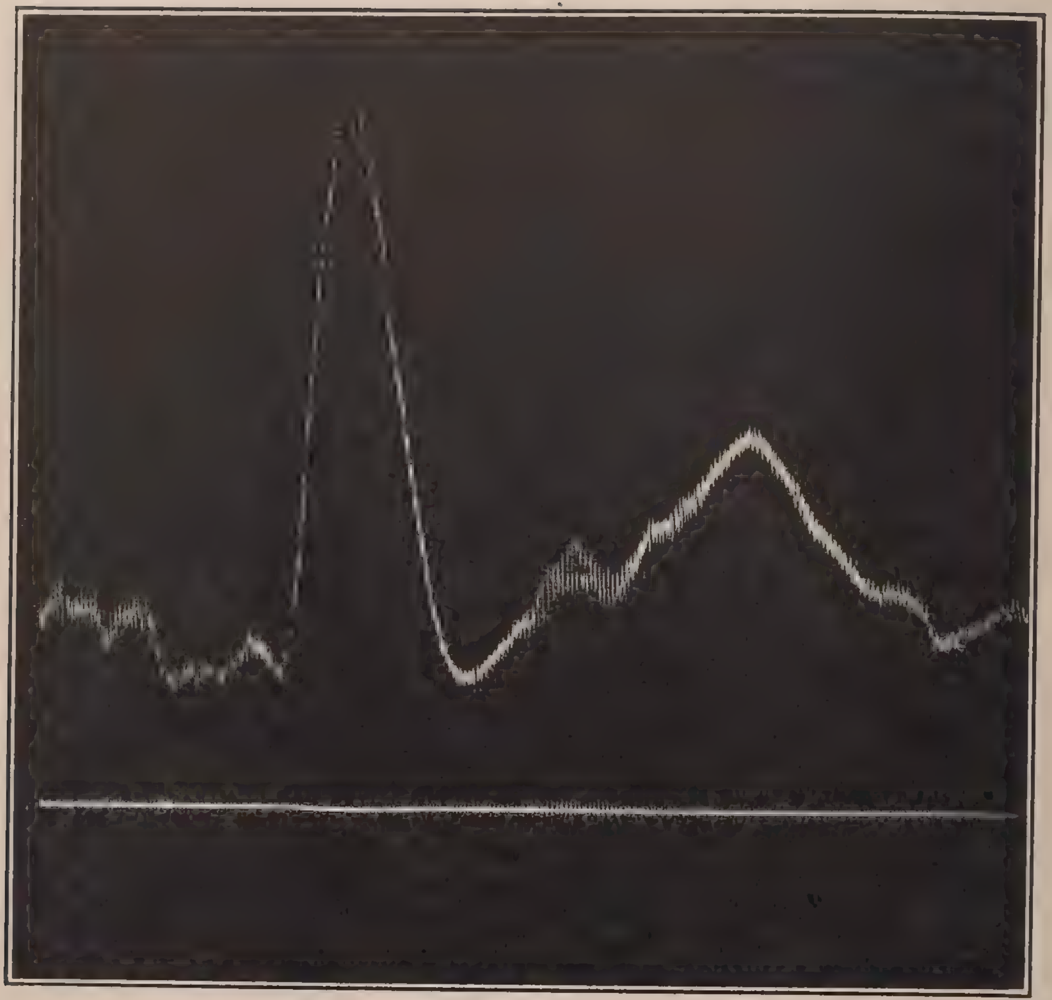

FIG. 18.—ROLL TO LEFT, MOTOR THROTTLED IN JN4H. MAXIMUM ACCELERATION IS 4.20 G., THE HIGHEST FOUND IN ANY MANEUVER. AIR SPEED AT BEGINNING WAS 100 M. P. H.

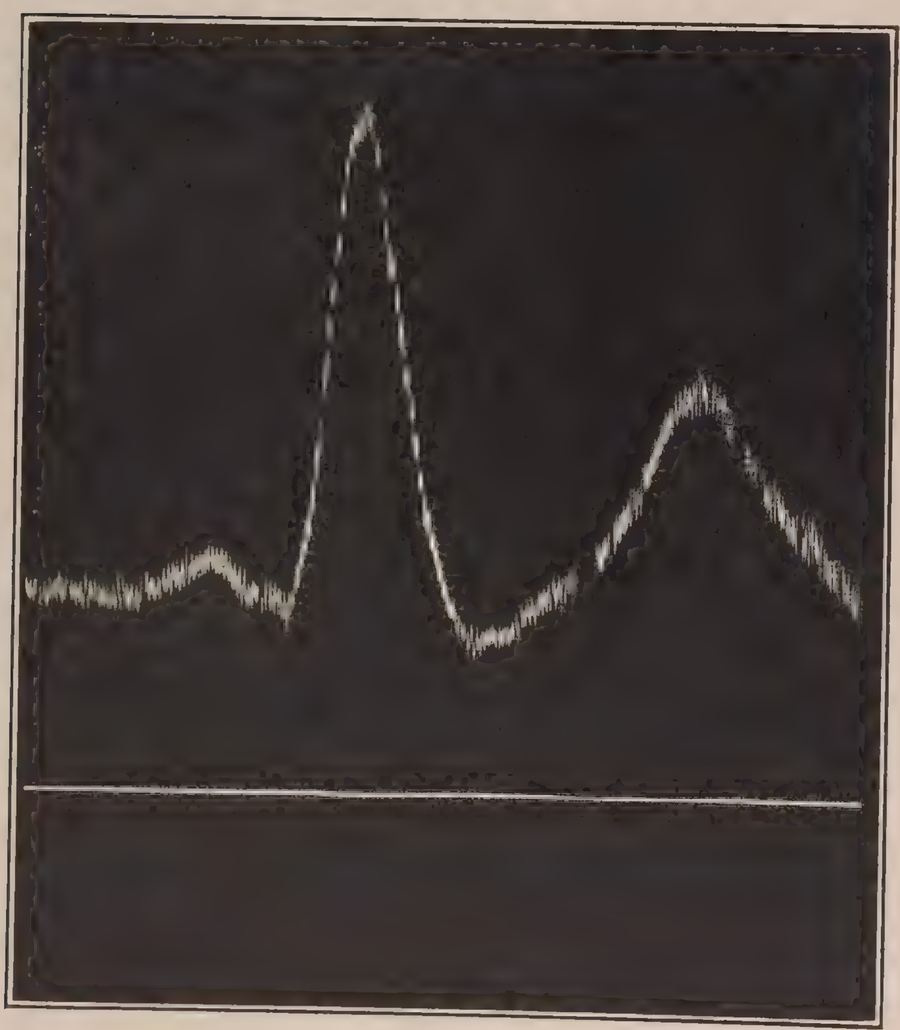

FIG. 19.—ROLL IN JN4H, TO THE LEFT. MOTOR ON. MAXIMUM ACCELERATION IS 4.15 G. AIR SPEED AT BEGINNING WAS 100 M. P. H.

FIG. 20.—TOP LOADING JN4H, STICK PUSHED FORWARD AS QUICKLY AS POSSIBLE AT 70 M. P. H. MAXIMUM NEGATIVE ACCELERATION IS −0.53 G.

FIG. 21.—TIGHT SPIRAL, MOTOR ON JN4H. MAXIMUM ACCELERATION IS 2.06 G.

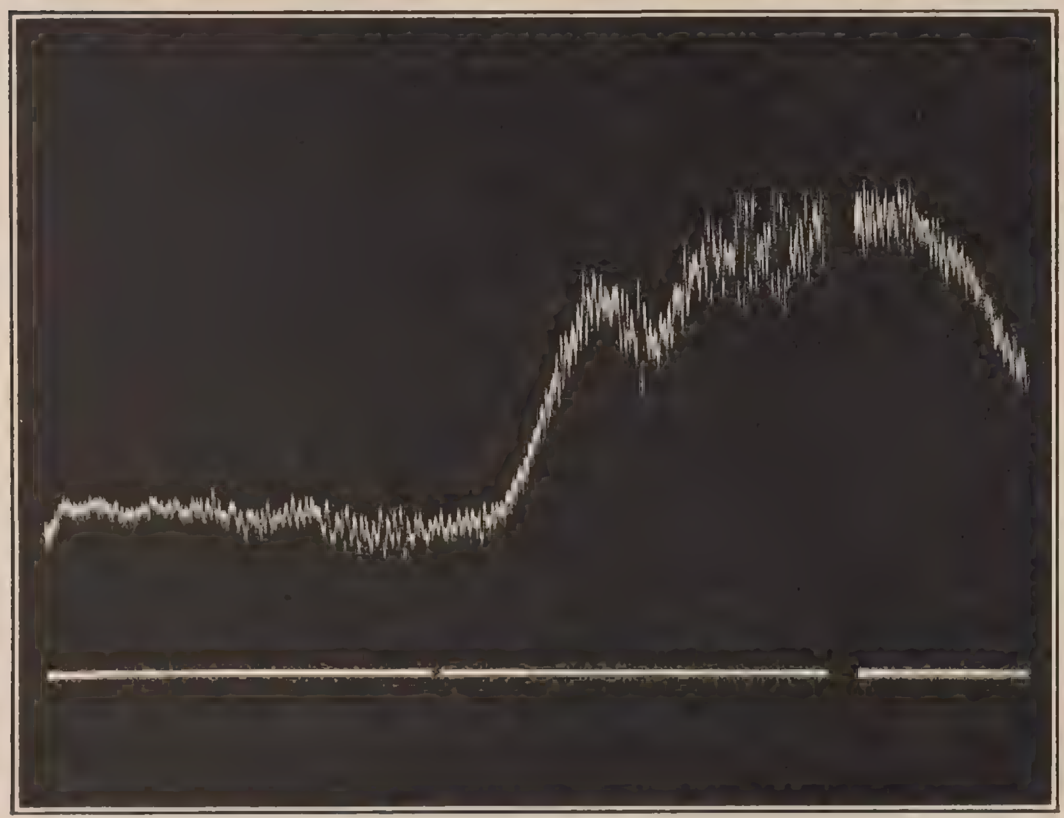

FIG. 22.—SPIN, DH4B, SMOOTH. MAXIMUM ACCELERATION IS 2.78 G. AS THE SPIN WAS A LONG ONE, THE FILM WAS STOPPED FOR A SHORT TIME IN THE MIDDLE OF THE SPIN.

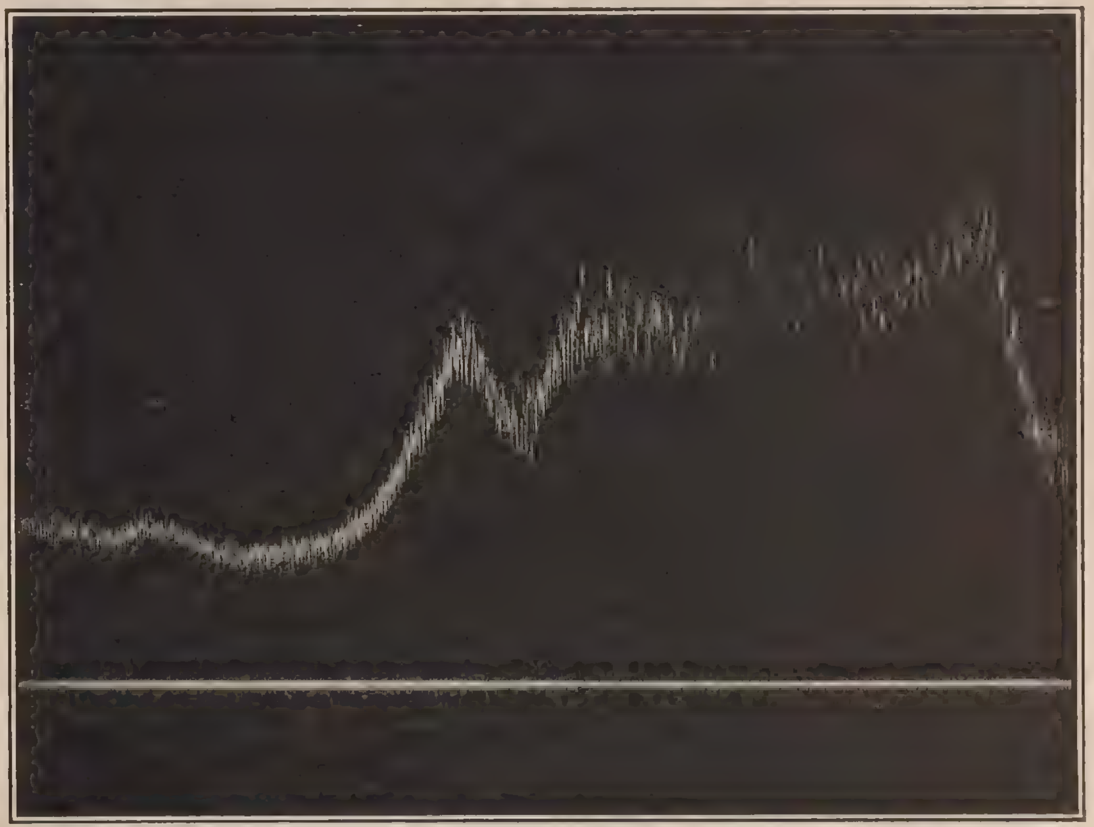

FIG. 23.—SPIN, BRISTOL FIGHTER (ENG.) MOTOR THROTTLED. THE MACHINE CAME OUT OF THE SPIN ITSELF. MAXIMUM ACCELERATION IS 2.72 G.

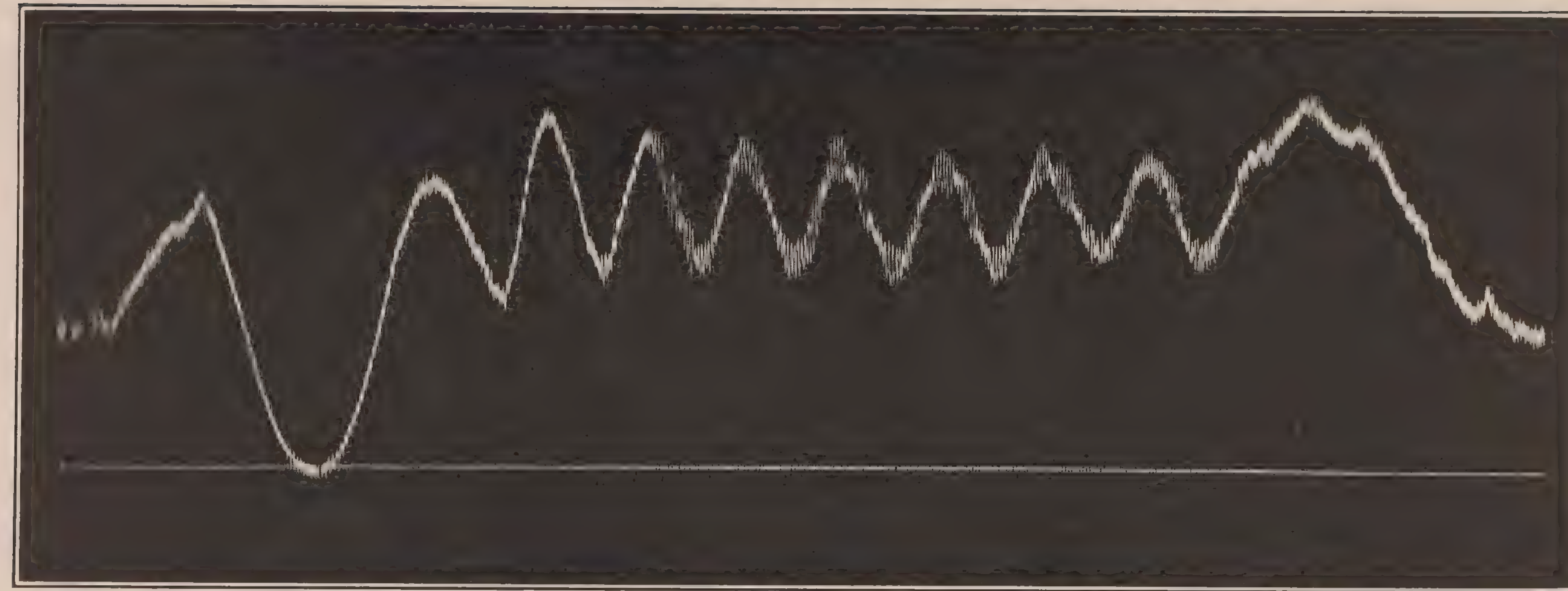

FIG. 24.—SPIN OF 9½ TURNS, JN4H. QUICK PULL UP INTO STALL, MOTOR THROTTLED. DESCENDED 1,700 FEET. MAXIMUM ACCELERATION IN THE SPIN WAS 2.57 G. AND WHEN PULLING OUT 2.70 G.

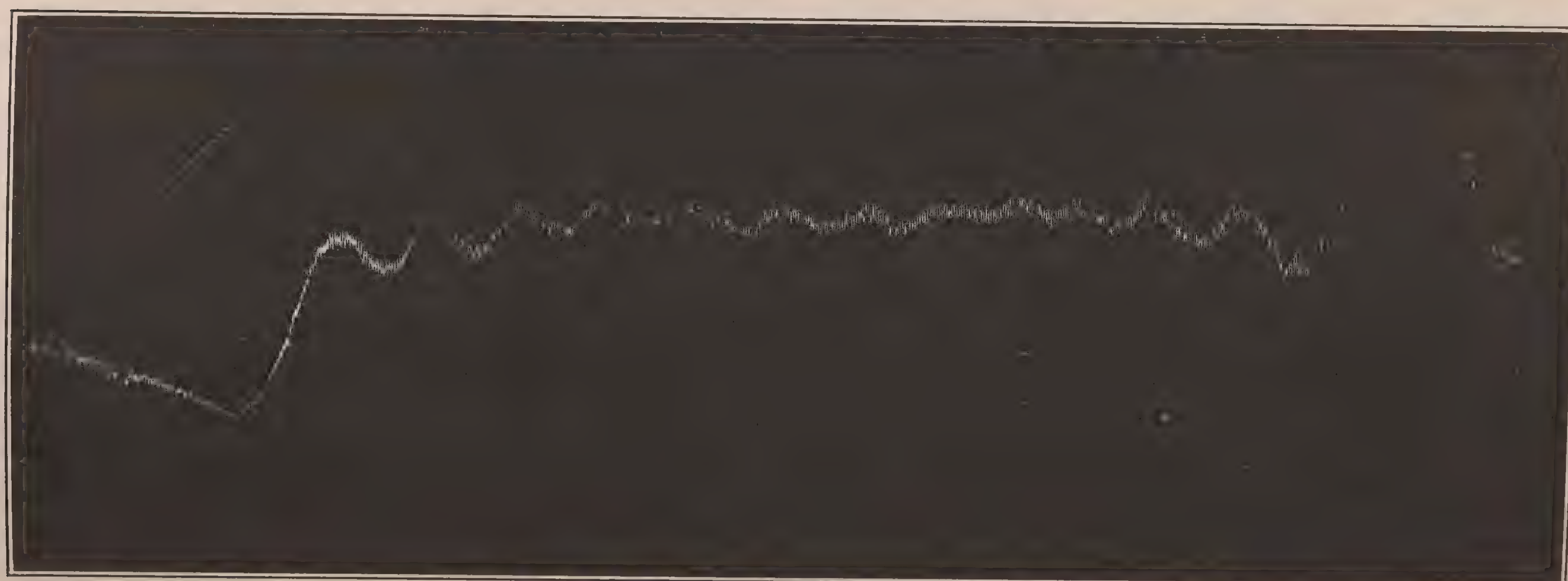

FIG. 25.—SPIN OF 7 TURNS, SMOOTH, JN4H. MAXIMUM ACCELERATION IN THE SPIN IS 2.21 G. AND IN PULLING OUT 3.12 G.

the fuselage, and although it was partly protected from the propeller blast, the vibrations set up were so great as to obscure the record. A much smaller instrument of the same type is now being constructed and it is hoped to get records on a machine of this type in the near future.

ACCELERATIONS IN PULLING OUT OF A DIVE.

In Figs. 30 to 33 are shown the records obtained when pulling back as suddenly as possible on the stick of a JN–4H at 50, 60, 70, and 80 miles per hour. The records show that the same time elapses for reaching the maximum acceleration regardless of the speed of the plane, and that this maximum is reached approximately 0.85 second after the stick is pulled back. Accelerations obtained from these records are plotted in Fig. 34 against the air speed of the plane, and to check them up a second set of runs were taken under the same conditions and their values plotted on the same curve, and the coincidence of the points is excellent. On the same sheet is also a curve of theoretical accelerations that would be experienced if the plane was instantly turned to its angle of maximum lift without losing any speed. This theoretical curve, as would be expected, lies slightly above the experimental curve, but the difference between the curves is much less than would be supposed, so that the stresses determined from theoretical considerations should give very closely the true accelerations when the controls are suddenly pulled back at any speed. By extending the experimental curve to higher speeds, which can certainly be done with very little error, it is possible to estimate the exact loading experienced in pulling out of a dive at any speed. It was not thought advisable to carry the experimental points up to any higher speeds with the machine at hand although it probably would have stood a loading as high as 8.

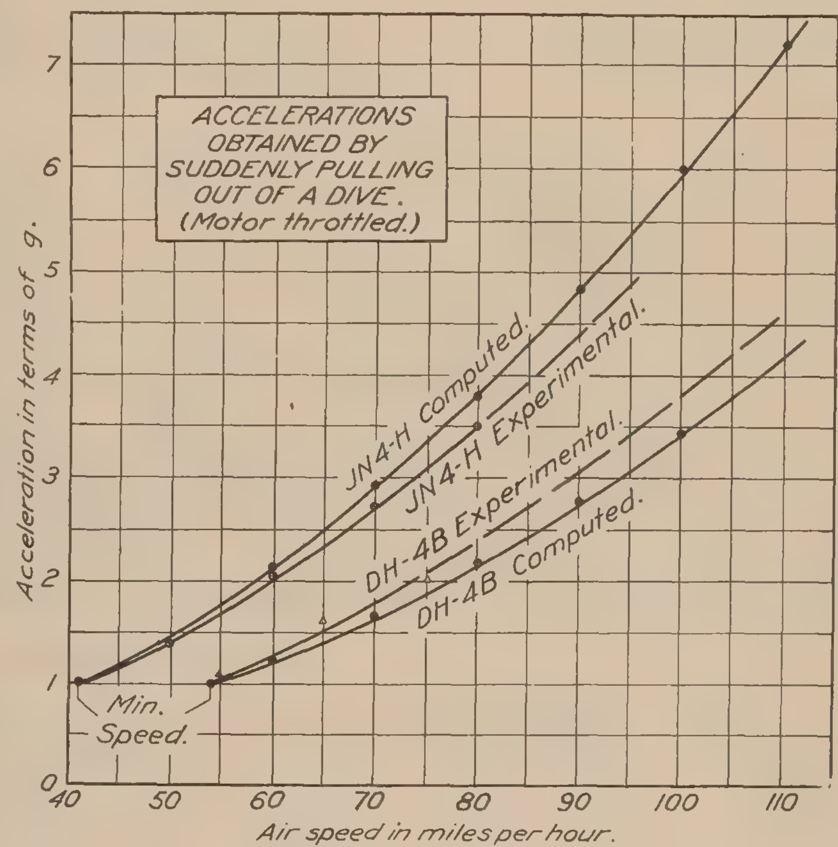

FIG. 34.—Accelerations obtained by suddenly pulling out of a dive (motor throttled).

A similar set of readings was taken in a DH–4B and one record is shown in Fig. 14, but as it was necessary to place the instrument at a considerable distance behind the center of gravity there was a considerable negative acceleration at the instant of pulling back the stick due to the angular acceleration of the whole machine. For this reason it is probable that the maximum acceleration obtained is not correct. The points, however, are plotted in Fig. 34 along with the theoretically determined curve, and the points lie slightly above the theoretical curve instead of below it as they should. This may be due to a wrong estimate of the minimum speed of the plane, as no careful performance tests have been made with this plane as they have with the JN–4H, or it may be due to an error caused by the angular acceleration of the machine. At any rate these tests seem to show conclusively that the loads determined in pulling out of a dive at any speed, from the assumption that the machine is instantly turned up to its maximum angle before losing any forward speed, will give the loads very closely to those actually experienced in flight.

Assuming that the limiting velocity of the JN–4H is 150 M. P. H., the theoretical maximum load in pulling out suddenly would be 14 g., but from the exterpolated experimental curve it would be only 12.5 g., and even this figure would be too high, as the pilot would relax his pressure on the stick to a considerable extent before reaching this acceleration.

If the designer could be assured that a machine would never experience an acceleration above a certain fixed limit, he could design his machine so that it would not carry any excess weight in the structure, thus greatly increasing the machine's efficiency. It would seem possible to make a device consisting principally of a weight on a spring that would nose the machine over whenever it underwent more than a certain acceleration. In other words, it would act as a safety valve for the airplane structure.

MACHINE VIBRATIONS.

It is noticeable from the records that whenever the wheels are free from the ground the machine has a certain vibration. It would naturally be supposed that this vibration is wholly due to the motor, and would vary in period as the motor speed is changed. A careful examination of the records shows, however, that the vibration for a given machine is nearly constant in period no matter what the motor speed. In fact, on one occasion the machine was glided for a considerable time with a dead stick and the vibrations persisted in exactly the same way as with the motor on Fig. 6. The vibrations are, however, of somewhat greater magnitude when the motor is full open as shown at the end of Fig. 11.

The vibration periods of the three machines tested are approximately as follows:

Type of machine.	Approximate weight (pounds).	Frequency of vibration.
Curtiss JN-4H	2,000	17.0
DH-4B	3,600	8.6
Bristol Fighter (English)	3,000	10.2

The period is then nearly inversely proportional to the weight of the machine. The vibrations recorded apparently are natural vibrations of the airplane structure, being excited either by passing through the air or by the motor. These vibrations are so much slower than the fundamental period of the instrument that they can not be accounted for in that way.

CONCLUSIONS.

These experiments show that a true factor of safety in the air (exclusive of the material factor) of 4.5 is sufficient for any stunting that would ordinarily occur in flight, and this value could only be exceeded by a deliberate attempt to break the machine. The load factor in as rough a landing as there is any excuse in making should not exceed 5, but no definite rule can be laid down, the landing loads being dependent on the condition under which the machine ia used. The accelerometer record is an excellent indication of the pilot's skill and should find extensive use in examining fliers. Smaller and lighter instruments are now being built, and accelerations will be recorded along all three axes of the airplane, as well as the angular accelerations about the center of gravity. The theory of the accelerometer will be taken up in a subsequent report.

REPORT No. 100

ACCELEROMETER DESIGN

By F. H. NORTON and EDWARD P. WARNER

Langley Memorial Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Field, Va.

NOTE.—Report prepared for the Bureau of Construction and Repair, Navy Department, and published by permission of the Chief Constructor, U. S. Navy.

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ACCELEROMETER DESIGN.

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INTRODUCTION.

To carry out the work on accelerometry for the Bureau of Construction and Repair, Navy Department, the first step necessary was to study all previous types of accelerometers, with the result that an instrument was developed by the technical staff of the National Advisory Committee for Aeronautics at the Langley Memorial Aeronautical Laboratory for the purpose of recording more accurately than had been done before the accelerations experienced in flight. The errors due to accelerations acting in other than the required direction and the errors due to angular accelerations were studied and as far as possible eliminated. The response of the instrument to shocks of short duration and the damping of the free vibration were analyzed mathematically, as well as the possibility of resonance with the vibration of the plane or engine. The results of this work are included in the present report, together with a description of the actual instrument.

THE PROPERTIES DESIRED IN AN ACCELEROMETER.

The ideal accelerometer should have a natural period very high compared with that of any shocks that it could experience. It should have a large enough deflection to be read with an error of not more than 0.1 per cent of the maximum acceleration, and it should be so damped that it will follow the actual acceleration in the closest manner. It may be contended that such accuracy is unnecessary, as any given maneuver can not be duplicated within several per cent, and the engineer does not need loads to better than 5 per cent. On the other hand, when it is desired to obtain the difference of accelerations in various parts of the airplane in order to calculate the rotary motions, an accuracy of 0.1 per cent is none too great. The accelerometer should only record linear accelerations along one axis, and should be unaffected by any other accelerations. Besides these qualities it should have compactness, ruggedness, and be simple to operate, and the record should be clear and strong and easy to reproduce.

PREVIOUS TYPES OF INSTRUMENTS.

A type of accelerometer developed by Dr. Zahm[1] consists of a number of styluses held against stops a short distance above a moving strip of paper, by springs of different tensions, as shown in figure 1. Each spring is adjusted so that its stylus is brought into contact with the paper whenever the acceleration exceeds a certain amount, so that by having a number of graded springs a curve of acceleration can be traced, as shown in figure 2. This type of instrument will not trace a continuous curve, and it is not practical to have a sufficient number of springs to trace the small and rapid vibrations; but, on the other hand, it has no lag, no natural period, and none of the errors inherent in the free spring instrument. This type of instrument is very valuable for studying the maximum values of landing shocks, but will not, of course, give a continuous curve of acceleration against time.

The R. A. F. accelerometer consists of a semicircular quartz fiber, illuminated by a small incandescent lamp. The deflection of this illuminated fiber is magnified and projected on to a moving film, as shown in figure 3, thus giving a curve of acceleration against time.[2] The natural

[1] Development of an Airplane Shock Recorder, by A. F. Zahm: Jnl. Franklin Inst., August, 1919.

[2] R. and M. No. 376 of British Advisory Committee for Aeronautics, September, 1917.

period of the fiber is about one-twentieth second, and the damping is solely by air friction. This instrument gives a continuous record, but its period and damping are low and the deflection on the record is small, so that it can not be read to better than 0.1 g. Also, as will be shown later, a rather serious error is introduced by components of acceleration acting at other than normal to the plane of the fiber. However, the instrument is very compact and simple to operate, and has been used very successfully to determine the loads on airplanes in flight, but is unsatisfactory for landing shocks.

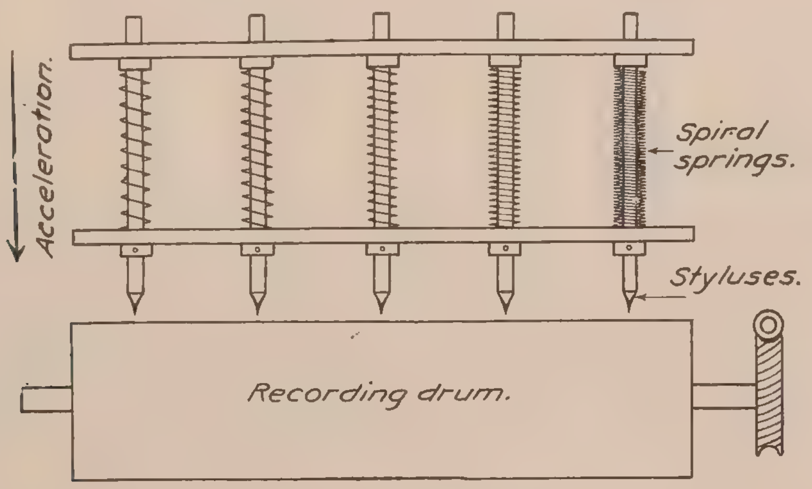

Fig. 1. ZAHM ACCELEROMETER

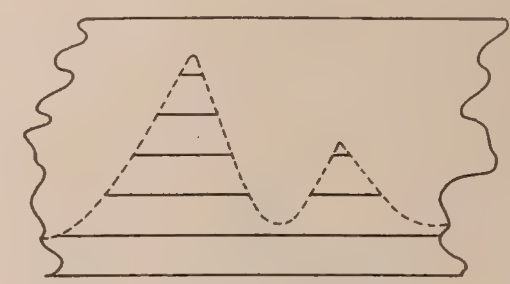

Fig. 2 RECORD FROM ZAHM ACCELEROMETER

A type of accelerometer[3] designed on the principle of a seismograph has been used to determine the properties of automobile springs. This instrument (fig. 4) records displacements so that the curve must be differentiated twice in order to obtain accelerations. This is an inaccurate and laborious method of obtaining accelerations, and it could obviously not be used for anything except short periods. The instrument is of value, however, for obtaining the period and amplitude of small high frequency vibrations.

The National Advisory Committee for Aeronautics' accelerometer consists of a flat cantilever spring, the deflection of whose end rotates a small mirror, thus reflecting a beam of light on to a moving film, as shown diagrammatically in figure 5. A more complete description of this instrument will be given later in this report.

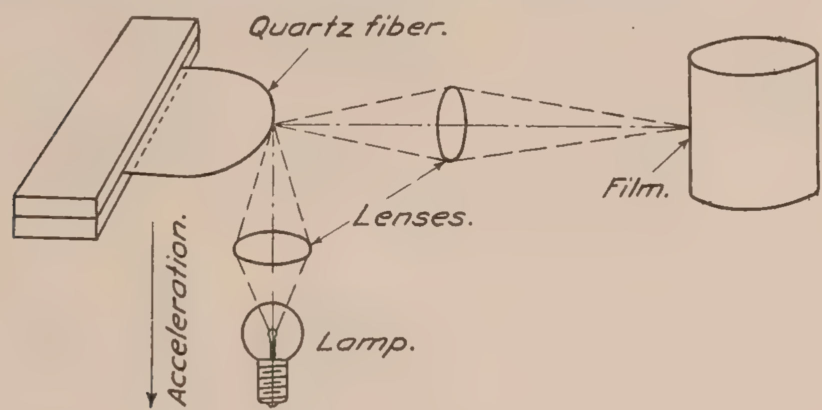

Fig. 3. R. A. F. ACCELEROMETER

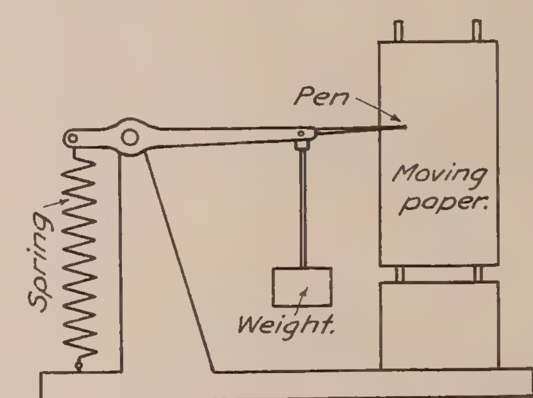

Fig. 4. SEISMOGRAPH TYPE OF ACCELEROMETER

DEFLECTIONS AND NATURAL PERIODS OF SPRINGS.

It is obvious that for an accelerometer the greatest deflection with the highest natural period is desired. Unfortunately the deflection and period of a uniformally loaded flat spring under static load, are connected by the formula:

$$D = K\,T^2$$

where D = deflection

T = time for complete oscillation

K = a constant varying slightly with the type of spring.

[3] The Journal of the Society of Automobile Engineers, January, 1920, p. 17.

This shows that to double the frequency it is necessary to quarter the deflection, so that it is obvious that the lowest frequency should be chosen consistent with accurately following the highest period accelerations to be encountered.

The worst landing shocks on the JN4H rise from zero to a maximum in 0.23 second, and on a DH4 in 0.37 second. On seaplanes the landing shocks will be sharper, but will probably not rise to a maximum in less than 0.02 of a second. On land airplanes the deceleration in landing obviously can not reach its maximum value until the shock absorbers are fully extended. The natural period of the airplane in flight may reach, on light machines, a frequency of 25 vibrations per second, so that in order to avoid resonance the natural frequency should be well above or below this figure. From these figures it seems that a natural frequency of 50 vibrations per second will be ample except for seaplane landing shocks, which may require a frequency of 100, but that the natural period of 20 in the R. A. F. instrument is in some cases too low.

If a frequency of 50 is assumed, the deflection under an acceleration of 1 g. will be in the neighborhood of 0.005 inch—an amount much too small for direct recording. This motion may be directly magnified as in the R. A. F. instrument; but as the object is magnified in the same amount as the motion it does not pay to use a large multiplication and therefore the record is small. In order to magnify this deflection in the N. A. C. A. instrument, a relatively heavy spring is used, so that the mass and friction of the rocking mirror will have no

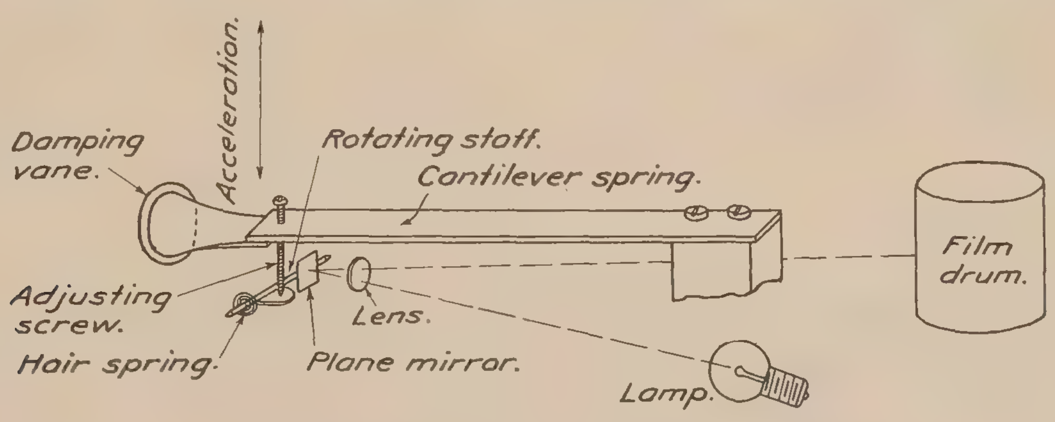

Fig. 5. DIAGRAM OF N. A. C. A. ACCELEROMETER

appreciable effect on the properties of the spring. In this manner the motion of the spring can be easily multiplied 400 times.

In an instrument used in airplanes it is desirable to use as compact a spring as possible; that is, the shortest spring for a given deflection. This leads to the use of a cantilever type of spring, the static deflection of which under gravity is determined by the load distribution, usually a combination of distributed and concentrated loads. The deflection due to weight of the spring is

$$f=\frac{W_1L^3}{8\,EI}$$

and the deflection due a concentrated load at the end is

$$f=\frac{W_2L^3}{3\,EI}$$

where f = the static deflection.

W_1 = the total weight of overhanging portion of spring.

W_2 = concentrated weight at the end of spring.

L = free length of spring.

E = modulus of elasticity of the spring.

I = moment of inertia spring section.

The total deflection is then:

$$f=\frac{W_1L^3}{8\,EI}+\frac{W_2L^3}{3\,EI}.$$

The frequency of any cantilever spring is given by the formula:[4]

$$n=\frac{1}{2\pi}\sqrt{\frac{3\,EIg}{l^3(W+\frac{33}{140}wl)}}\text{ per second}$$

where W=concentrated load at end of spring.
w=weight of spring per unit length.
l=length of spring.
g=acceleration of gravity.
n=the number of vibrations per second.

In the R. A. F. instrument a semicircular quartz fiber is used as a cantilever spring, the deflection being given by the formula:[4]

$$D=Kg\frac{\gamma^4}{A^2}$$

where γ=radius of the semicircle.
A=radius of fiber section.
K=material constant.

The period is given by [5]

$$T=\frac{B\gamma^2}{A}$$

where T=time of one complete period.
B=material constant.

If the deflection of the spring under $1g$ is D_1, and the deflection under Xg is D_2, then the unknown quantity X is given by

$$\frac{D_1}{D_2}=\frac{1g}{Xg}$$

or

$$X=\frac{D_2}{D_1}.$$

The material of the spring should be chosen to give the least hysteresis and the most constant zero. Fused quartz probably has the nearest to any material the desired qualities, but it is hard to obtain in large sizes. Spring steel, hardened and tempered, seems to be very satisfactory, the creep of the zero being negligible. It is quite evident that the spring and the base on which it is mounted should have the same coefficient of expansion when used as on the N. A. C. A. instrument, otherwise the end of the spring will move relative to the mirror axle with varying temperature, thus changing the scale and possibly the zero of the instrument.

RESPONSE OF SPRINGS TO SHOCKS OF VARIOUS DURATIONS.

In order to study the motion of an accelerometer spring when acted on by varying forces, it is necessary to apply the general equation of harmonic motion:

$$\frac{d^2y}{dx^2}+\frac{2K\,dy}{dt}+p^2y=f\sin n\,t$$

the complete solution of which is:

$$y=\frac{f\sin\delta}{2\,Kn}\sin\,(nt-\delta)+ae^{-Kt}\sin\,(qt+\epsilon) \qquad (1)$$

$$\text{where, }\tan\delta=\frac{2\;Kn}{p^2-n^2}$$

$$q=\sqrt{p^2-K^2}$$

a and ϵ are arbitrary constants determined by the initial conditions.

$p=\frac{2\,\pi}{T}$, where T is the natural period of the spring.

$n=\frac{2\,\pi}{\tau}$, where τ is the period of the forcing vibration.

f is the amplitude of the forced vibration.
K is the damping coefficient.

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[4] Morley, Strength of Materials, page 454.

[5] R+M=No. 376 of British Advisory Committee for Aeronautics, September, 1917.

Assuming that

$$y=O \text{ when } t=O.$$

then $a \sin \epsilon = \frac{f (\sin \delta)^2}{2 \ Kn}$ (2)

Differentiating,

$$\frac{dy}{dt} = \frac{f \sin \delta}{2 \ K} \cos \ (nt-\delta) + a[-Ke^{-Kt} \sin \ (qt+\epsilon) + e^{-Kt} \ q \cos \ (qt+\epsilon)] \tag{3}$$

Assuming that

$$\frac{dy}{dt} = O \text{ when } t=O$$

$$\frac{f \sin \delta}{2 \ K} \cos \delta - Ka \sin \epsilon + aq \cos \epsilon = O \tag{4}$$

$$\text{or } \frac{f \sin \delta}{2 \ K} \cos \delta - K \ a \sin \epsilon + q \ a \sin \epsilon \cot \epsilon = O \tag{5}$$

as $a \sin \epsilon$ is known from (2) the value of ϵ and a can be found, thus determining all the constants of the equation.

The magnitude of the dynamical force acting on the spring is given by,

$$F = f \sin \ n \, t \tag{6}$$

By substituting values of t in these equations, the motion of the spring can be plotted. By the use of Fourier's series the effect of a nonharmonic force can be studied in the same manner; but as the variations of the accelerations usually follow sine curves very closely, it was thought that nothing of value would be learned by this method.

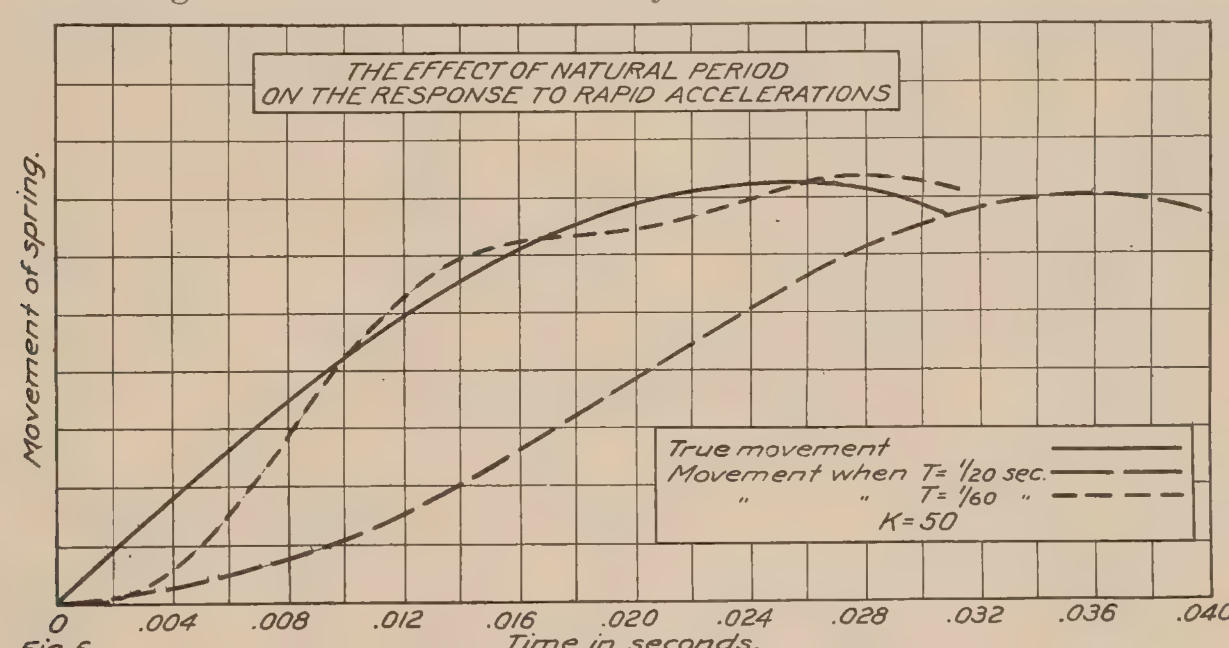

Fig. 6.

THE EFFECT OF NATURAL PERIOD ON RESPONSE TO SHOCKS.

In figure 6 is plotted a curve of acting force rising to a maximum harmonically in 0.025 second. On the same sheet are plotted the motions of two similar springs, except that one has a period of one-twentieth second and the other a period of one-sixtieth second. The higher period spring with six times the period of the acting force oscillates about the true curve, reaching a maximum 0.003 second later and 1.8 per cent higher. The slower spring, with a period of twice the acting force, rises very slowly, reaching a maximum 0.010 second later and 3 per cent lower. These curves show that a spring with a natural period of six times that of the forced vibration will give the height of the maximum within 2 per cent and its time of occurrence within 10 per cent. The curve can, however, be corrected so that it should be possible to obtain the true maximum within 0.5 per cent in magnitude and 1 per cent in time. If a spring of relatively longer period is used the lag increases, but if proper damping is used the recorded maximum will be ver nearly correct, until the natural period of the spring exceeds the period of the forced vibration. Whenever possible a natural frequency of at least five times the frequency of the variation of the acceleration should be used.

THE EFFECT OF DAMPING ON RESPONSE TO SHOCKS.

In figure 7 are plotted, first, the same acting force as before rising to a maximum in 0.025 second, and the curves of motion of three springs having a natural period of one-sixtieth second and damping coefficient of 5, 50, and 300. With the lowest damping the oscillation about the true curve is very marked and the maximum reached is 10.3 per cent too high and 0.003 second

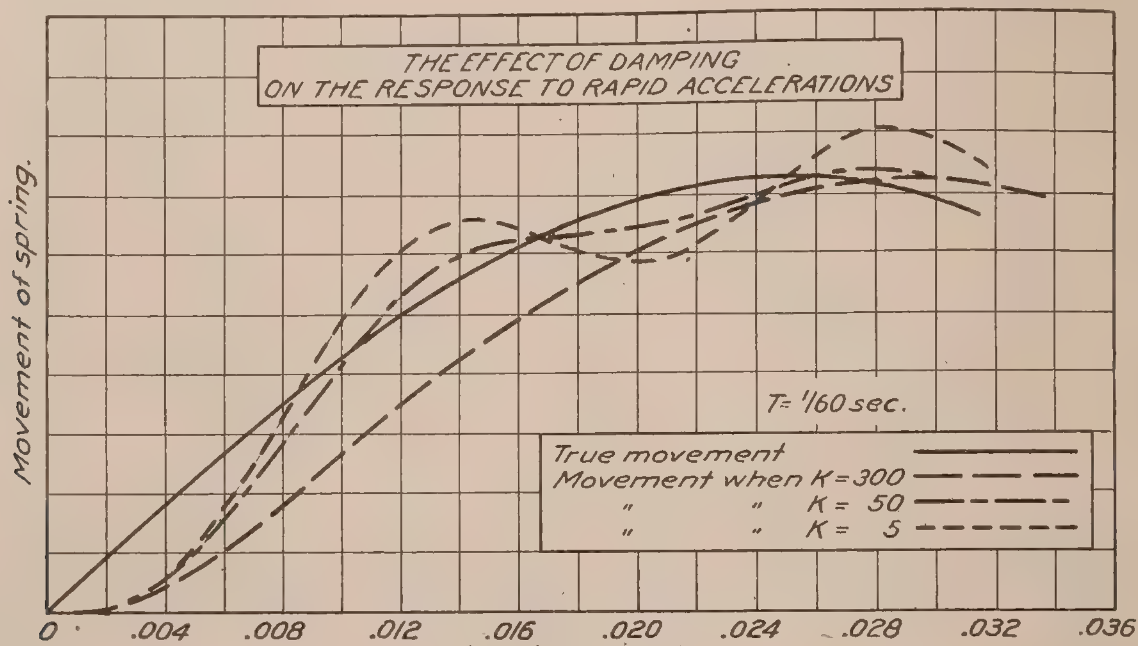

Fig. 7.

too late. With a damping of 50 the oscillations are less marked and the maximum is 1.8 per cent too high. With a damping coefficient of 300, the motion starts slowly, but reaches a maximum of the correct height 0.005 second late. A damping coefficient between 50 and 200 would seem to give the best results. The present N. A. C. A. instrument has a damping factor in the neighborhood of 5, and the R. A. F. instrument has probably a coefficient of even lower magnitude, so that in the new instruments the damping will be increased at least 10 times by using more efficient and more powerful magnets or by employing a liquid dashpot.

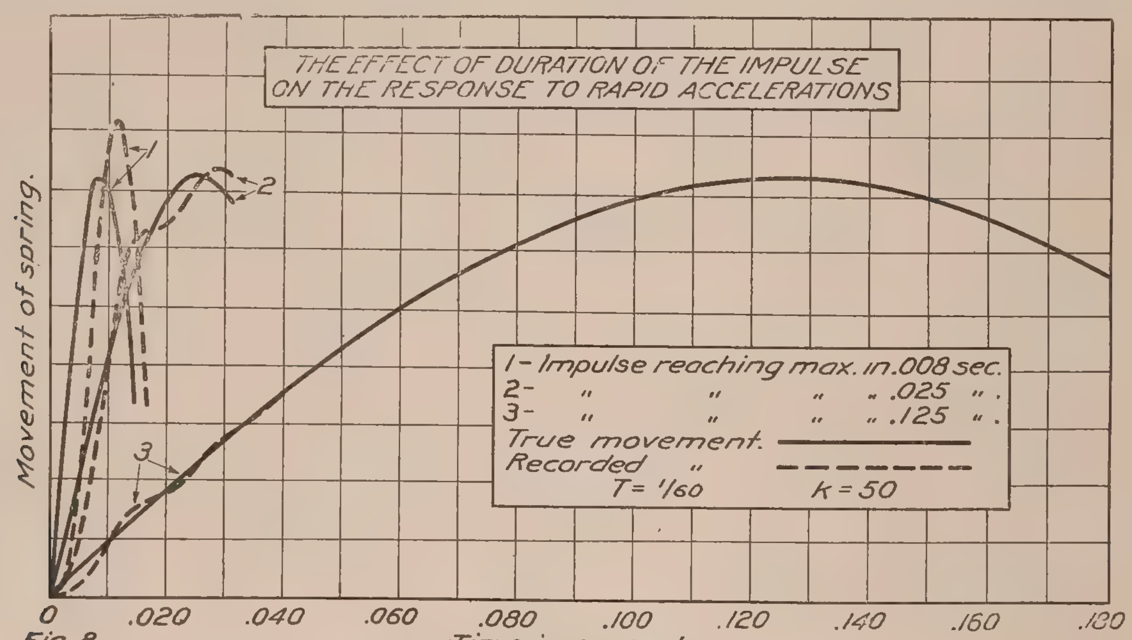

Fig. 8

THE EFFECT OF THE DURATION OF THE SHOCK ON THE MOTION OF THE SPRING.

In order to determine the error in recording shocks of various periods, the motion of a spring of one-sixtieth second natural period and a damping of 50 is plotted when acted upon by impulses reaching a maximum in 0.008, 0.025, and 0.125 second (fig. 8). The height of the recorded maximum increases from the true value for slow impulses to higher and higher peaks as the shock becomes sharper. In order to show this effect more clearly a curve of error in maximum reading is plotted against ratio of natural to forced vibration (fig. 9).

RESONANCE WITH MOTOR AND AIRPLANE VIBRATIONS.

As shown in technical report No. 99, an airplane in the air has a certain fundamental period which is independent of the motor vibration. This period would probably never exceed 25 or 30 vibrations per second for the structure as a whole. It is quite probable, however, that certain portions of the structure would have a secondary vibration of a shorter period. In recording the accelerations on an airplane there are really two problems: The first and the only one that has been dealt with is the acceleration of the center of gravity of the complete machine, due either to air or landing loads, and in this case it is desirable to damp out as completely as possible the high frequency vibrations by means of shock-absorbing supports. On the other hand these high-period vibrations set up local stresses in the structure, and it would be of interest to determine their period and amplitude for different portions of the machine. For this purpose the accelerometer must be rigidly attached to the vibrating part, and, what is rather difficult to accomplish at the present time, the mass of the instrument must be small compared with the mass of the vibrating part.

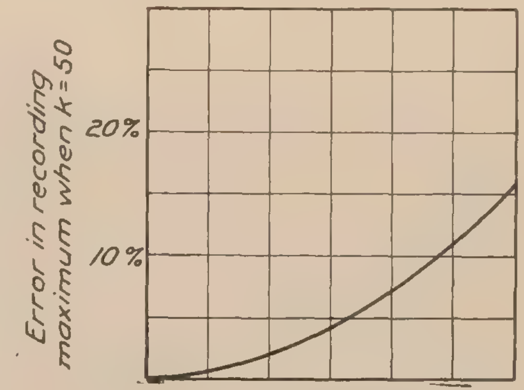

FIG. 9.—Ratio of $\frac{\text{Free period.}}{\text{Forced period.}}$

To return to the problem of measuring the slow accelarations, it is evidently impossible to construct a mounting that will absorb the rapid vibrations, and yet hold the instrument closely to a given position in respect to the machine, so that a compromise is made, and some of the high-frequency vibrations are necessarily transmitted to the instrument while a small part of the maximum shock applied to the airplane as a whole is absorbed by the shock absorbing mounting and so are not recorded.

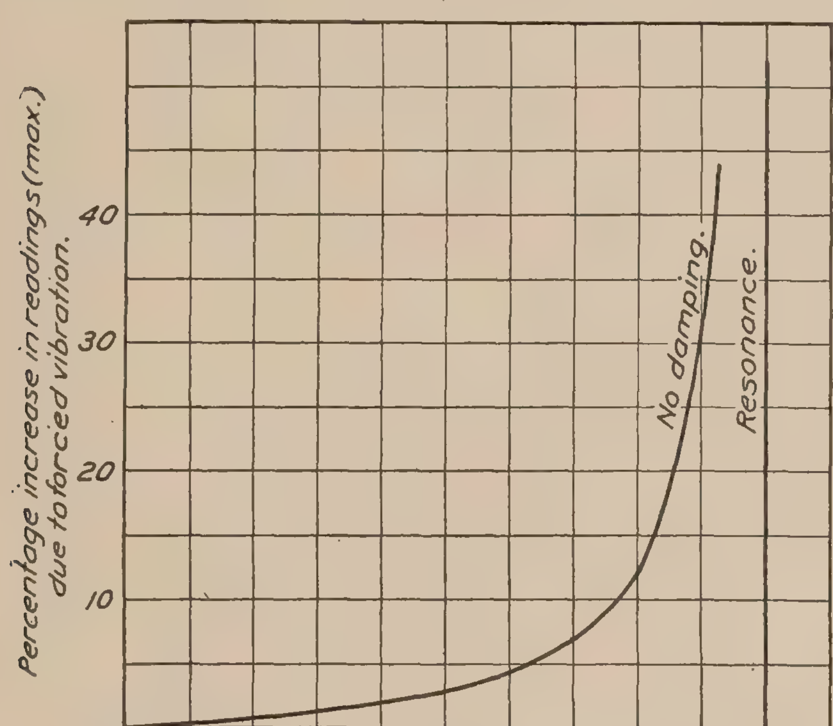

FIG. 10.—Ratio of $\frac{\text{Free period.}}{\text{Forced period.}}$

If there is no damping the errors in the recorded maxima depend on the ratio of the natural to the imposed period, and increase rapidly as the periods approach each other. The nature of the variation of these errors is shown in figure 10. With a damped spring (and in any actual case there is always some damping) the amplitude would not reach infinity at resonance. A simple expression can not be found for the amplitude in terms of the damping, but the error due to an approach to resonant conditions will be decreased nearly proportionally as the damping coefficient is increased. In order to show the effect clearly a curve of acting force and the motion of the spring are shown in figure 11. With a natural period of one-sixtieth second, a forced period of one-thirtieth second, and a damping coefficient of 50, the maximum recorded is not in error by more than 1 per cent; so that it may be concluded that if the natural frequency is not less than twice the forced frequency, no appreciable error will result.

ERRORS DUE TO ACCELERATIONS ACTING AT OTHER THAN NORMAL TO THE PLANE OF THE SPRING.

Flat spring accelerometers are subject to errors due to forces acting along the axis of the spring. Assuming that the spring is normally straight, homogeneous, of constant section, and submitted to an unvarying load, the bending moment due to normal acceleration is:

$$M_x = \frac{-K_x Wx^2}{2} \tag{8}$$

where W = the weight per unit length of spring.

K_x = the acceleration, in terms of g, acting normal to the plane of the spring.

x = the distance from the free end of the spring.

Integrating (8) twice gives the deflection at any point as:

$$y_x = \frac{K_x W}{6\,E\,I}\left(l^3 x - \frac{x^4}{4}\right) \tag{9}$$

where E = modulus of elasticity of the spring.

I = moment of inertia of the spring section.

l = free length of the spring.

y_x = normal deflection under K_x, measured relative to the free end of the spring.

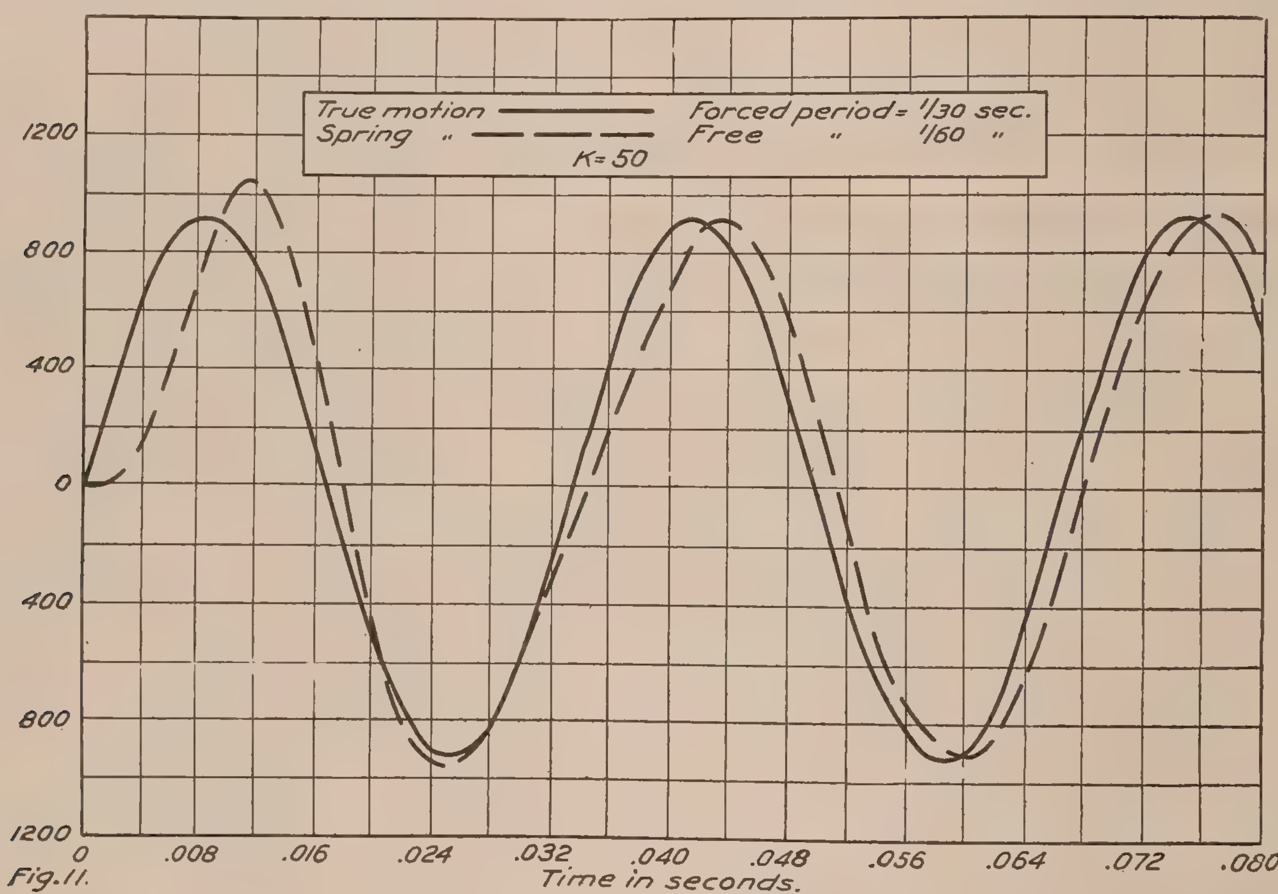

Fig. 11.

The bending moment due to accelerations along the axis of the spring would be zero if the spring was in an undeflected position. If the spring is deflected, the moment arm of an element of mass at any point with respect to an axis of bending at any other point is equal to the difference between the normal deflections of the two points. The bending moment is, then:

$$M_y = \int_0^{x_o} K_y W\,(y_o - y)\,dx \tag{10}$$

where K_y = the longitudinal acceleration in terms of g.

Substituting in this equation in the values of y and y_o found in (9)—

$$M_y = \frac{K_x K_y W^2}{6\,E\,I}\int_0^{x_o}\left(l^3 x_o - \frac{x_o^4}{4} - l^3 x + \frac{x^4}{4}\right)dx$$

$$= \frac{K_x K_y W^2}{6\,E\,I}\left[l^3 x_o x - \frac{x_o^4 x}{4} - l^3\frac{x^2}{2} + \frac{x^5}{20}\right]_0^{x_o}$$

$$= \frac{K_x K_y W^2}{6\,E\,I}\left(l^3 x_o^2 - \frac{x_o^5}{4} - l^3\frac{x_o^2}{2} + \frac{x_o^5}{20}\right)$$

$$= \frac{K_x K_y W^2}{6\,E\,I}\left(\frac{l^3 x_o^2}{2} - \frac{x_o^5}{5}\right)$$

Dropping the subscript, since x_0 can have any value, and integrating twice:

$$y_y=\frac{K_x K_y W^2}{36\ E^2\ I^2}\left(\frac{l^3x^4}{4}-\frac{x^7}{35}-\frac{4l^6x}{5}\right)$$

Substituting l for x in the expressions for y_x and y_y, it appears that the deflections of the free end of the spring due to the accelerations in the two directions are:

$$y_x=\frac{K_x\,Wl^4}{8\,E\,I}$$

$$y_y=\frac{-9\,K_x K_y\,W^2}{560\ E^2\ I^2}$$

$$\frac{y_y}{y_x}=\frac{-9\,W\,l^3 K_y}{70\ E\ I}=\frac{72\,K_y y_x}{70\ K_x\cdot l}$$

The ratio $\frac{y_x}{K_x}$ is a constant for any given instrument, and is equal to the static deflection of the free end of the spring. The ratio of the deflections is, then:

$$\frac{y_y}{y_x}=-\frac{36}{35}\frac{\delta_0}{l}K_y$$

where $\delta_0=$ the static deflection $\frac{y_x}{K_x}$

Since any change in deflections perpendicular to the plane of the spring gives rise to a change in the deflection due to forces acting parallel to that plane, there is a secondary effect which modifies y_y. If, for example, K_y acts toward the free end of the spring, so that y_y and y_x are in the same direction, the increase of y due to the addition of y_y will itself produce a further increase, and the total effect of longitudinal acceleration will be greater than that given by the first approximation written above. If the two deflections are opposed, on the other hand, the actual value of y_y will be less than that given by the approximate formula. These effects can be allowed for by substituting y_t, the total deflection, for y_x in the above equation, writing:

$$\frac{y_y}{y_t}=\frac{y_y}{y_x\pm y_y}=-\frac{36}{35}\frac{\delta_0}{l}K_y$$

$$\frac{y_y}{y_x}=\frac{-\frac{36}{35}\frac{\delta_0}{l}K_y}{1\pm\frac{36}{35}\cdot\frac{\delta_0}{l}K_y}$$

In the glass fiber instrument devised and used at the Royal Aircraft Establishment l is 1.3 cm. and δ_0 ranges from 0.05 to 0.08 cm. For an acceleration of 1g along the axis of the spring $\frac{y_y}{y_x}$ would therefore be 0.05. The accelerometer may be placed with the axis of the spring coinciding either with the X or the Y axis of the airplane. The accelerations along the Y axis certainly never exceed 1g, whereas the computation of the behavior of a JN2 during a loop[6] shows that the longitudinal deceleration of that machine when pulling out of a dive may be as great as 1.92 g. The conditions assumed in this problem were unduly severe, and 1g may be taken as the maximum acceleration along the spring axis to which the accelerometer will be submitted. In the R. A. E. accelerometer an acceleration of this magnitude would produce an error of $+5.43$ per cent or -4.89 per cent in the determination of the normal acceleration, or a maximum error of 0.23 g for the largest normal acceleration so far recorded. This is considerably greater then the sensitivity of the instrument, and shows that it is not safe to rely on its indications to within 0.1 g.

In the N. A. C. A. accelerometer as originally designed l is 12.7 cm., and δ_0 is 0.006 cm. The maximum error due to a longitudinal acceleration of g. under these conditions would be 0.05 per cent, the plus and minus errors being practically identical. This is 0.002 g for the maximum normal acceleration. It is evident that in this instrument the errors due to accelerations at right angles to the one to be measured will be small enough to be neglected with perfect safety.

[6] Forces in Dive and Loop: Bulletin Airplane Engineering Department, U. S. A. June, 1918.

As has already been noted, the assumption of a uniform distribution of weight along the spring does not accord closely with the facts. If it be assumed, as an alternative, that all the weight is concentrated at the free end of the spring, the bending moment due to the normal acceleration is:

$$M_x = -W \cdot K_x \cdot x$$

The deflection due to normal acceleration is found by integrating twice:

$$i = \frac{1}{EI.} W. K_x \left(\frac{-x^2}{2} + \frac{l^2}{2} \right)$$

$$y_x = \frac{1}{EI.} W. K_x \left(\frac{l^2 x}{2} - \frac{x^3}{6} \right).$$

Proceeding in the same manner as for a distributed load—

$$M_y = K_y W y_x = \frac{W^2 K_x K_y}{EI} \left(\frac{l^2 x}{2} - \frac{x^3}{6} \right)$$

$$i_y = \frac{W^2 K_x K_y}{E^2 I^2} \left(\frac{l^2 x^2}{4} - \frac{x^4}{24} - \frac{5 l^4}{24} \right)$$

$$y_y = \frac{W^2 K_x K_y}{E^2 I^2} \left(\frac{l^2 x^3}{12} - \frac{x^5}{120} - \frac{5 l^4 x}{24} \right).$$

The deflection at the free end, due to longitudinal acceleration, is:

$$y_{y_0} = \frac{-W^2 K_x K_y l^5}{E^2 I^2} \cdot \frac{2}{15}.$$

Then

$$\frac{y_{y_0}}{y_{x_0}} = -\frac{W K_y l^2}{EI} \cdot \frac{2}{5} = -\frac{K_y}{K_x} \cdot \frac{y_{x_0}}{l} \cdot \frac{6}{5}.$$

The error due to longitudinal acceleration in this case is therefore about 17 per cent greater than in the case of a uniformly distributed loading. These cases are the extreme antitheses of each other and the true value of the error in either the R. A. E. or the N. A. C. A. instrument will lie somewhere between the two values found, as both these accelerometers have a tendency to concentrate the active mass near the free end of the springs.

Instruments of the Zahm type, using helical springs, are free from these types of error due to longitudinal accelerations, except in so far as the friction of the stylus is increased. This effect certainly can be neglected if the mounting is carefully made.

THE ERRORS DUE TO ANGULAR ACCELERATIONS.

The effect of angular acceleration appears in two ways. In the first place the spring, no matter where it may be placed, is affected by the angular acceleration as such. Secondly, if the origin of coordinates in the spring does not coincide with the center of gravity of the airplane an angular acceleration about the center of gravity will give a linear acceleration to the spring.

The origin will be taken at the base of the spring as a first assumption, being shifted later to a more convenient and logically chosen location. An angular acceleration of K_a radians per second per second, the base of the spring being assumed to remain stationary, imposes upon every element of length d_x a load—

$$\frac{x. K_a. W.dx}{g}$$

where W is the weight per unit length. The shear at a distance x from the base is, integrating from the free end of the spring to the point in question—

$$S_a = \int_x^l \frac{x. K_a. W.dx}{g} = -\frac{K_a. W}{g} \left(\frac{l^2 - x^2}{2} \right)$$

and the bending moment is—

$$M_a = \frac{-K_a \cdot W}{g}\left(\frac{l^2x}{2} - \frac{x^3}{6} - \frac{l^3}{3}\right).$$

Integrating twice more—

$$i_a = \frac{K_a \cdot W}{gEI}\left(\frac{-l^2x^2}{4} + \frac{x^4}{24} + \frac{l^3x}{3}\right)$$

$$y_a = \frac{K_a \cdot W}{gEI}\left(\frac{x^5}{120} + \frac{l^3x^2}{6} - \frac{l^2x^3}{12}\right)$$

The deflection at the free end is, then:

$$y_{ao} = \frac{K_a \cdot W}{gEI}\left(\frac{11\,l^5}{120}\right) = \frac{11}{15} X \frac{l}{g} X K_a X$$

The direct error arising from angular accelerations is therefore directly proportional to the length of the spring, and the R. A. E. instrument, with its very short spring, would seem to have a marked advantage in this particular. It is, however, evident that a judicious location of the origin of coordinates with respect to the center of gravity of the airplane will introduce linear accelerations, resulting from angular accelerations, which will counterbalance the direct effect of the accelerated rotational motion.

The normal acceleration required to produce a deflection equivalent to that produced by the angular acceleration, K_a would be of the magnitude—

$$K_x = \frac{11}{15} X \frac{l}{g} X K_a$$

where K_x is expressed in terms of feet per second per second. It is then evident that if the center of gravity of the airplane lies in the plane of the spring, and eleven-fifteenths of its length from its base there will be no deflection of the free end of the spring due to angular accelerations, and the two manners in which the effects of such accelerations appear just canceling each other. If the weight is concentrated at the tip of the spring, instead of being uniformly distributed along its whole length, the free end should obviously be at the center of gravity. Compromising between the two conditions it may be said that, for the accelerometers now in use, the location of the mounting should be such that the center of gravity lies from 75 per cent to 80 per cent of the way out along the spring.

If the center of gravity is not at the point thus defined there are, as has already been pointed out, two possible sorts of error. The first of these is the error due to angular acceleration when the center of gravity lies in the axis of the spring. By properly choosing the origin in the spring the direct effect of angular acceleration can be eliminated, and the total effect can be reduced to that of a linear acceleration given (in terms of g) by the expression:

$$K_x = \frac{K_a X d}{g}$$

where d is the distance from the center of gravity of the airplane to a point in the spring and 75 per cent of its length from the base.

The analysis of the "loop problem," already mentioned, showed that, under the conditions assumed, the angular acceleration about the Y axis has a maximum value of 10.5 radians per second per second at the instant when the elevator was pulled up, the angular acceleration falls to 2.4 radians per second per second in 0.3 second, and that it never rises above 1.0 radian per second per second after 0.9 second until the loop is completed. The assumption made in this analysis, that the elevator is pulled up instantaneously, is, of course, much too severe, and it is probable that 6.5 radians per second per second is the largest acceleration in pitch that an airplane would ever have to undergo. Experiments on the rolling moment due to the ailerons suggest that the acceleration about the X axis has a maximum value, on small and medium sized airplanes, of about 5 radians per second per second.

If d_x, d_y, and d_z be the projections on the three axes of the distance from the origin of coordinates in the spring to the center of gravity of the airplane, it is evident that the maximum error due to acceleration in roll is 0.15 g when d_y is 1 foot, and that the similar error arising from the pitching motion is 0.20 g when d_x is 1 foot. Corrections can be made, using estimated

values for the angular accelerations, which can be relied upon to reduce these errors by about 60 per cent. In order that the normal accelerations, thus corrected, may be accurate within 0.05 g., the value of d_x must be less than $7\frac{1}{2}$ inches and d_y must not exceed 10 inches.

Another source of error is the centripetal acceleration due to angular velocity. In the usual case, where the accelerations along the Z axis are being measured, centripetal accelerations arise whenever there is any rolling or pitching motion if the active mass of the instrument is above or below the center of gravity. The acceleration is, of course,

$$K_x = \frac{\omega^2 d_z}{g}.$$

The loop analysis showed a maximum angular velocity of 1.81 radians per second occuring 0.4 second after the elevator was pulled up. Since the theoretical time for completing the loop was about a third less than actual measurement shows to be required, this maximum is about 50 per cent too high, and the true maximum may be taken as 1.2 radians per second. Since it is reported that an airplane can be rolled onto its back in 3 seconds, the maximum rolling velocity must be about 1.5 radians per seconds. The error when d_z is 1 foot would then be 0.07 g. As in the case of angular accelerations, approximate corrections can be made, and the error reduced by at least 50 per cent. To keep the corrected value of the normal acceleration within 0.05 g. of the truth, d_z must not exceed 16 inches. This condition is easy to realize, and it will usually be found that the best place for mounting an accelerometer is directly above or below the center of gravity.

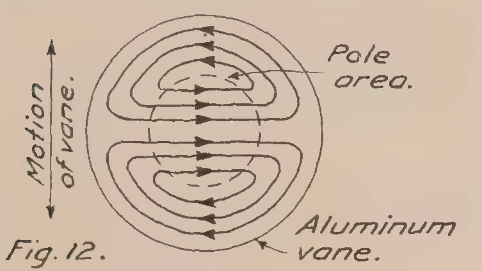

Fig. 12.

EDDY CURRENTS IN DAMPING VANE

MECHANICAL CONSTRUCTION.

As it is desirable to obtain as large a deflection on the film as possible with a high frequency, some device must be used for magnifying the of the end of the spring. In the N. A. C. A. instrument a fairly heavy spring is used, and the motion of its end is transmitted to a very light staff, mounted in hardened-steel sockets. The staff has a small horizontal platform on which a pointed screw from the end of the spring rests, and a thin plane mirror is mounted on the staff and reflects a beam of light through a lens onto a moving film. A watch hairspring attached to the staff holds the platform tightly against the pointed screw, as shown in figure 5. The moment of inertia of the moving parts are so low that they do not appreciably affect the period of the spring, but they do increase the damping due to pivot and air friction. The mirror staff sockets are mounted on a steel base, which runs under the spring and is rigidly fastened to it at its fixed end.

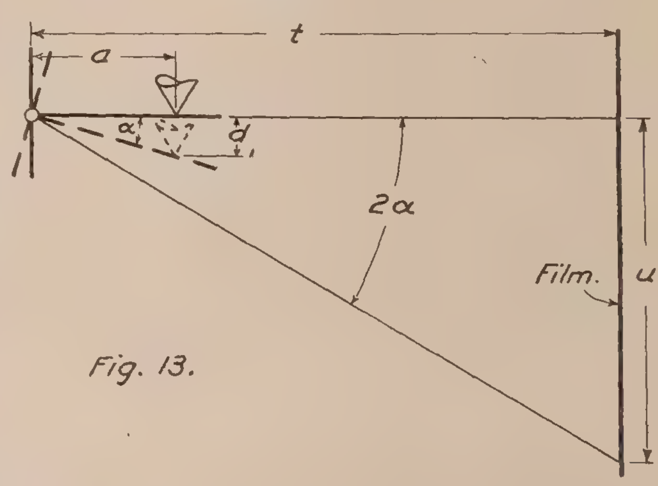

Fig. 13.

The damping of the spring is accomplished by attaching a light aluminum or copper vane to the free end of the spring and allowing it to vibrate between the poles of an electromagnet. When any conducting plate is moved across the lines of force, as shown in figure 12, a current is induced in the plate in such a direction that it tends to oppose the motion of the plate. In order to obtain the maximum damping from a given weight of magnet the vane must be thick enough to carry a heavy current, and it should have ample area outside the magnetic field for the return flow. The air gaps can be reduced to 0.005 inch if the magnet frame is stiff enough to prevent the poles from drawing together. The damping of the N. A. C. A. instrument is not nearly high enough, and in the new instruments the damping will be increased at least 10 times by larger magnets and improved design of the vane.

The scale of accelerations on the film is not quite uniform, as is made clear in figure 13. Let the pointed screw (s) on the end of the spring rest on the platform (p), which is assumed

to be initially horizontal, and the light beam at the same time is horizontal. If the screw (s) is now deflected a distance d vertically downward the angle of rotation of the staff a is given by—

$$\alpha = \tan^{-1}\frac{d}{a}$$

and the deflection on the film in respect to the deflection at the end of the spring is:

$$u = t\tan\left(2\tan^{-1}\frac{d}{a}\right) = \frac{2t\frac{d}{a}}{1-\left(\frac{d}{a}\right)^2}$$

Where d is the deflection of the spring,

a = the moment arm,
t = the distance from the mirror to the film,
u = the deflection on the film,
d = the deflection on the spring.

The ratio of proportionality at any point is then

$$\frac{du}{d\left(\frac{d}{a}\right)} = \frac{2t\left[1-\left(\frac{d}{a}\right)^2+2\left(\frac{d}{a}\right)^2\right]}{\left[1-\left(\frac{d}{a}\right)^2\right]^2} = \frac{2t\left[1+\left(\frac{d}{a}\right)^2\right]}{\left[1-\left(\frac{d}{a}\right)^2\right]^2}$$

It is therefore desirable, in order that the scale may be as uniform as possible, that $\frac{d}{a}$ should be kept as small as possible without making the record inconveniently small.

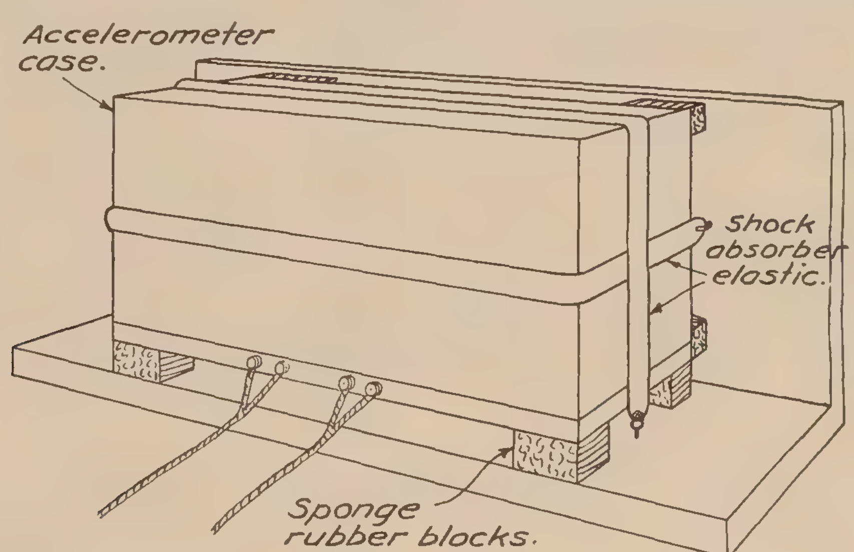

Fig. 14.
METHOD OF MOUNTING ACCELEROMETER.

It is quite essential on an accelerometer record to have an accurate time scale, as in many cases the duration of an acceleration is quite as important as its magnitude. There are two methods of accomplishing this—first, and most satisfactory, is to run the film at constant speed; and second, to run the film at any speed and impress on the record a line or dot at definite time intervals by means of a clock-controlled light or by a tuning fork. This latter method gives a very accurate time record, but it is indirect and awkward to use. Satisfactory methods are discussed in N. A. C. A. Technical Note No. 22 for driving the film at very nearly constant speed, and sufficient synchronization can be obtained between various instruments by connecting all the lamps to one switch, so that the records are started and stopped simultaneously.

It is desirable to either carry a large amount of film, as in the R. A. F. accelerometer or to be able to change light tight film drums in the same way as plate holders on a camera. This latter method is preferable as it separates the records and makes identification simpler, as described in N. A. C. A. Technical Note No. 22.

It is quite important that the accelerometer should be mounted in such a way that the small high-frequency vibrations of the airplane are not directly transmitted to it. The instrument can be very well insulated from shocks by holding the instrument in the hands, but as this is rather a makeshift, and as it can not be easily held near the center of gravity of the machine, a more permanent mounting is desirable. The most satisfactory mounting for an accelerometer is that shown in figure 14. Several blocks of sponge rubber about 1½ inches thick are placed under and at the back of the instrument and shock-absorber elastics hold it firmly against these blocks, thus allowing the instrument to move only a few millimeters, and yet absorbing the shocks satisfactorily.

OTHER USES OF THE ACCELEROMETER.

Up to the present time the accelerometer has been used only in airplane work, but there are many other problems that could be studied to advantage with this type of instrument. Perhaps its most obvious use is in the design of automobile spring suspensions, as the riding comfort of a car can be accurately recorded. A few records of this type have been taken with the N. A. C. A. instrument, and every oscillation of the car springs can be seen clearly on the record. The riding qualities of tires could also be studied by mounting the accelerometer directly on the axle and running over a definite obstacle. The engine vibration can be studied in the same way, and the merits of various types of motors easily compared. A longitudinal accelerometer would be a convenient method of measuring the pickup and braking power of automobiles, and a lateral accelerometer would record the side load on the tires when rounding curves.

In the same way the riding qualities and stresses in steam and electric cars could be studied, particularly as to types of rail joints and switches, and the banking of curves. Another use that would be more interesting than valuable is the study of amusement devices, such as roller coasters, in order to furnish data for new design and for advertising purposes.

If a high-period accelerometer is fastened rigidly to a machine which is out of balance, such as a gasoline motor, a curve will be obtained showing the unbalanced component acting normal to the accelerometer spring. By analyzing this curve not only the amount of the forces can be measured, but also the components that make up the unbalanced force can be separated and studied. The N. A. C. A. accelerometer is so sensitive to slight vibrations that the shock of hammering or a slamming door in a distant part of the building is distinctly recorded, and the effect of anyone walking even lightly in the same room is quite evident. By making the instrument much more sensitive, which could easily be done, it might be used for studying building vibrations due to heavy machinery, such as printing presses, and to detect very slight tremors in any structure.

REPORT No. 101

THE CALCULATED PERFORMANCE OF AIRPLANES EQUIPPED WITH SUPERCHARGING ENGINES

IN TWO PARTS

By E. C. KEMBLE
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REPORT No. 101.

THE CALCULATED PERFORMANCE OF AIRPLANES EQUIPPED WITH SUPERCHARGING ENGINES.

By E. C. Kemble, Harvard University.

PART I.

THE CALCULATION OF PERFORMANCE CURVES FOR AN AIRPLANE ENGINE FITTED WITH A SUPERCHARGING CENTRIFUGAL COMPRESSOR.

RÉSUMÉ, PART I.

The following report was prepared by Mr. E. C. Kemble at the request of the National Advisory Committee for Aeronautics and covers the theoretical discussion of the performance of an airplane as affected by the use of a supercharging engine, and also includes a very thorough discussion of the respective merits of the different types of superchargers that are considered.

The power developed by an aircraft engine under any given external conditions can be computed approximately if the normal power at the given speed is multiplied by appropriate temperature and pressure correction factors.

The temperature correction factor is given by equation (1), which is taken from Report No. 45.

When the intake and exhaust pressures are equal, it is best to use an equation based on the work of Report No. 46 for the pressure correction factor.

For unequal intake and exhaust pressures, the correction factor for a small range of values may be taken from figure 2, which is copied from Report No. 45.

The temperature rise in the compressor, which has an important part in determining the power output of the engine, can easily be computed when the pressure ratio, the shaft efficiency, and the heat "radiated" per pound of air by the compressor and discharge pipe are known. Under typical conditions, with the compressor exposed to the full force of the propeller slip stream, the computed value of the ratio of the actual temperature rise to the theoretical rise without heat "radiation" is 0.864.

The efficiency of the compressor and the power which it absorbs depend on the quantity of air handled per unit time. It therefore becomes necessary to discuss the variation of the volumetric efficiency of the engine with the intake temperature and the exhaust back pressure.

It is assumed that the compressor is designed for operation at a certain normal altitude and normal speed. The calculation of the net horsepower available at the propeller under these normal conditions is particularly simple. In a numerical example it is assumed that the Liberty engine is fitted with a gear-driven compressor designed to furnish sea-level carburetor pressure at 18,000 feet and an engine speed of 1,700 revolutions per minute. The shaft efficiency of the compressor is assumed to be 64 per cent. The computed horsepower is 371.

In calculating the power of an engine equipped with a turbine-driven compressor, it is assumed that the back pressure created by the turbine is equal to the increase in the carburetor pressure produced by the blower. The computed power to be expected from a Liberty engine fitted with a turbine-driven supercharger under the conditions of the preceding problem is 394.

In laying out performance curves showing the power to be expected from an engine-compressor unit at various speeds and altitudes, the variation in the efficiency of the compressor should be taken into account. The computation is somewhat involved, but can be carried through graphically.

Figure 11 shows comparative performance curves evaluated in this manner for the turbine-driven compressor, the gear-driven compressor, and for the engine operating without the compressor. The curves for the gear-driven installation are not carried to the highest altitudes on account of lack of data regarding the pressure correction coefficient for very low exhaust pressures. In carrying the computation through it was assumed that the maximum safe speed of the compressor was that required to give sea-level carburetor pressure to the engine at 18,000 feet when the crank-shaft speed was 1,700 revolutions per minute.

Curves showing the relative fuel consumption at different speeds and altitudes are easily obtained if it is assumed that the carburetor of the engine is adjusted for maximum power. (Cf. fig. 13.) They show an increase of about 20 per cent in the fuel economy at normal speed and an altitude of 20,000 feet. An even larger gain is to be expected in practice as a result of avoiding carburetor troubles due to the low temperatures which prevail at great altitudes.

1. INTRODUCTION.

This report is the outgrowth of a set of calculations made during the war on the probable performance characteristics of an airplane whose engine is equipped with a supercharging compressor of the gear-driven type. The discussion is here extended to the case of the turbine-driven type of compressor on the basis of the rough empirical rule that the exhaust back pressure created by the turbine is equal to the rise in the intake pressure due to the compressor.

The purpose of the report is twofold. It aims, in the first place, to outline a method of predicting the probable performance curves of an airplane fitted with a supercharging centrifugal compressor, and in the second place to apply this method to the case of a typical modern airplane in order to determine, as nearly as possible with the somewhat meager data now available, the gains which the use of a supercharger may be expected to bring in the near future.

Part I of the report is devoted exclusively to the discussion of the performance of the engine-compressor unit itself. This part is itself separable into two main divisions. In the first of these only so much of the theory is taken up as is necessary for the evaluation of the power which the engine and supercharger will deliver under the conditions for which the latter is designed. In the second division the variation in the efficiency of the compressor is considered, and a semi-graphical method of laying out performance curves for all speeds and altitudes is evolved.

2. POWER DEVELOPED BY ENGINE WITH KNOWN INTAKE PRESSURE AND TEMPERATURE.

The computation of the power developed at various altitudes by an airplane engine operating with or without a supercharging compressor is greatly facilitated by the results of recent tests made at the Bureau of Standards and embodied in Reports No. 45 and No. 46 of the National Advisory Committee for Aeronautics.

The power developed by an airplane engine at any given speed depends on three externally variable quantities, viz, the temperature of the air entering the carburetor (intake temperature), the pressure of the air entering the carburetor (intake pressure), and the exhaust pressure. The variation in the power delivered by an engine with each of these quantities has been studied in the tests cited above.

In order to determine from the horsepower observed at the temperature t_o (F.) the horsepower to be expected at the temperature t, we multiply by the correction factor.[1]

$$F_t = \frac{H.P.}{(H.P.\ \text{at}\ t_o)} = \frac{920 + t_o}{920 + t}. \tag{1}$$

Wherever the intake and exhaust pressures of an engine are equal, the following formula may be used to determine the variation of the power with variation in the common value of these two pressures:

$$r_p = \frac{H.P.}{(\text{H.P.})_{76}} = 1 - \frac{1}{\eta_{76}}\left[1 - \frac{P}{76}\right]. \tag{2}$$

[1] Cf. Report No. 45, National Advisory Committee for Aeronautics, 1920, Part 3. It is unfortunately necessary to use the above factor for temperatures outside the range of its experimental verification.

Here p is the intake and exhaust pressure in centimeters of mercury; η_{76} and $(H.\ P.)_{76}$ are respectively the mechanical efficiency and the brake horsepower at 76 cm.[2] The above equation follows directly from two hypotheses strongly supported by Report No. 46,[3] viz, (a) that the friction horsepower is independent of the intake and exhaust pressures, and (b) that the indicated indicated horsepower is directly proportional to the pressure at constant temperature. Figure 1 shows r_p plotted against p in accordance with (2) for three different values of the mechanical efficiency.

The experiments of Moss[4] show that when a centrifugal compressor is driven by an exhaust gas turbine of careful design, the pressure rise generated by the compressor under the best conditions is approximately equal to the back pressure created by the turbine. In order to avoid excessive complication in the calculations it will be assumed throughout this report that the intake and exhaust pressures of an engine fitted with a turbine driven compressor are always equal. In adopting this rough assumption we admittedly overestimate somewhat the per-

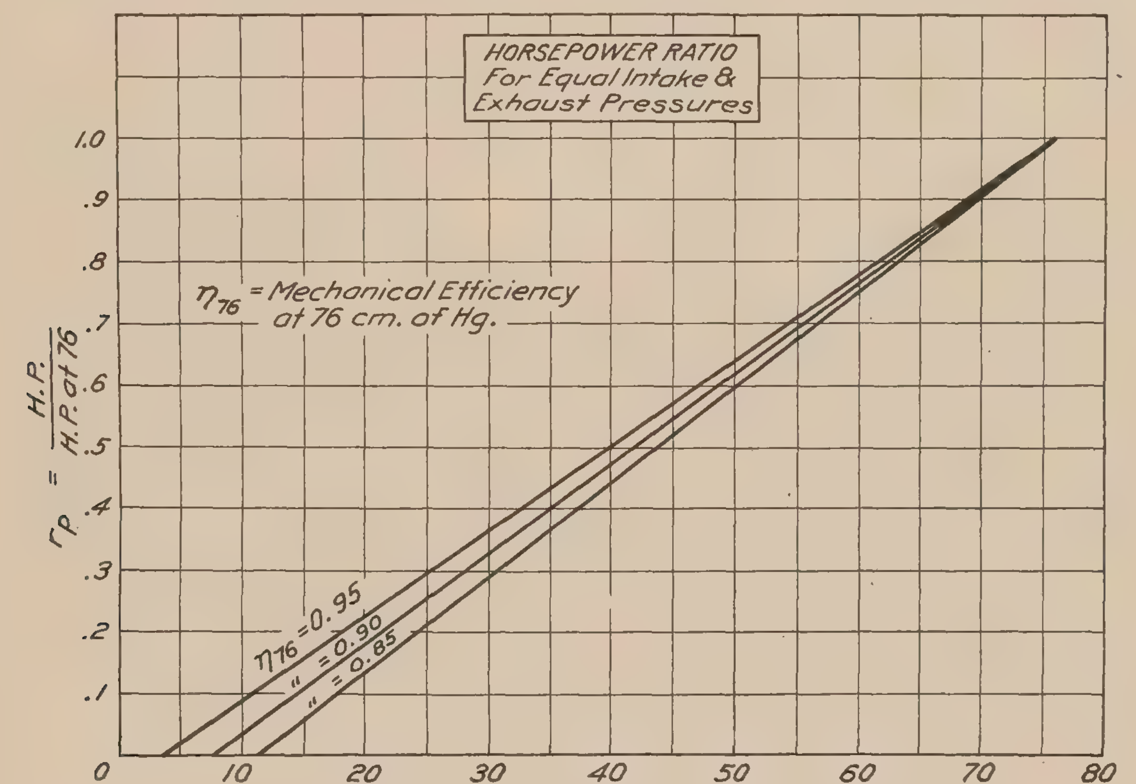

Fig. 1.

formance to be expected under conditions which depart from the normal. In the writer's opinion, however, the error involved is of a minor character.

In the case of a gear-driven compressor, on the other hand, the engine-exhaust pressure is less than the carburetor pressure, and the gross power output of the engine depends on this pressure difference as well as on the carburetor pressure. In dealing with a problem of this type figure 2 may be used. This diagram, which is taken from Report No. 45,[5] shows values of the ratio R_p of the horsepower developed with any given carburetor and exhaust pressures to the horsepower developed when the two pressures are each 76 cm. of mercury. It is based on tests of a Hispano-Suiza 150-horsepower engine with a compression ratio 5.3 to 1 at 1,500 revolutions per minute. At this speed the engine in question has mechanical efficiency of 92 per cent. Strictly speaking this set of curves is applicable only to engines having the same compression ratio and mechanical efficiency, but, in default of better information, it may be used as a first approximation for engines of other compression ratios and other mechanical efficiencies. It will be observed that the variation of r_p (fig. 1) with the mechanical efficiency for equal carburetor and exhaust pressures increases as the pressure is lowered. On this account the curves of

[2] A summary of the notation used is given at the end of each part of the report.
[3] "A Study of Airplane Engine Tests", by Victor R. Gage, Report No. 46, National Advisory Committee for Aeronautics, 1920.
[4] "The General Electric Turbo Supercharger for Airplanes", by Sanford A. Moss, Aviation and Aeronautical Engineering, VIII, p. 147, 1920.
[5] "Effect of Compression Ratio, Pressure, Temperature, and Humidity, on Power," Part 2, by H. C. Dickinson and G. V. Anderson.

figure 2 may not be extrapolated to low pressures and used there for mechanical efficiencies other than 92 per cent without danger of serious error.

In order to predict the performance curves of an engine operating at various altitudes, a sea-level horsepower-speed curve (fig. 3) and curves showing the variation in the mean atmospheric temperature and pressure with altitude are needed. Figure 4 shows the relationship between temperature, pressure, and altitude recently agreed upon as standard by the French and Italian Governments. The temperature-altitude graph, from which the pressure-altitude graph is computed, diverges quite appreciably from the curves for the observed mean temperature both for very small and very great altitudes, but not enough to seriously affect the present computation.

Figure 5 shows the horsepower of an average Liberty engine as a function of altitude and speed, computed from figures 1, 3, and 4, together with the temperature correction factor of equation (1).

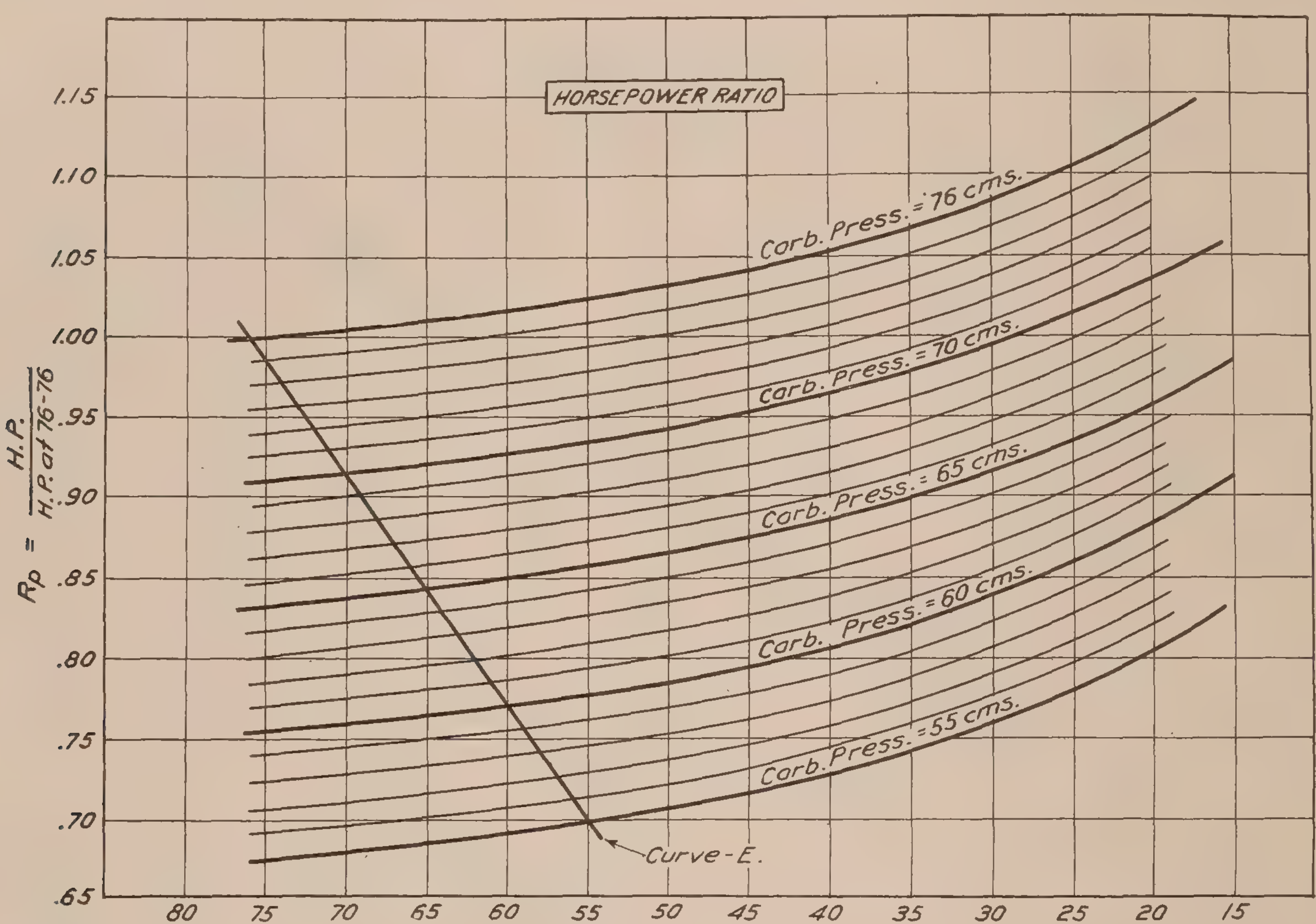

Fig. 2.

3. CALCULATION OF INTAKE PRESSURE AND TEMPERATURE FOR STANDARD ALTITUDE AND ENGINE SPEED: LIMITATIONS ON VALUE OF CARBURETOR PRESSURE.

In order to predict the performance of an engine operating with a supercharger, it is further necessary to know the increase in pressure and temperature of the air as it passes through the compressor. If the compressor is gear-driven, we must also know the power which it absorbs. Taking up first the rise in pressure, we observe that it is limited, in general, by two factors, viz, the necessity for avoiding "preignition" and the maximum safe speed of the compressor. The power available for compression is practically unlimited either in the case of a gear-driven compressor or of one driven by an exhaust-gas turbine.

The experiments of Moss have already shown that a high-compression engine, which is on the point of "preignition," or pinking, at sea level, may be operated with sea-level carburetor pressure at any altitude in spite of the high temperature of the compressed air. It is possible that higher carburetor pressures can be used with high carburetor temperatures than with low, but this point has yet to be settled, and in the present calculation it will be assumed that the carburetor pressure may not exceed a standard sea-level atmosphere. It is

worthy of note that a small net gain in power, even at sea level, is theoretically possible by using a reduced compression ratio and a carburetor pressure which is above normal sea-level value. This gain in power is accompanied by a loss of efficiency, however, and is probably of no practical importance.

The pressure ratio of a centrifugal compressor of given construction depends on the ratio of the speed to the volume of air handled, but is independent of the intake pressure. It follows that when the engine and compressor speeds are kept constant, the carburetor pressure is directly proportional to the pressure of the atmosphere. In any case there will be a certain maximum altitude for each engine speed at which the compressor can develop sea-level pressure. Above this altitude the carburetor pressure and the net available horsepower must drop off steadily as they do for an engine which is not equipped with a supercharging compressor.

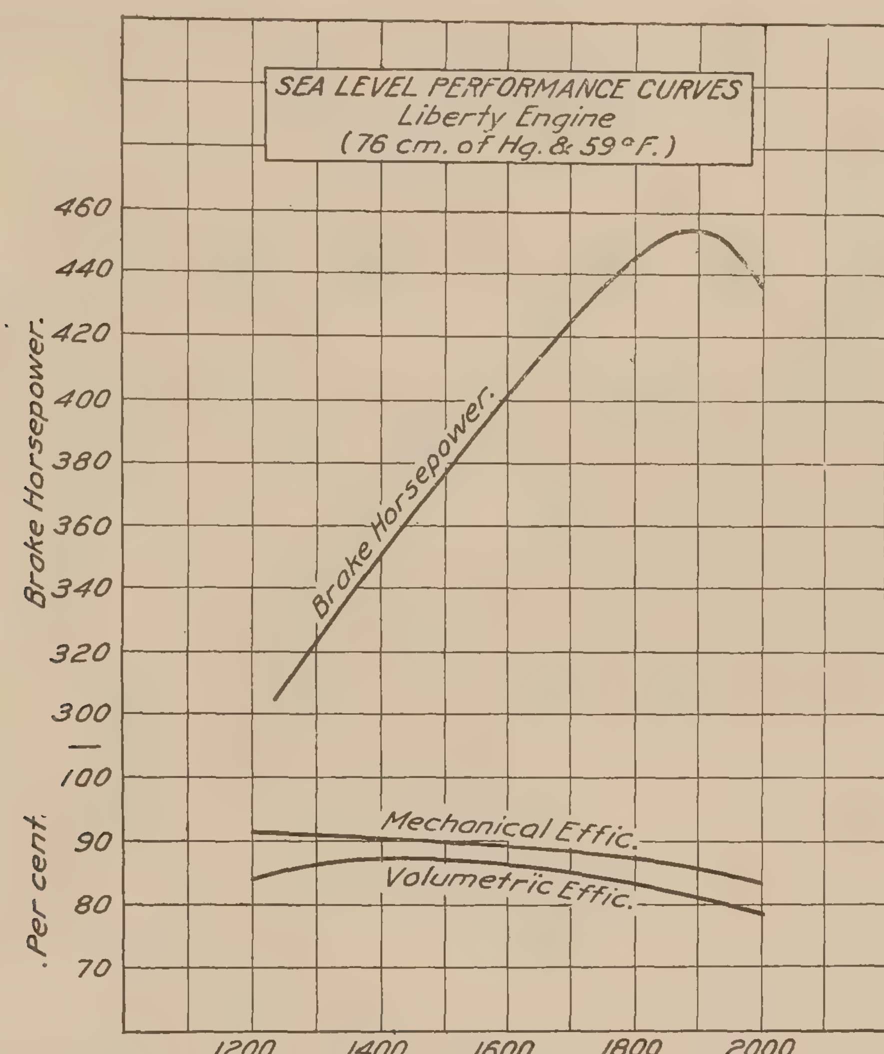

Fig. 3.

Below this altitude some means must be adopted for keeping the carburetor pressure from exceeding sea-level value. This may be done either by decreasing the compressor speed or by throttling the inlet to the compressor. The former method of control is the better, if practicable, since throttling the inlet leads to excessive heating of the air.

4. THE NORMAL ALTITUDE AND SPEED OF OPERATION.

It will be assumed that the compressor is designed to operate under the conditions prevailing at a certain definite altitude, which we will designate as the "normal" altitude. It will further be assumed that when engine and compressor operate at their normal speeds at this normal altitude, the compressor will just develop sea-level carburetor pressure and will work with maximum, or nearly maximum, shaft efficiency. The calculation of the gain in power due to supercharging is particularly simple for this one set of conditions, since the hydraulic and shaft efficiencies of the compressor may be treated as known quantities.

The normal speed of the compressor may be equal to, or less than, the maximum safe speed. In the former case the normal altitude will be the maximum altitude for normal engine speed at which the compressor can develop sea-level pressure, and consequently the altitude at

which the compressor gives the maximum increase in power. At this altitude the airplane will attain the maximum possible horizontal flight speed consistent with normal engine speed.

In the numerical example discussed in this report, it is assumed that the normal compressor speed is its maximum safe speed.

We shall first consider the operation of the compressor under the normal conditions just described, taking up the general problem later on.

5. TEMPERATURE RISE IN COMPRESSOR.

The carburetor pressure is assumed to have its sea-level value, i. e., 76 cm. of mercury. The carburetor temperature can be computed from the shaft efficiency if the heat lost due to "radiation" is known. The method is as follows:

Let the subscripts 1 and 2 refer to the states of the air as it enters the compressor and enters the carburetor, respectively. (We assume that the carburetor is located between the compressor and the engine.)

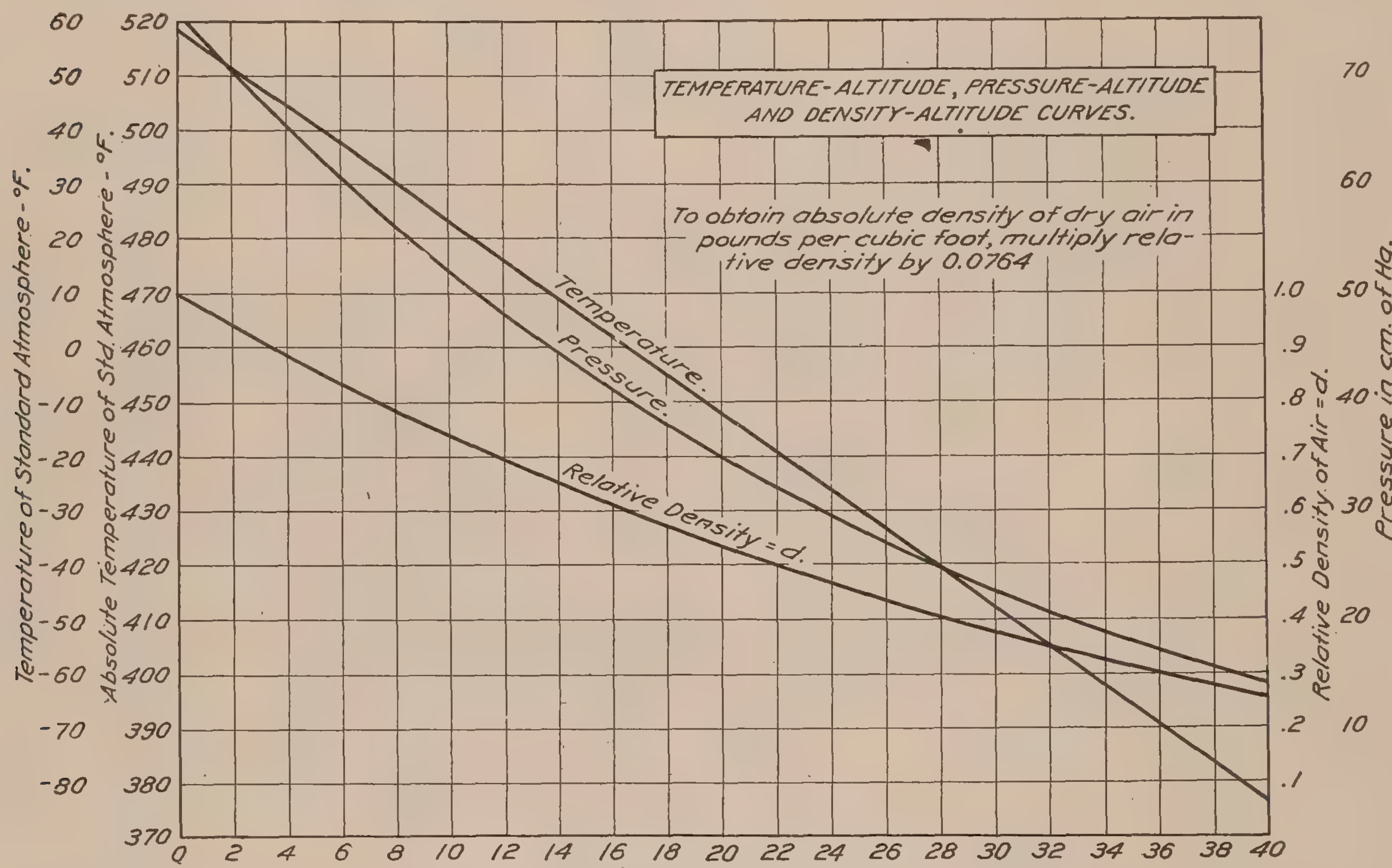

Fig. 4.

Let P, p = absolute pressure in pounds per square foot and centimeters of mercury, respectively;

V = volume in cubic feet;

T = absolute temperature in Fahrenheit degrees;

C_p = specific heat of air at constant pressure in B. t. u. per pound, = 0.241;

C_v = specific heat of air at constant volume in B. t. u. per pound, = 0.171;

$\gamma = C_p/C_v = 1.406$;

J = mechanical equivalent of heat, = 778 foot-pounds per B. t. u.;

I = input of mechanical energy per pound of air handled;

M = air flow through compressor in pounds per minute;

h = heat radiated per minute by the compressor and any cooling device which may be put between the compressor and the carburetor.

Neglecting the kinetic energy of the air in the discharge pipe of the compressor, we equate the net energy input per pound of air handled to the increase in the total heat of the air. Thus

$$\frac{I}{J}-\frac{h}{M}=C_p(T_2-T_1). \tag{3}$$

Let T_2' = the temperature to which the air would rise if the compression were adiabatic;
I' = corresponding energy input per pound;

and let the function A be defined by the equation

$$A(P_2/P_1)=\left(\frac{P_2}{P_1}\right)^{\frac{\gamma-1}{\gamma}}-1=\left(\frac{P_2}{P_1}\right)^{0.289}-1.$$

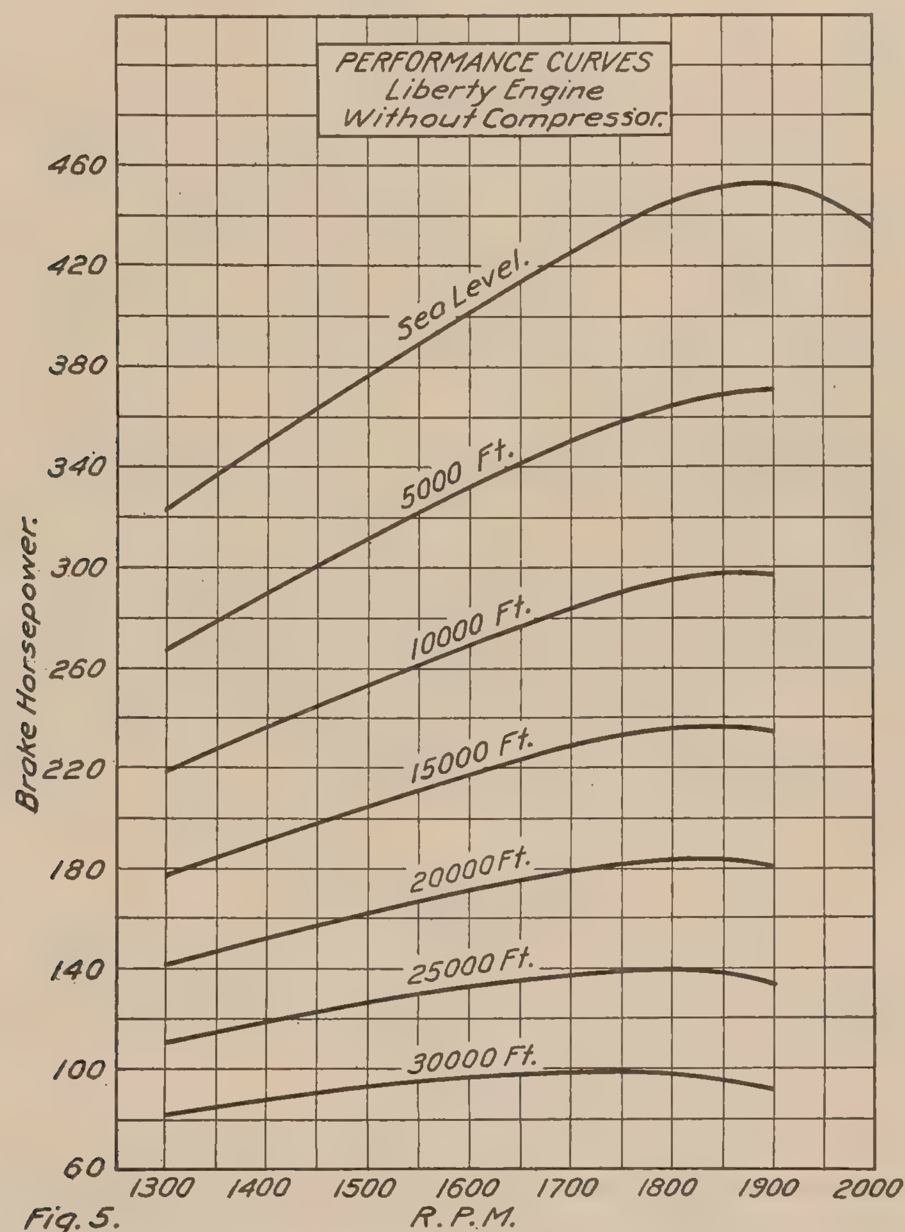

Fig. 5.

Then the familiar formula for the adiabatic compression of a perfect gas gives the relation

$$T_2'-T_1=A(P_2/P_1)T_1,$$

and equation (3) yields

$$\frac{I'}{J}=C_pA(P_2/P_1)T_1. \tag{3a}$$

A graph of the function A is shown on figure 6. The ratio of I' to I is, by definition, the shaft efficiency of the compressor, which we denote by E_s. Therefore

$$\frac{1}{E_s}=\frac{h/M+C_p(T_2-T_1)}{C_pA(P_2/P_1)T_1},$$

and

$$T_2-T_1=\frac{A(P_2/P_1)T_1}{E_s}-\frac{h}{MC_p}. \tag{4}$$

Since the heat radiated should be proportional to T_2-T_1, we introduce the quantity k defined by the relation

$$h=k(T_2-T_1).$$

Equation (3) then becomes

$$T_2-T_1=\frac{\mu A(P_2/P_1)T_1}{E_s}, \tag{5}$$

where

$$\mu=\frac{MC_p}{MC_p+k}. \tag{6}$$

It is evident at once that μ is the ratio of the actual temperature rise to that which would occur if there were no radiation.

The radiation coefficient k will obviously vary a good deal with the installation and also with the conditions of operation. If the compressor is mounted behind the engine where it is exposed to little or no air current, the radiation may be nearly negligible. If it is placed at the front of the engine and exposed to the full propeller blast, the radiation may be quite important, while if a specially designed air-to-air radiator is employed the radiation coefficient k may be made as large as desired, but at the expense of increased head resistance.

An accurate theoretical evaluation of k for any given installation is not possible, but some idea of its order of magnitude and of the extent of its variation with external conditions can be obtained from theoretical considerations.

Let us set ourselves the problem of computing an approximate value of k for a supercharging compressor which is placed in front of the engine with 4 square feet of radiating area exposed to the full velocity of the propeller slip stream. Let the aeroplane have a speed of 150 miles per hour at 18,000 feet altitude, and let the compressor deliver to the engine 700 cubic feet of air per minute at sea-level pressure and at the temperature T_2. (This is approximately the volume of air required by the Liberty engine at 1,700 revolutions per minute.)

An analysis of the recent radiator tests made at the Bureau of Standards shows that the coefficient of heat transfer from a radiating surface to a stream of air is given with considerable accuracy by the empirical equation [6]

$$C_h=29\left(\frac{\rho v}{10}\right)^{0.83}, \tag{7}$$

where C_h = coefficient of heat transfer in B. t. u. per square foot per degree Fahrenheit mean temperature difference per hour.

ρ = density of air in pounds per cubic foot.

v = air speed in feet per second.

This equation shows that the rate of heat transfer increases rapidly with the air speed and air density. Now in practice the speed and density of the air inside the compressor casing and discharge pipe will generally be a good deal larger than the speed and density outside the casing. Consequently it is to be expected that the mean temperature difference between the casing and the external air will be a good deal greater than the mean temperature difference between the compressed air and the casing. With this fact in mind, but without making a detailed computation of the rate of heat transfer from the compressed air to the casing, we make the arbitrary assumption that the mean temperature difference between the exposed surface of the casing and the external air is three-fourths of the net temperature rise, T_2-T_1. We take the air speed v to be the full air speed of the slip stream, or about 1.2 times the speed of advance of the plane. Thus

$$v=1.2\times150\times\frac{88}{60}=264 \text{ feet per second.}$$

[6] From data privately communicated to the writer by Mr. R. V. Kleinschmidt, formerly of the Bureau of Standards.

The relative density of the air (fig. 4) is 0.57, and the absolute density in pounds per cubic foot is 0.0436. Hence

$$C_h = 29\ (11.5)^{0.83} = 35.6.$$

The radiation coefficient is therefore

$$k = \frac{h}{T_2 - T_1} = \frac{4 \times 35.6 \times (3/4)}{60} = 1.78.$$

The weight of air flowing through the compressor in pounds per minute is

$$M = \frac{700 \times P_2}{RT_2},$$

where R is the gas content for air. Inserting the numerical values of R and P_2, we obtain

$$M = \frac{700 \times 144 \times 14.7}{53.3\ T_2} = \frac{27{,}800}{T_2}.$$

Hence equation (6) becomes

$$\mu = \frac{6{,}700}{6{,}700 + 1.78\ T_2}. \tag{8}$$

The carburetor pressure is exactly twice the intake pressure (see fig. 4), and the corresponding value of A (fig. 6) is 0.2218. Let the shaft efficiency of the engine be 0.64. Then (5) becomes

$$T_2 = T_1 \left[1 + \frac{0.2218}{0.64} \left(\frac{6{,}700}{6{,}700 + 1.78 T_2} \right) \right]. \tag{9}$$

The absolute intake temperature is $-5°$ F. Hence

$$T_1 = 460 - 5 = 455°,$$

and equation (9) is transformed into

$$T_2 = 455 + \frac{1{,}057{,}000}{6{,}700 + 1.78 T_2}.$$

The root of this equation is 591. Hence

$$\mu = \frac{6{,}700}{6{,}700 + 1.78 \times 591} = 0.864.$$

This value of μ will be used throughout the remainder of the present paper. The reader should bear in mind the fact, however, that μ will vary in practice with the installation and with the external conditions, i. e., with the values of v and M.

6. POWER ABSORBED BY COMPRESSOR.

Before making a specific application of the theory to a gear-driven compressor it is necessary to determine the power absorbed by the compressor. The theoretical input per pound of air is given by (2). To get the actual input we divide by the shaft efficiency. Thus

$$I = \frac{J C_p A (P_2/P_1) T_1}{E_s}. \tag{10}$$

The horsepower absorbed by the compressor is accordingly

$$H_c = \frac{J\ M C_p A (P_2/P_1) T_1}{33{,}000\ E_s}. \tag{11}$$

In order to evaluate the air flow M exactly, we introduce the following notation:

D = total piston displacement of engine in cubic feet;

ϵ = volumetric efficiency of engine;

N_e = engine speed in revolutions per minute of crank shaft;

and

ρ_2 = density of the air as it enters the carburetor.

Then

$$M=\frac{DN_e\epsilon\rho_2}{2}=\frac{DN_e\epsilon P_2}{2RT_2} \tag{12}$$

(11) now becomes

$$H_e=\frac{JC_pA(P_2/P_1)}{66{,}000\ E_s}\times\frac{DN_e\epsilon P_2}{R}\times\frac{T_1}{T_2}. \tag{13}$$

In order to apply the above formula, an estimate of the volumetric efficiency of the engine must be made. It is desirable in making supercharging calculations to have an experimental curve showing the relationship between volumetric efficiency and speed at sea level. Figure 3 shows such a curve for the Liberty engine. If experimental data are not available, a volumetric efficiency curve must be "fudged" with the aid of the curve for the brake, or, better, the indicated, mean effective pressure.

7. VARIATION OF VOLUMETRIC EFFICIENCY WITH INTAKE TEMPERATURE AND EXHAUST BACK PRESSURE.

Experiments recently made at the Bureau of Standards altitude laboratory, and privately communicated to the writer by Mr. S. W. Sparrow, show that the volumetric efficiency increases with the intake temperature and also with the ratio of the carburetor pressure to the exhaust back pressure. The ratio of the volumetric efficiency of the Hispano-Suiza 150-horsepower engine at +10° C. to that at −10° C. is 1.022. If the volumetric efficiency is assumed to be a linear function of the intake temperature, the following equation is easily deduced:

$$\epsilon=\epsilon_{59}+0.00054\ (t-59). \tag{14}$$

Here t is the intake temperature in degrees Fahrenheit, and ϵ_{59} is the volumetric efficiency for 59° F.

The experimental data available (see table below) on the variation in the volumetric efficiency with the ratio of the intake to the exhaust pressure are too meager to be of service for our present purpose without the help of theoretical considerations. We will therefore proceed to derive a theoretical formula containing one adjustable constant which can be fitted to the available experimental results.

When the intake pressure P_i exceeds the exhaust pressure P_e, the volumetric efficiency of the engine will be increased, owing to the fact that there is less residual exhaust gas left in the cylinder at the end of each exhaust stroke. When the inlet valve is opened, the residual exhaust gas will be compressed from the pressure P_e to the pressure P_i. The new volume of these gases being less than the volume of the compression space, the volume left to be filled by the incoming charge is greater than normally.

To a first order approximation, the mass of the charge which enters the cylinder is independent of the heat exchange which takes place between it and the residual exhaust gas. This is because the decrease in the density of the incoming charge due to heat absorption is offset by the increase in volume available due to the cooling and shrinkage of the residual gas. If it were not for the wiredrawing which occurs when the charge begins to enter the low pressure cylinder, we might compute the effective volume of the residual gas as if it were compressed adiabatically from the pressure P_i. On account of the wiredrawing the rise in temperature which accompanies the compression will be somewhat greater than for adiabatic compression. This can be taken into account by assuming polytropic compression with an appropriate index.

Let V_s = stroke volume of one cylinder;
V_c = compression volume of one cylinder;
r = compression ratio.

Then

$$V_c = \frac{V_s}{r-1}.$$

This is the volume occupied by the residual gas at the pressure P_e. The volume occupied at P_i will be

$$V_c\left(\frac{P_e}{P_i}\right)^{\frac{1}{m}} = \frac{V_s}{r-1}\left(\frac{P_e}{P_i}\right)^{\frac{1}{m}},$$

where m is the index of compression, which would be 1.4 if there were no wiredrawing. The volume to be filled by the incoming charge will then be

$$V_s + V_c - \frac{V_s}{r-1}\left(\frac{P_e}{P_i}\right)^{\frac{1}{m}} = V_s\left[\frac{r}{r-1} - \frac{1}{r-1}\left(\frac{P_e}{P_i}\right)^{\frac{1}{m}}\right].$$

But with equal intake and exhaust pressures this volume would be simply V_s. Hence the ratio of the volumetric efficiency, when P_e and P_i are different, to the normal volumetric efficiency is

$$\sigma = \left[\frac{r}{r-1} - \frac{1}{r-1}\left(\frac{P_e}{P_i}\right)^{\frac{1}{m}}\right]. \qquad (15)$$

The value of m might, perhaps, be computed theoretically, but it is easier to treat it as a constant to be determined empirically. The accompanying table shows the results of a short series of tests on the variation of the volumetric efficiency of a Hispano-Suiza engine with a 7½ to 1 compression ratio. In the fourth column are tabulated the theoretical values of σ as computed from (15), using 2 for the value of m. The agreement is within the limits of experimental error and establishes 2 as an approximate and convenient value for the index m.

TABLE No. 1.

P_i.	$P_i - P_e$.	Relative air flow.		Remarks.
Inches of mercury.		Exp.	Theor.	
25	1.73	1.005	1.006	Engine driven by dynamometer.
25	−1.73	.99	.995	Do.
17.5	1.73	1.01	1.008	Do.
17.5	−1.73	.99	.992	Do.
14.5	1.73	1.005	1.01	Do.
14.5	−1.73	.995	.991	Do.
14.5	1.73	1.015	1.01	Engine driven by own power.
14.5	−1.73	.991	.991	Do.

In the case of an engine fitted with a gear-driven centrifugal compressor, P_i is equal to the compressor exhaust pressure, P_2. Hence (15) may be rewritten in the form

$$\sigma = \left[\frac{r}{r-1} - \frac{1}{r-1}\sqrt{\frac{P_e}{P_i}}\right]. \qquad (15a)$$

In dealing with turbine-driven compressors we shall use equation (14) for the volumetric efficiency. For gear-driven compressors this is to be replaced by

$$\epsilon = \sigma\,[\epsilon_{59} + 0.00054\,(t-59)]. \qquad (16)$$

8. APPLICATION TO LIBERTY ENGINE, GEAR-DRIVEN COMPRESSOR.

Let us assume a gear-driven centrifugal compressor having a maximum shaft efficiency of 64 per cent and capable of doubling the carburetor pressure of the Liberty engine when engine and compressor are working at their normal speeds. We inquire regarding: (*a*) The volume of air which the compressor must be designed to handle; (*b*) the horsepower required to drive the compressor under normal conditions; and (*c*) the net gain in power of the engine when coupled with the compressor at the "normal altitude."

(*a*) The conditions are the same as those used in determining the value of k (cf. latter part of art. 5). The normal pressure ratio being 2:1, the normal altitude must be that for which the barometric pressure has half its sea level value, i. e., 18,000 feet. The corresponding intake temperature is $-5°$ F., or 455° absolute (cf. fig. 4.) The value of $A(P_2/P_1)$ is given by figure 6. Inserting numerical values into equation (5), we obtain 136° as the net temperature

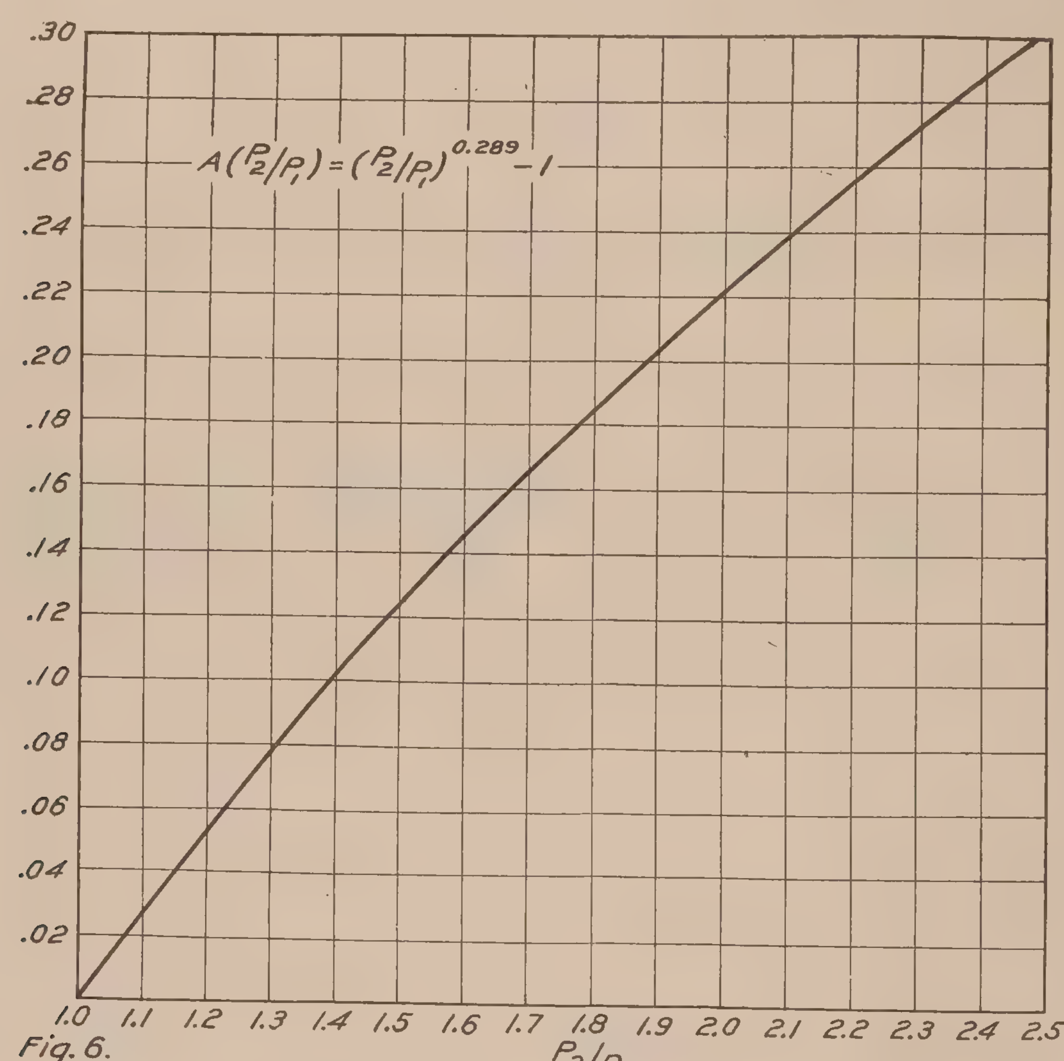

Fig. 6.

rise in the compressor. Hence the temperature of the air entering the carburetor is 131° F., or 591° absolute. The normal speed of the Liberty engine is 1,700 revolutions per minute and its piston displacement is 0.96 cubic foot. With the aid of these numerical values we reduce (12) to the form

$$M=\frac{0.96\times1700\times14.7\times144\epsilon}{2\times53.3\times591}=54.8\epsilon.$$

The normal value of the volumetric efficiency at 1,700 revolutions per minute, and with a carburetor temperature of about 59° F., is 0.85. The compression ratio of the engine is 5.42. Hence the corrected volumetric efficiency for 131° F. and a 2 to 1 pressure ratio is (cf. equation (16)):

$$\begin{aligned}\epsilon&=(0.85+0.00054\times72)\ (1.226-0.226\ \sqrt{\tfrac{1}{2}}),\\&=0.95\end{aligned}$$

The corresponding value of M is 52 pounds per minute. This is a little greater than the value given by the approximate formula (7).

Let Q denote the intake volume for the compressor in cubic feet per minute. Q is equal to M divided by the density of the air entering the compressor. Figure 4 gives 0.57 as the relative density at 18,000 feet. The corresponding absolute density is 0.0436 pounds per cubic foot. Hence

$$Q=\frac{52}{0.0436}=1{,}193 \text{ cubic feet per minute.}$$

The above intake volume, the normal speed, and normal pressure ratio are the three fundamental quantities which determine the design of the compressor.

(*b*) All the quantities which enter into the right-hand member of (13) are now known. The numerical value of H_c computed from this formula is 46.5 horsepower.

(*c*) Let H_G denote the gross horsepower developed by the engine with the intake pressure and temperature P_2 and T_2, respectively, and with the exhaust pressure P_1. The temperature correction factor, equation (1), is

$$F_t=\frac{920+59}{920+131}=0.931.$$

The pressure correction factor for a 76 cm. intake pressure and a 38 cm. exhaust pressure is 1.06. (See fig. 2.) The normal power developed by the engine at 1,700 revolutions per minute is 423. Hence the value of H_G is

$$H_G=423\times1.06\times0.931=417.5 \text{ horsepower.}$$

Subtracting the power required to drive the compressor, we obtain the net power available for driving the propeller, which is

$$H=417.5-46.5=371 \text{ horsepower.}$$

9. TURBINE-DRIVEN COMPRESSOR.

As previously stated, we assume equal intake and exhaust pressures for the turbine-driven compressor. Under the normal working conditions just considered, these pressures will have the common value 76 cm. The pressure correction factor drops out, and the net power becomes equal to the gross power. Thus

$$H=H_G=423\times0.931=394 \text{ horsepower.}$$

Since the pressure correction factor for the volumetric efficiency drops out, the normal volume of air which the turbine-driven compressor must be designed to handle is a little less than that for the gear-driven compressor. The numerical value is

$$Q=1{,}115 \text{ cubic feet per minute.}$$

We may estimate the weights of the gear-driven and turbine-driven compressors at 75 and 100 pounds, respectively. The weight of the engine without the compressor, dry, is 844 pounds. Hence the weight per horsepower at 18,000 feet with the compressor, works out to be 2.48 pounds per horsepower and 2.40 pounds per horsepower, for the gear-driven and turbine-driven jobs, respectively. The weight per horsepower without the compressor at this altitude is 4.5. Thus the reduction in the weight per horsepower ratio due to the compressor is 44.9 per cent and 45.7 per cent for the gear-driven and turbine-driven jobs, respectively.

10. CALCULATION OF PERFORMANCE CURVES: COMPRESSOR THEORY.

In order to extend the calculation to other than normal conditions and to draw up performance curves for the engine compressor unit at all altitudes and speeds, it is necessary to make use of the characteristic curves for the shaft and hydraulic efficiencies of the compressor.[7]

[7] See article on Centrifugal Compressors by Dr. L. C. Lœwenstein in Marks's "Mechanical Engineers' Handbook."

Experiment shows that these efficiencies depend primarily on the ratio of the volume of intake air to the speed of the compressor. This ratio is called the quantity coefficient, and will be designated by the symbol q.

$$q=Q/N.$$

It is convenient to plot the efficiencies as ordinates against values of q as abscissae, as in the characteristic curves shown in figure 7.

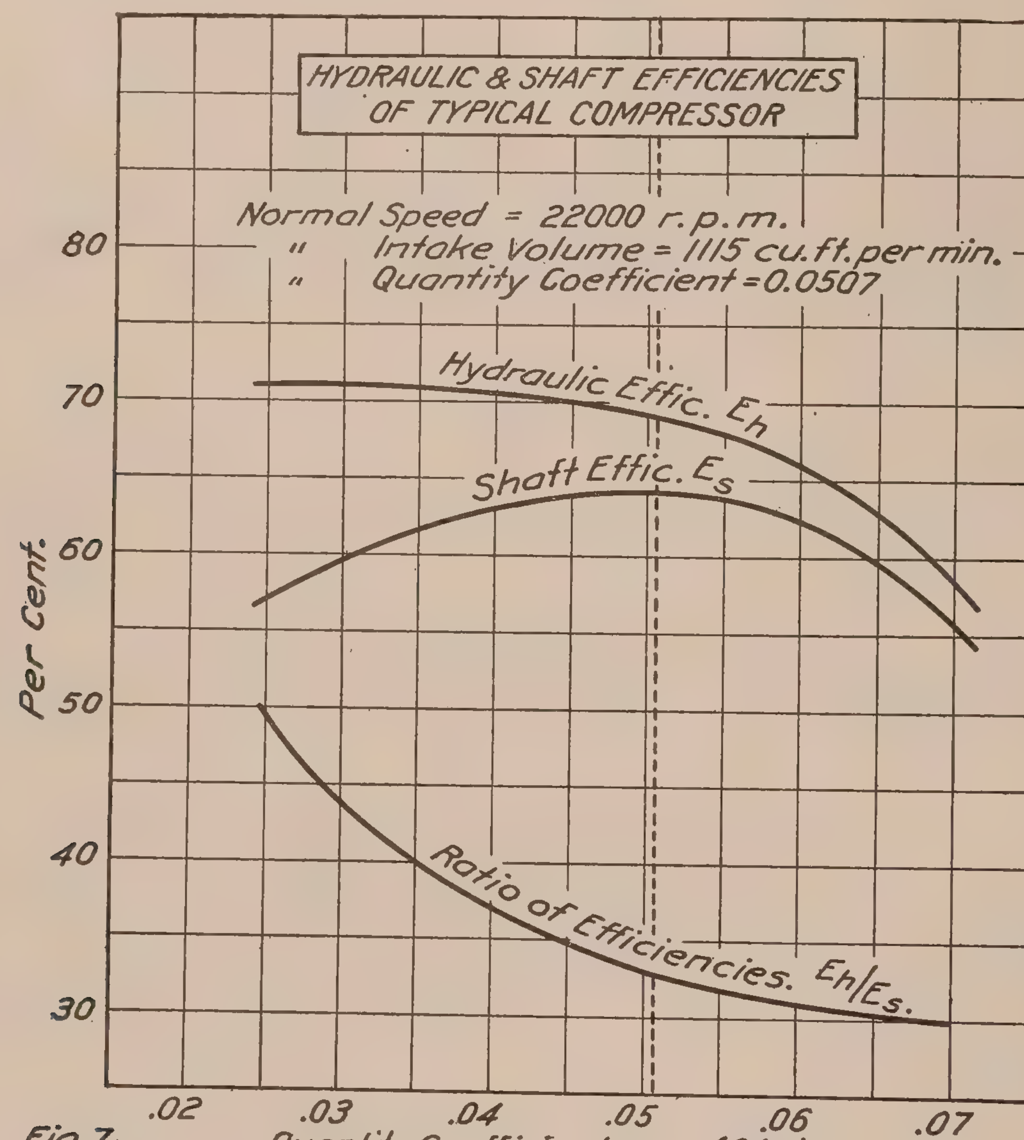

Fig. 7.

The fundamental formula which determines the pressure rise is

$$A(P_2/P_1)=\frac{E_h(q)}{g}\left(\frac{\gamma-1}{\gamma}\right)\frac{u_a^2}{RT_1} \tag{17}$$

where $E_h(q)$ = hydraulic efficiency;
g = acceleration of gravity;
u_a = peripheral speed of compressor impeller in feet per second.

The other symbols have already been defined. Since the peripheral speed is proportional to N, the above equation can be rewritten as

$$A(P_2/P_1)=\frac{\alpha E_h(q)N^2}{T_1}, \tag{18}$$

where α is a constant for any given compressor. Solving for P_2/P_1, we obtain

$$P_2/P_1=\left[1+\frac{\alpha E_h(q)N^2}{T_1}\right]^{3.46} \tag{19}$$

In order to use the above equation for the determination of P_2/P_1, or to use (5) to evaluate the carburetor temperature, it is necessary to calculate the value of the quantity coefficient, q. To this end we divide (12) by $N\rho$, where ρ_1 denotes the density of the air entering the compressor, and obtain

$$q=\frac{M}{\rho_1 N}=\frac{D}{2}\frac{N_e}{N}\frac{P_2}{P_1}\frac{T_1}{T_2}\epsilon. \tag{20}$$

The right-hand member involves unknown quantities which must be eliminated before (20) can be solved for q.

In carrying out this elimination we have two cases to consider. (I) In the case of a turbine-driven compressor working at an altitude below its normal altitude of operation, the speed of the compressor will be adjusted to give sea-level carburetor pressure. In this case, the intake pressure being known, the pressure ratio is known, and the speed N must be eliminated from equations (18) and (20) in order to solve for q. (II) In the case of a gear-driven compressor operating at any altitude, or of a turbine-driven compressor operating above the normal altitude, the speed of the compressor is determined either by the gear ratio, or by the maximum safe speed of the compressor, and the pressure ratio is the unknown quantity to be eliminated from equations (18) and (20).

11. CASE I. TURBINE-DRIVEN COMPRESSOR: PRESSURE RATIO GIVEN.

Consider first the case where the pressure ratio is known. Eliminating N between (18) and (20), we obtain

$$q=\frac{DN_e}{2}\frac{P_2}{P_1}\frac{T}{T_2}\epsilon\sqrt{\frac{\alpha E_h(q)}{AT_1}}.$$

Eliminating $\frac{T_1}{T_2}$ by means of (5) the above becomes

$$q=\frac{DN_e}{2}\frac{P_2}{P_1}\epsilon\sqrt{\frac{\alpha E_h(q)}{AT_1}}\frac{1}{\left[1+\frac{\mu A}{E_s(q)}\right]}$$

or

$$\frac{2\sqrt{AT_1}\,P_1}{DN_e\epsilon\sqrt{\alpha}P_2}=\frac{E_s(q)\sqrt{E_h(q)}}{q[E_s(q)+\mu A(P_2/P_1)]}. \tag{21}$$

The right-hand side is a function of q and (P_2/P_1). The left-hand side is sensibly independent of q, and its value is easily calculated. In computing the value of ϵ, we combine (16) with (5), and obtain

$$\epsilon=\sigma\left[\epsilon_{59}+0.00054\{T_1\left(1+\frac{\mu A}{E_s}\right)-519\}\right] \tag{22}$$

In this equation E_s may be treated as a constant without serious error.

It is convenient to introduce the notation

$$\phi(q, P_2/P_1)=\frac{E_s(q)\sqrt{E_h(q)}}{q[E_s(q)+\mu A(P_2/P_1)]}. \tag{23}$$

$$\psi(P_2/P_1, T_1, N_e)=\frac{2\sqrt{A(P_2/P_1)}\,T_1}{\sqrt{\alpha}DN_e\epsilon}\frac{P_1}{P_2} \tag{24}$$

Equation (21) then becomes

$$\psi\left(\frac{P_2}{P_1}, T_1, N_e\right)=\phi\left(q, \frac{P_2}{P_1}\right) \tag{25}$$

To solve this equation, a set of curves showing ϕ as a function of q for various constant values of P_2/P_1, may be drawn, as in figure 8. The value of ψ can be computed directly from (24). To find q we simply follow the line $y=\psi$ horizontally across to the point which corresponds to the appropriate value of P_2/P_1 and note the corresponding abscissa.

Having calculated the quantity coefficient in this manner, the shaft efficiency can be found from figure 7, and equation (5) employed to determine the temperature at the carburetor.

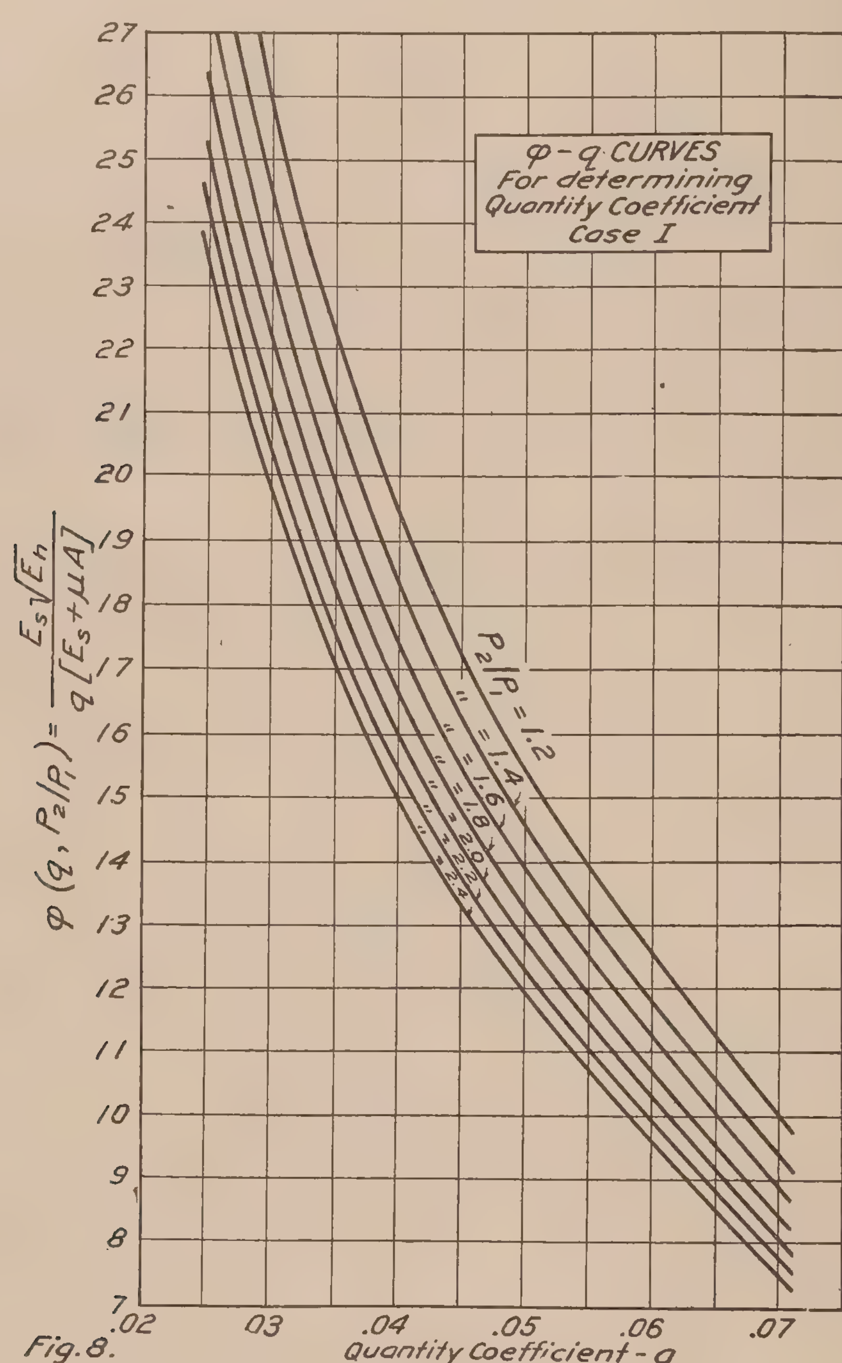

Fig. 8.

The remainder of the computation for the horsepower of the engine-compressor unit is similar to that already carried out for the normal conditions of operation in article 8.

12. NUMERICAL APPLICATION.

To illustrate with a numerical example, we assume the turbine-driven engine compressor unit of the problem of article 8. The constant α of equation (18) can be computed from the data already assumed for normal operation at 18,000 feet altitude. The normal speed of the compressor is 22,000 revolutions per minute. The normal intake volume is 1,115 cubic feet per minute. Hence the normal value of the quantity coefficient is 0.0507 and the normal hydraulic efficiency is 0.69. The normal pressure ratio is 2, and the corresponding value of A (fig. 6) is 0.2218. The normal absolute intake temperature (fig. 4) is 455°. Solving (18) for α, we obtain

$$\alpha = \frac{AT_1}{E_h N^2} = \frac{0.2218 \times 455}{0.69 \times (22{,}000)^2} = 0.303 \times 10^{-6}.$$

Inserting the numerical values of α and D in (24), we obtain

$$\psi=\frac{3.788\sqrt{AT_1}}{N_e\epsilon}\cdot\frac{P_1}{P_2} \tag{24'}$$

Giving E_s the constant value 0.64, putting σ equal to unity, and giving μ the value previously computed, viz, 0.864, we reduce (22) to the form

$$\epsilon=\epsilon_{59}+0.00054[T_1(1+1.35A)-519]. \tag{25}$$

Figure 8 shows the ϕ—q curves plotted from (23) for $\mu=0.864$.

Let us apply these formulas and curves to the problem of the determination of the power delivered by the engine and turbine-driven compressor at 1,800 revolutions per minute and an altitude of 10,000 feet.

The intake pressure (fig. 4) is 52.1 cm. of mercury, and the pressure ratio required to give sea-level carburetor pressure is accordingly

$$P_2/P_1=76/52.1=1.46.$$

From figure 6

$$A\left(\frac{P_2}{P_1}\right)=0.1156.$$

The volumetric efficiency at 59° F. and 1,800 revolutions per minute (fig. 3) is 0.83, and the compressor intake absolute temperature at 10,000 feet is 483.4°. Then by (25)

$$\epsilon=0.83+0.00054[483(1+1.35\times 0.1156)-519]=0.8505.$$

Equation (24′) yields

$$\psi=\frac{3.788\sqrt{0.1156\times 483}}{1800\times 1.46\times 0.8505}=12.67.$$

The corresponding value of q (fig. 8) is 0.0561. The compressor speed, equation (18), is 16,810 revolutions per minute. Its shaft efficiency (fig. 7) is 0.6345, and the absolute carburetor temperature, equation (5), is

$$T_2=483(1+0.864\times 0.1156/0.6345)=559°.$$

Hence the temperature correction factor, equation (1), is

$$F_t=\frac{920+59}{920+99}=0.961.$$

The sea-level horsepower at this speed is 445. Hence the power at 1,800 revolutions per minute and 10,000 feet altitude is

$$H=0.961\times 445=427.5.$$

13. CASE II. TURBINE-DRIVEN COMPRESSOR: ROTATIONAL SPEED GIVEN.

Consider next the case where the compressor speed is known and the pressure ratio is unknown. In this case P_2/P_1 must be eliminated from (20) by means of (19). Then

$$q=\frac{D}{2}\frac{N_e}{N}\frac{T_1}{T_2}\epsilon\left[1+\alpha\, E_h(q)\frac{N^2}{T_1}\right]^{3.46}. \tag{26}$$

Combining equations (5) and (18), we obtain

$$T_2=T_1\left[1+\mu\,\alpha\frac{E_h(q)}{E_s(q)}\frac{N^2}{T_1}\right]. \tag{27}$$

Eliminating T_1/T_2 from (26) by means of (27) leads to the following equation:

$$q=\frac{D}{2}\frac{N_e}{N}\epsilon\frac{\left[1+\alpha E_h(q)\frac{N^2}{T_1}\right]^{3.46}}{\left[1+\mu\alpha\frac{E_h(q)}{E_s(q)}\frac{N^2}{T_1}\right]}. \tag{28}$$

In order to calculate ϵ, we combine (16) with (27), and obtain

$$\epsilon=\sigma\left[\epsilon_{59}+0.00054\left(T_1-519+\mu\alpha N^2\frac{E_h}{E_s}\right)\right]. \tag{29}$$

Here again it is sufficiently accurate to treat E_h/E_s as a constant. A further simplification results from the fact that in the case of the turbine-driven compressor, now under consideration, σ is always unity. It is convenient to introduce the notation

$$x\left(q,\frac{N^2}{T_1}\right)=\frac{\left[1+\alpha E_h(q)\frac{N^2}{T_1}\right]^{3.46}}{q\left[1+\mu\alpha\frac{E_h(q)}{E_s(q)}\frac{N^2}{T_1}\right]}; \tag{30}$$

$$\zeta(N_e,N,T_1)=\frac{2N}{DN_e\left[\epsilon_{59}+0.00054\left(T_1-519+\mu\alpha N^2\frac{E_h}{E_s}\right)\right]}. \tag{31}$$

Then (28) reduces to

$$\zeta(N_e,N,T_1)=\chi\left(q,\frac{N^2}{T_1}\right). \tag{32}$$

To solve this equation a set of curves showing χ as a function of q for various constant values of N^2/T_1 is drawn up, as on figure 9. The value of ζ for any particular case can be computed directly from (31), since ζ does not involve q. To find q, we proceed as in Case I. Follow the line $y=\zeta$ horizontally across to the point which corresponds to the appropriate value of N^2/T_1 and note the corresponding abscissa.

When q is known, the pressure ratio can be found at once from (18) and figure 6. Combining equations (5) and (18), we obtain

$$T_2=T_1+\mu\alpha N^2\frac{E_h(q)}{E_s(q)}. \tag{33}$$

Equation (33) serves for the determination of the temperature rise, and the calculation of the horsepower is carried through as before.

14. NUMERICAL APPLICATION OF THEORY FOR CASE II.

We illustrate the above theory with a numerical example. Consider the operation of the engine-compressor unit of the preceding examples above the altitude of normal operation. For maximum power the compressor will operate at its maximum safe speed, 22,000 revolutions per minute, under all conditions. The values of μ and α are 0.864 and 0.303×10^{-6}, respectively. Figure 9 shows the $\chi-q$ curves obtained from (30). Giving E_h/E_s the value 1.077, equation (31) reduces to

$$\zeta=\frac{45,800}{N_e[\epsilon_{59}+0.00054\,(T_1-383.5)]}. \tag{34}$$

Equation (33) becomes

$$T_2=T_1+126.7\,E_h(q)/E_s(q), \tag{35}$$

and (18) takes the form

$$A=146.8\,E_h(q)/T_1. \tag{36}$$

Let us use the above equations to evaluate the power delivered by the engine at 40,000 feet and 1,900 revolutions per minute. From figure 4 the temperature T_1 is 376.5°. The value of ζ, equation (34), is 28.95. N^2/T_1 is 1.285×10^6. Hence the quantity coefficient, q, (fig. 9) is 0.0554. E_h/E_s (fig. 7) is 1.066, and the carburetor temperature, equation (35), is 495° absolute. Th temperature correction factor, equation (1), is 1.025. The hydraulic efficiency

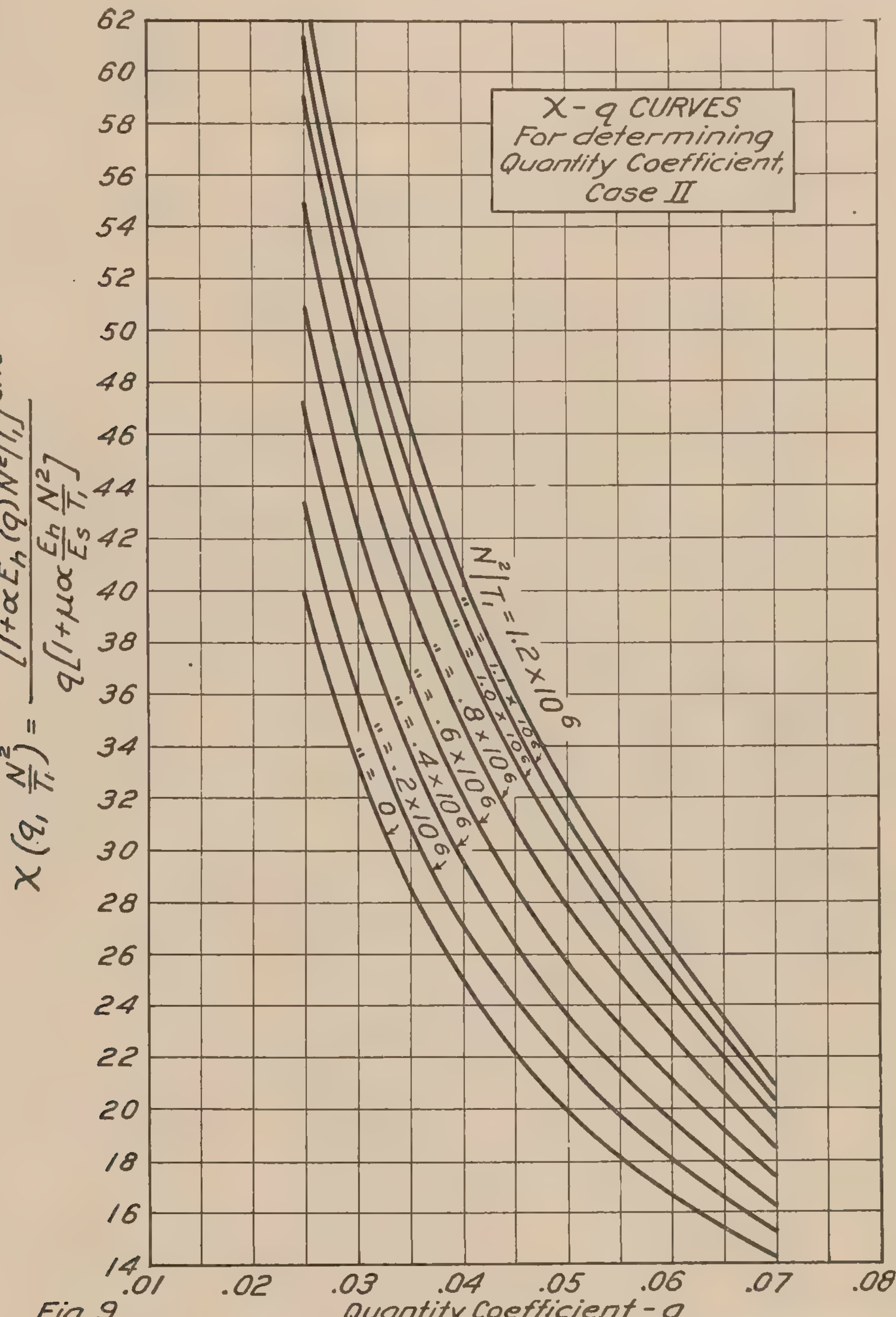

Fig. 9.

(fig. 7) is 0.678, and the corresponding value of A, equation (36), is 0.2645. The pressure ratio (fig. 6) is 2.251. The compressor intake pressure (fig. 4) is 14 cm. of mercury, and the carburetor pressure is accordingly 31.5 cm. of mercury. The mechanical efficiency at 76 cm. and 1,900 revolutions per minute is 0.855 (fig. 3), and the pressure correction factor (fig. 1) is 0.315. The sea-level horsepower is 453, and consequently the horsepower under the conditions assumed is

$$H = 453 \times 1.025 \times 0.315 = 146.$$

Complete performance curves for the Liberty engine fitted with a turbine-driven compressor, and worked out in the above manner, are shown on figure 10.

15. THE GEAR-DRIVEN COMPRESSOR.

The great advantage of the gear-driven type of compressor is that by its use the engine can be made to develop a high power at great altitudes with a low exhaust pressure. Such a low exhaust pressure involves a correspondingly low exhaust temperature and should materially

increase the life of the exhaust valves, which would be comparatively short in an engine equipped with a turbine-driven compressor and operated continuously with sea-level intake and exhaust pressures. It also seems not improbable that if the mechanical problem of designing a slipping clutch which will take excessive acceleration stresses from the gears can be solved, the gear-driven type of compressor will prove the more durable of the two.

On the other hand, the computation of article 9 predicts that the net power available per unit weight from a given engine operating with a gear-driven compressor at its normal altitude is about 3 per cent less than the net power available from the same engine under the same conditions, with a turbine-driven compressor. If this result be accepted as correct, the case for the gear-driven compressor would seem to be a poor one.

The simplest mechanical arrangement for a gear-driven compressor involves a single set of gears and a constant value for the ratio of the compressor speed to the engine crank-shaft speed.

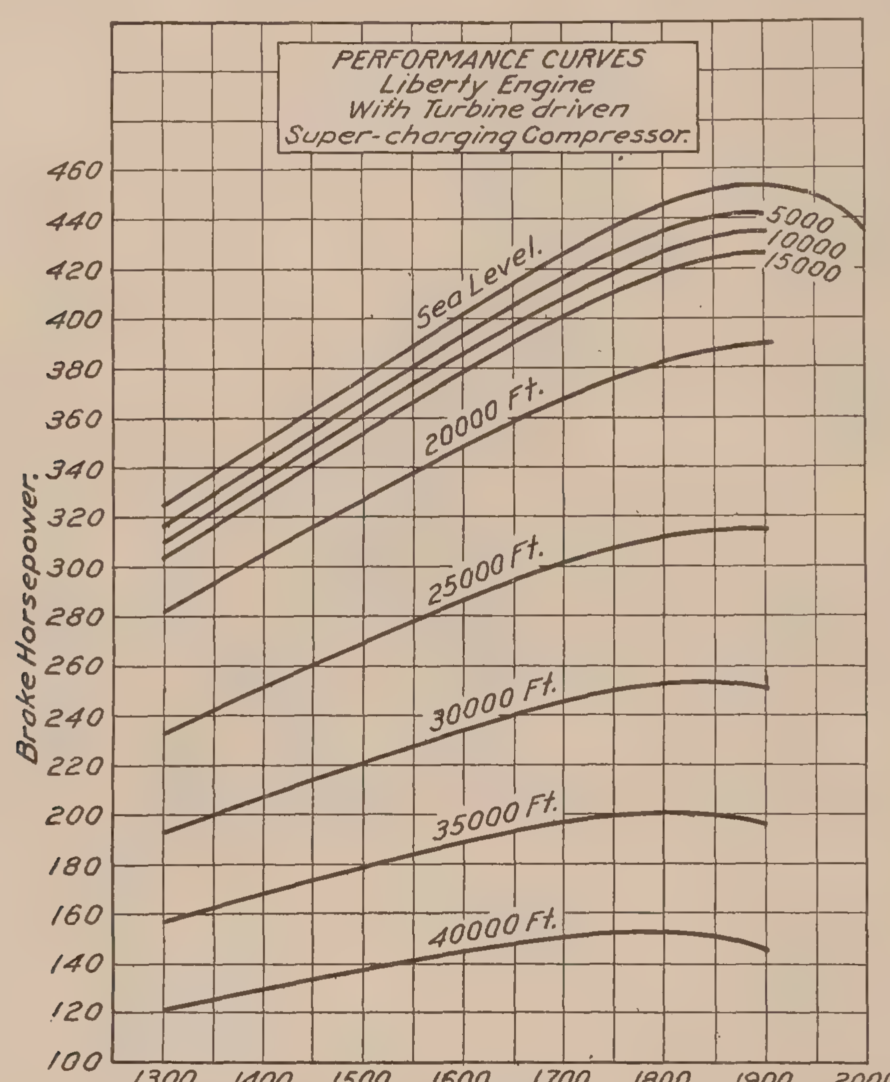

Fig. 10.

This lack of flexibility is an additional distinct disadvantage which might possibly be overcome, in part, by the use of a two-speed gear, or through the use of a constantly slipping clutch. The mechanical difficulties involved in these forms of speed control are great, however, and in view of the proved feasibility of the turbine-driven compressor, neither arrangement will be considered here.

In order to prevent the carburetor pressure from rising above sea-level value at low altitudes with a gear driven compressor, it is necessary to disconnect the compressor entirely, or to throttle the air at the inlet to the compressor. In our computation of performance curves, it will be assumed that means are provided for inlet throttling at moderate altitudes. The horsepower-altitude curve for each speed is then divided into three parts, viz, a portion *ab* (see fig. 11) for the lowest altitudes where the compressor can not be used to advantage at all, and is assumed to be disconnected, a portion *bc* over which the compressor is assumed to be throttled at the inlet in such a manner as to maintain the carburetor pressure at the constant value 76 cm., and

finally, a portion *cd* for the highest altitudes, where no inlet throttling is necessary, and where the carburetor pressure is less than at sea level.

Let us consider, first, the region over which inlet throttling is necessary. It will be assumed that the kinetic energy developed by the air as it passes through the throttle valve is immediately converted into heat. Then the temperature of the air as it enters the compressor will be sensibly equal to the temperature of the external atmosphere. The carburetor pressure is known, but the pressure of the throttled air P_1 is not, so that the pressure ratio must be treated as an unknown quantity. Since the speed of the compressor is known the method of computing the quantity coefficient is practically the same as for the turbine-driven compressor above the normal

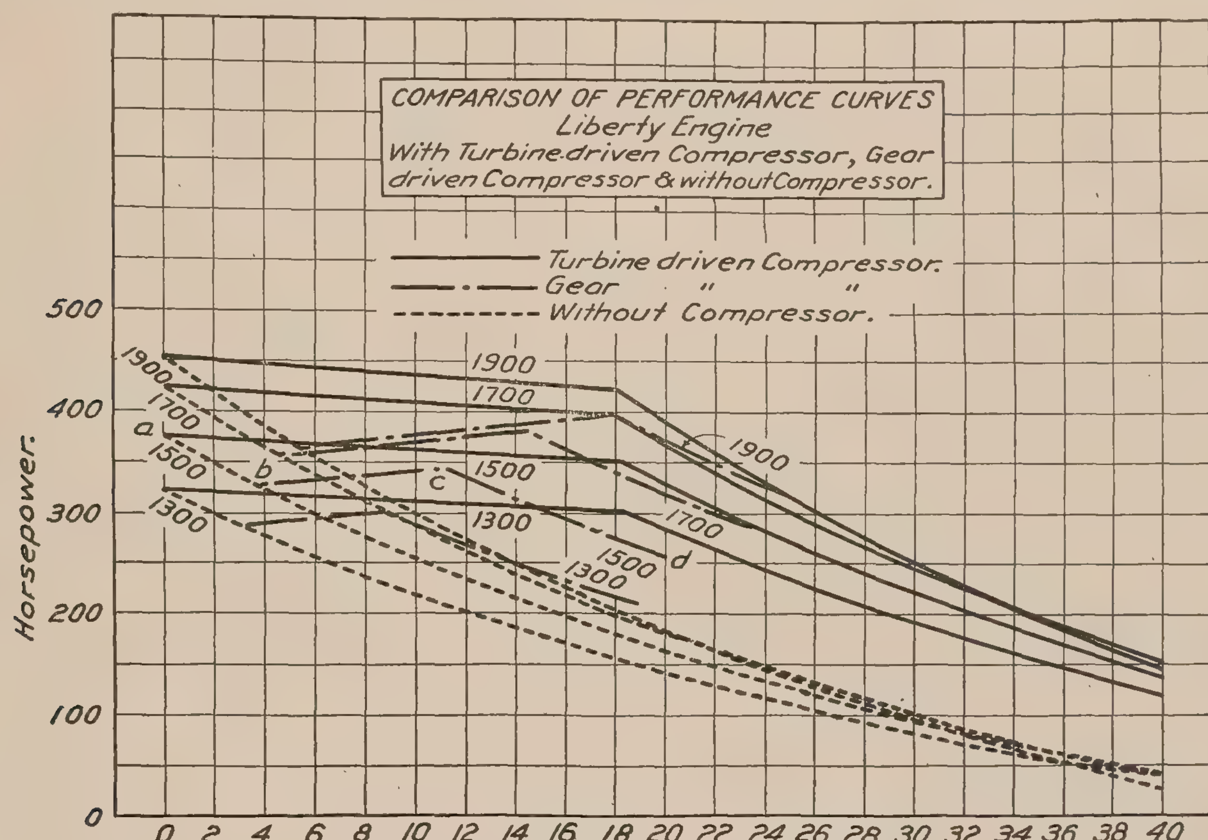

Fig. 11.

altitude of operation. The one difference is that the quantity σ, which gives the pressure correction to the volumetric efficiency, does not reduce to unity. Equation (32), which was used for the determination of the quantity coefficient q, is replaced by

$$\zeta_1 = \chi\left(q, \frac{N^2}{T_1}\right) \tag{37}$$

where

$$\zeta_1 = \frac{\zeta(N_e, N, T_1)}{\sigma}. \tag{38}$$

The value of σ is to be computed from (15a), P_e being identified with the pressure of the external atmosphere. When q is evaluated by means of (37) and a $\chi - q$ chart, such as figure 9, the compressor efficiencies are determined, and equation (18) is used to calculate the carburetor temperature. The temperature correction factor and the pressure correction factor are taken from equation (1) and figure 2, respectively.

The product of the normal sea-level power into the temperature and pressure correction factors is the gross horsepower developed by the engine. From this must be subtracted the power absorbed by the compressor, which may be computed from the following equation, derived by combining (13) and (18):

$$H_c = \frac{J C_p D \alpha N^2 N_e}{66{,}000 R} \frac{E_h}{E_s} \frac{\epsilon P_2}{T_2}. \tag{39}$$

In computing the performance curves for the region where inlet throttling is *not* necessary, P_2 is an unknown, and P_e is to be identified with P_1. σ can not be determined until the value of q is known. We therefore substitute from (19) into (15a) and obtain the following expression for σ in terms of q and N^2/T_1:

$$\sigma=\frac{r}{r-1}-\frac{1}{r-1}\left[1+\alpha E_h(q)\frac{N^2}{T_1}\right]^{-1.73}. \tag{40}$$

Equation (32) is replaced by

$$\zeta=\chi_1\left(q,\frac{N^2}{T_1}\right), \tag{41}$$

where

$$\chi_1\left(q,\frac{N^2}{T_1}\right)\equiv\chi\left(q,\frac{N^2}{T_1}\right)\left[\frac{r}{r-1}-\frac{1}{r-1}\left(1+\alpha E_h(q)\frac{N^2}{T_1}\right)^{-1.73}\right]. \tag{42}$$

The computation of q and of the temperature correction factor then goes through as in the preceding case. The pressure ratio is determined from figure 6 and equation (18). Figure 2 is used to determine the pressure correction factor, and the gross horsepower, compressor horsepower, and net horsepower computed as before.

Figure 11 shows a set of predicted performance curves for the Liberty engine equipped with a gear-driven compressor, calculated by the method described above. In arriving at this set of curves (dot-and-dash lines) it was assumed that gear ratio was 11.6:1, making the compressor speed 22,000 revolutions per minute (maximum safe speed) when the engine runs at 1,900 revolutions per minute. For convenience in computation the same compressor was assumed as in the calculations for the turbine-driven job. As we have already shown (Art. 9) that the gear-driven compressor should have a slightly larger volume capacity than the turbine-driven compressor for normal operation on the same engine, it is evident that our method of procedure involves a slight handicap to the gear-driven job, when operating at its highest speed. This handicap, which consists in assuming compressor efficiencies which are somewhat smaller than the maximum obtainable efficiencies, is small, however, compared with the differences between the outputs of the gear driven and turbine driven arrangements, and does not materially affect the relative merits of the two schemes. For comparison the performance curves for the turbine-driven supercharging unit and for the engine without the compressor are also shown on figure 11. The curves of the gear-driven compressor are not carried to very great altitudes because the pressure correction factors involved lie outside the chart of figure 2.

Figure 11 shows that at low speeds for all altitudes, and at all speeds for medium altitudes, the gear-driven compressor without speed control produces much less power than the turbine-driven compressor. In view of this fact, and of the mechanical difficulties involved in the gear drive, the turbine-driven compressors alone will be considered in the remainder of this report.

16. FUEL CONSUMPTION.

The formulas for the temperature and pressure correction factors which we have used are based on tests in which the carburetor was adjusted for the maximum fuel economy consistent with maximum power. The corresponding fuel consumption curves must accordingly presuppose carburetor adjustment for maximum power at all altitudes. The investigations of Mr. P. S. Tice [8] at the Bureau of Standards show that the mixture ratio which gives maximum power is independent of the barometric pressure, and we therefore base our computation of fuel economy on the assumption of a constant air-fuel ratio. Power-altitudes and fuel-economy curves based on the assumption that the carburetor is adjusted for maximum fuel economy would be of value, but data for their computation are not available at present.

Let the subscript s indicate quantities pertaining to operation under standard sea level conditions (76 cm. pressure and 59° F.). The relative fuel consumption for a constant mixture ratio

[8] "Carbureting Conditions Characteristic of Aircraft Engines," by P.S.Tice, Report No. 48, National Advisory Committee for Aeronautics, 1920.

is equal to the quotient of the relative air flow divided by the relative horsepower output. Hence the relative fuel consumption is

$$r.f.c.=\frac{H_s}{H}\frac{N_e}{(N_e)_s}\frac{\epsilon}{\epsilon_s}\frac{P_2}{P_s}\frac{T_s}{T_2}, \tag{43}$$

where P_2 is the carburetor pressure in centimeters of mercury and T_2 is the absolute temperature of the air entering the carburetor.

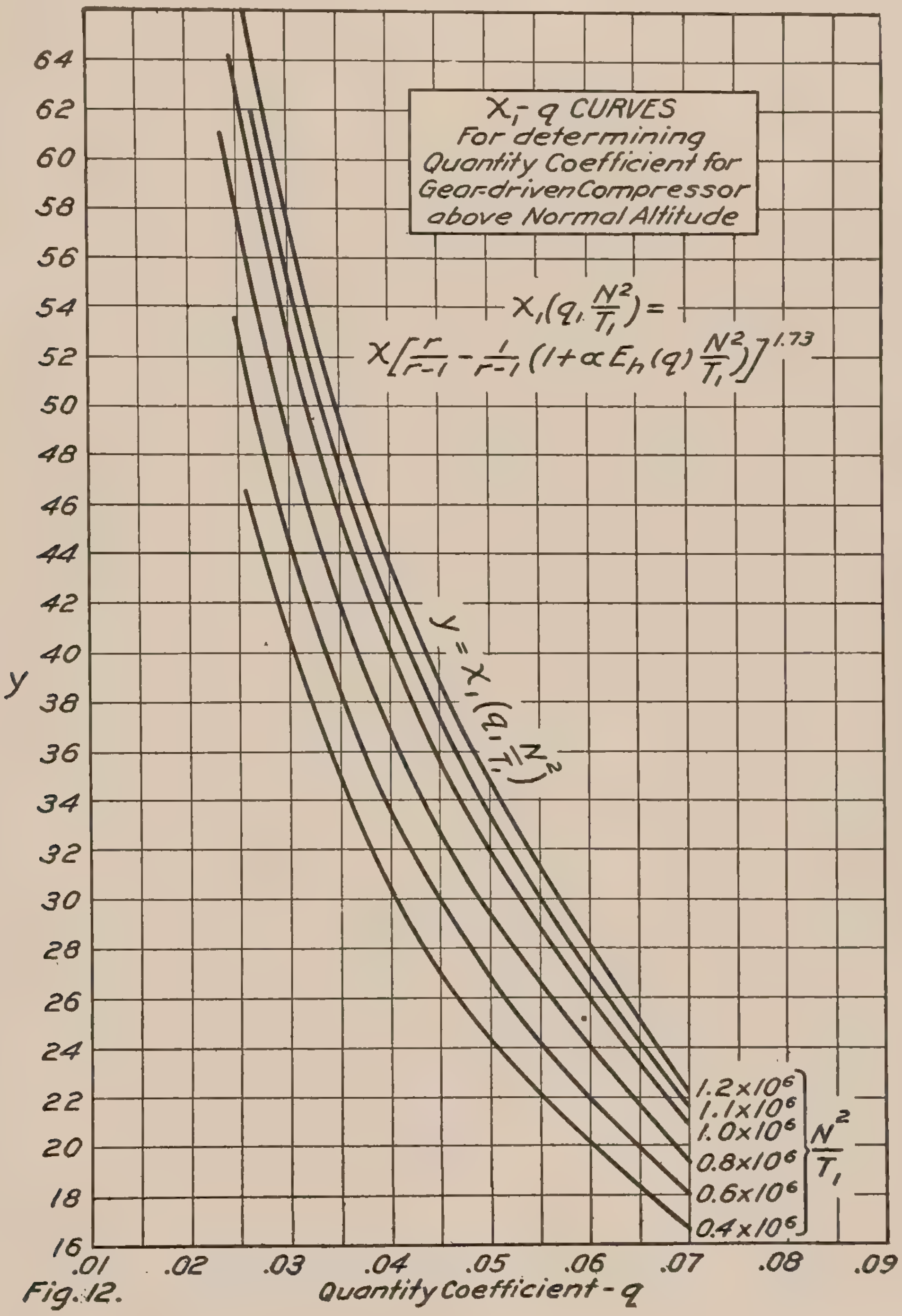

Fig. 12.

In some cases it is desirable to refer the fuel consumption for any given speed under high altitude conditions to the fuel consumption at the same speed on the ground. Then (43) becomes

$$r.f.c.=\frac{H_s}{H}\frac{\epsilon}{\epsilon_s}\frac{P_2}{P_s}\frac{T_s}{T_2}. \tag{44}$$

As a check on the above formula we insert Table 2, which gives a comparison of computed and experimental values of the relative fuel consumption at low pressures. The experimental values are taken from Report No. 46 ("A Study of Airplane Engine Tests" by Victor R. Gage). The temperature was the same for all the tests and the intake and exhaust pressures were in all cases equal. Hence the factors T_s/T_2 and $\frac{\epsilon}{\epsilon_s}$ reduce to unity and drop out of the computation.

TABLE 2—*Computed and experimental values of relative fuel consumption of Hispano-Suiza 150 horsepower engine at low pressure.*

Engine speed.	Ground.		10,000 feet.		20,000 feet.		30,000 feet.	
	Computed.	Experimental.	Computed.	Experimental.	Computed.	Experimental.	Computed.	Experimental.
1,300	1.00	1.00	1.05	1.025	1.10	1.087	1.21	1.26
1,500	1.00	1.00	1.05	1.04	1.105	1.091	1.19	1.27
1,700	1.00	1.00	1.05	1.03	1.10	1.091	1.20	1.213
1,900	1.00	1.00	1.045	1.043	1.10	1.122	1.233	1.269
2,100	1.00	1.00	1.06	1.047	1.14	1.14	1.284	1.314

The very appreciable discrepancies in the above table are presumably to be attributed to the uncertainty of a carburetor setting for maximum power. It will be observed that the tendency is for the experimental values to exceed the theoretical ones for the lowest pressures

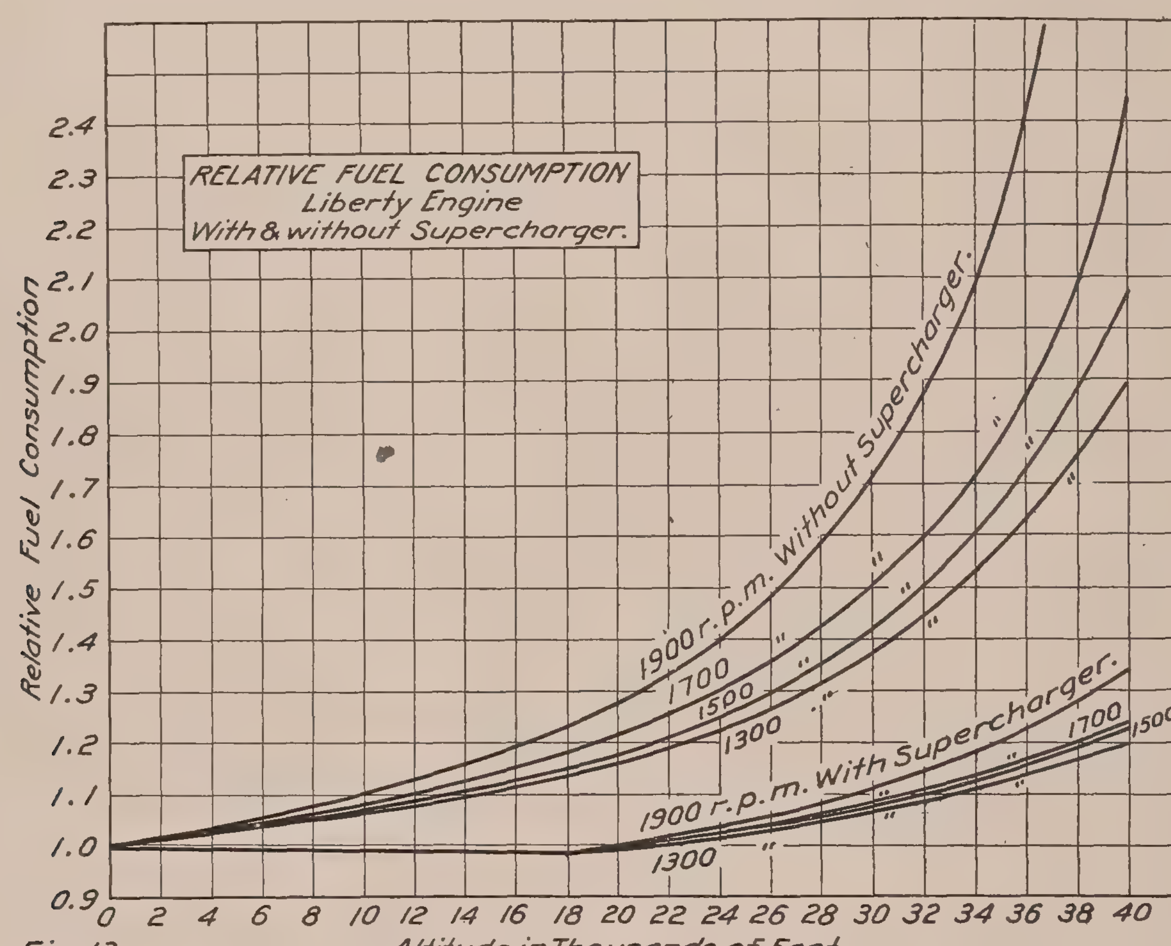

Fig. 13.

(greatest altitudes). This tendency would be greatly augmented under flying conditions at great altitudes as a result of the very low temperatures and consequent poor carburetion and distribution. It is to be expected, therefore, that in using (44) we will underestimate the fuel consumption at great altitudes for the engine without the supercharger. Since the carburetor pressures and temperatures for the supercharging engine are relatively high, however, the above remark does not apply to the estimated fuel consumption for the engine compressor unit.

Figure 13 shows curves giving the relative fuel consumption of the Liberty engine with and without the supercharging compressor at all altitudes, as computed from (44). The great waste of fuel at high levels is very evident. Its physical explanation lies in the fact that as the power drops off, the mechanical losses, which are assumed to be constant for each speed, use up a larger and larger percentage of the energy of the fuel. The supercharging compressor, by maintaining the power, maintains the mechanical efficiency.

The actual value of the supercharging device as a means for saving fuel is best seen, however, when we are in a position to plot curves showing the fuel consumption per mile instead of per brake horsepower hour. This subject will be taken up in the second part of the report.

SUMMARY OF NOTATION FOR PART 1.

The subscripts 1 and 2 refer to the state of the air as it enters the compressor and the carburetor, respectively.

F_t = temperature correction factor for horsepower of engine.
r_p = pressure correction factor for equal intake and exhaust pressures.
R_p = pressure correction factor for unequal intake and exhaust pressures.
p, P = air pressure in cm. of mercury and pounds per square foot, respectively.
P_i, P_e = engine intake and exhaust pressures, respectively.
t, T = Fahrenheit and absolute Fahrenheit temperatures, respectively.
V = volume in cubic feet.
R = gas constant for air in engineer's units (= 53.34).
ρ = density of air (dry) in pounds per cubic foot.
C_p = specific heat of air at constant pressure in B. t. u. per pound (= 0.241).
C_v = specific heat of air at constant volume in B. t. u. per pound (= 0.171).
$\gamma = C_p/C_v = 1.406$.
J = mechanical equivalent of heat = 778 foot pounds per B. t. u.
Q = compressor intake volume in cubic feet per minute.
M = air flow through compressor in pounds per minute.
I, I' = energy input of compressor per pound of air (actual and theoretical).
T'_2 = theoretical carburetor temperature (adiabatic compression).
h = heat radiated per minute.
C_h = coefficient of heat transfer in B. t. u. per square foot per degree Fahrenheit mean temperature difference per hour.
v = air speed in feet per second.
$k = h/(T_2 - T_1)$.
$\mu = MC_p/(MC_p + k)$.
$A\left(\frac{P_2}{P_1}\right) = \left(\frac{P_2}{P_1}\right)^{0.289} - 1$.
E_s = shaft efficiency of compressor.
E_h = hydraulic efficiency of compressor.
N = compressor speed in revolutions per minute.
$q = Q/N$ = quantity coefficient.
u_a = peripheral speed of impeller of compressor in feet per second.
g = acceleration of gravity.
H_c = horsepower absorbed by compressor.
η = mechanical efficiency of engine.
ϵ = volumetric efficiency of engine.
σ = ratio of volumetric efficiency for actual intake and exhaust pressures to volumetric efficiency for equal intake and exhaust pressures.
r = compression ratio of engine.
V_s = displacement volume for one piston in cubic feet.
D = total piston displacement of engine in cubic feet.
V_c = compression volume in cubic feet.
N_e = crankshaft speed in revolutions per minute.
H_G = gross horsepower developed by engine.
H = net horsepower available for driving propeller.
$\alpha = \frac{1}{gR}\left(\frac{\gamma - 1}{\gamma}\right)\left(\frac{u_a}{N}\right)^2$.

For the definitions of the remaining symbols see the equations indicated in the table below:

Symbol	φ	ψ	χ	ζ	ζ_1	χ_1
Equation	(23)	(24)	(30)	(31)	(38)	(42)

REPORT No. 101.

PART II.

THE CALCULATION OF AIRPLANE PERFORMANCE FROM THE ESTIMATED PERFORMANCE CURVES OF ENGINE AND COMPRESSOR.[1]

RÉSUMÉ, PART II.

If the heat leak from the gas turbine and exhaust pipes to the water jackets is prevented, and if the cooling system is kept under a constant pressure independent of that of the atmosphere, no additional radiator equipment should be required when a supercharging compressor is fitted to an airplane engine.

The total additional weight of the propelling plant due to the use of a supercharger is estimated at about 120 pounds.

A method of estimating airplane performance at altitudes with the aid of curves for the "reduced" thrust horsepower available and required, is developed. This method simplifies the graphs of the thrust horsepower required at altitudes, and is particularly useful in comparing the performance of planes of different sizes, wing loadings, and propelling plant characteristics, which have the same lift and drag coefficients.

Two methods for drawing curves of the thrust horsepower available with a variable pitch propeller are indicated.

Horizontal flight speed and maximum climbing speed curves for the LePere two-seater fighter when equipped with supercharging and nonsupercharging engines, and with both fixed blade and variable pitch propellers, are worked out with the aid of the estimated performance curves for the Liberty engine with turbine-driven supercharging compressor shown on figure 10, part 1.

Altitude-time curves at maximum climbing rate are plotted for the LePere when equipped with each of the four types of propelling plant just mentioned.

Curves showing the relative fuel economy (i. e., relative distance traversed per pound of fuel) with the engine wide open at all altitudes, are plotted and discussed.

A supercharging installation suitable for commercial use is described, and it is shown that with the aid of the compressor a great saving in fuel and a considerable increase in carrying capacity can be effected simultaneously.

The outcome of the investigation is distinctly favorable to the use of a supercharging compressor as a means for obtaining better performance for both military and nonmilitary airplanes. The variable pitch propeller would be a valuable adjunct to the supercharger, but is not essential to its utility.

In an appendix the writer derives a theoretical formula for the correction of the thrust coefficient of an airscrew to offset the added resistance of the plane due to the slip stream effect.

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[1] A summary of the notation for Part II will be found at the end of this part.

1. INTRODUCTION.

In this second part of the report, the estimated performance curves for the engine with turbine-driven supercharger are applied to the discussion of airplane performance at great altitudes. The methods of calculating altitude performance curves are of general application, although they are particularly convenient for the treatment of the specific problem here under discussion.

The reader who is not interested in methods of computation can omit articles 4, 5, 6, and the major portion of article 8.

The appendix on "The correction of the propeller thrust coefficient for the slip stream resistance" has only an indirect connection with the supercharging problem.

2. RADIATOR EQUIPMENT FOR AIRPLANES WITH SUPERCHARGERS.

In the recent trials of the General Electric turbine-driven supercharger on the LePere airplane, additional radiator equipment was needed, over and above that required for the engine without the compressor. Calculation shows, however, that in the future we may expect to dispense with this excess radiator equipment. A primary reason for the use of a large radiator with the General Electric supercharger has been that there is a considerable heat leak from the gas turbine and the exhaust manifolds to the water jackets. Suitable heat insulation should reduce this leak to negligible magnitude and so greatly decrease the amount of heat to be disposed of in the radiator.

The heat radiated per square foot of radiator area per unit time is proportional to the mean temperature difference between the water and the air, and to the 0.83 power of the product of the density of the air and the speed of advance.[2] The area required under any given conditions should be roughly proportional to the power of the engine and should vary inversely as the rate of heat transmission per unit area. We have already seen that the power of the engine remains nearly constant from sea level up to the maximum altitude at which sea-level carburetor pressure is obtainable. At this altitude the ratio of the density of the air to the power developed by the engine has its least value. Hence if the radiator is large enough to take care of the heat to be dissipated at sea level and also at this critical altitude, it should be large enough for all altitudes.

Let us therefore compute the ratio of the radiator area S required at the maximum altitude for sea level carburetor pressure, to the area S_o required at the ground. It will readily be seen from the preceding paragraph that this ratio is

$$\frac{S}{S_o}=\frac{H}{H_o}\left(\frac{d_o V_o}{dV}\right)^{0.83}\times\left(\frac{t_o'-t_o}{t'-t}\right) \tag{1}$$

where H = horsepower of engine,

t = temperature of air,

t' = mean temperature of water,

V = speed of advance,

d = relative density of air,

and the subscript $_o$ indicates sea level values.

In radiator calculations for altitude work it is customary to assume that the difference between the mean temperature of the water and the boiling point of water is to be kept constant as the atmospheric pressure changes. The lowering of the boiling point as the pressure drops off then largely compensates for the decrease in the air temperature, and greatly reduces the available temperature difference at great altitudes. It is possible, however, to put the cooling system under a constant pressure, so that the boiling point of the water will not vary with the pressure of the external atmosphere, and our calculation will be based on the assumption of the existence of such a fixed radiator pressure.

[2] Cf. article 5, Part I.

Consider the special case of the Liberty engine and turbine-driven compressor discussed in Part 1. The critical altitude is 18,000 feet. According to figure 4 the mean free-air temperatures at sea level and 18,000 feet are 59° F. and −5° F., respectively. We assume that the mean water temperature is 182° F., or 30° less than the sea level boiling point. Then

$$\frac{t_0'-t_0}{t'-t}=\frac{182-59}{182+5}=0.658.$$

The relative density of the air at 18,000 feet is 0.57, while that at sea level is by definition unity. We assume the speeds of advance at sea level without the compressor and at 18,000 feet with the compressor to be 138 miles an hour and 160 miles an hour, respectively. (This assumption will be justified later on.) The horsepowers at sea level and 18,000 feet for normal speed (fig. 11, 1700 revolutions per minute) are 425 and 395 respectively. Then

$$\frac{S}{S_0}=\frac{395}{425}\times 0.658\left(\frac{138}{0.57\times 160}\right)^{0.83}=0.862.$$

This shows that under the conditions of the above calculation the required radiator area should not be increased by the use of a supercharging compressor unless there is a heat leak from the turbine to the water jackets.

If the radiator is kept at the pressure of the external atmosphere, on the other hand, a small increase in radiator area is needed. The temperature factor becomes

$$\frac{t_0'-t_0}{t'-t}=0.796,$$

and the relative radiator area required at 18,000 feet is

$$\frac{S}{S_0}=1.043.$$

Such an increase of a little more than 4 per cent in the radiator area would not be a serious handicap to the performance of an airplane. In the calculations which follow we will assume, however, that the radiator area is *not* increased by the use of the supercharging compressor.

3. ADDITIONAL WEIGHT DUE TO SUPERCHARGER.

The one handicap to airplane performance involved in the use of a supercharger is additional weight. We estimate the weight of the compressor, turbine, and mountings at 100 pounds for the special case considered in part 1. The increase in the weight of the propeller would be about 20 pounds, making a total additional weight due to the supercharger of about 120 pounds.

The weight of the compressor would not vary greatly with the size of the engine to which it is fitted. Consequently the greatest increase in the ratio of the horsepower to the weight is to be expected in supercharging with *large* engines.

4. CALCULATION OF AIRPLANE PERFORMANCE CURVES FOR GREAT ALTITUDES: REDUCED THRUST HORSEPOWER AND REDUCED SPEED OF ADVANCE.

The method which we shall employ for computing altitude performance curves is new in part.

The simplest means for finding the maximum horizontal flight speed and maximum climbing speed at sea level is to draw up curves showing the thrust horsepower available and required for propulsion as functions of the speed of advance. The high speed intersection of these two curves gives the maximum horizontal flight speed of the airplane, and the maximum difference of the ordinates of the two curves is usually taken to be the maximum power available for climbing. In order to use this method for altitude performance calculations it is necessary to draw up a pair of curves for the horsepower required and available at each altitude

considered. If a comparison of the performances of the same airplane when equipped with two, or more, different propelling plants is desired, a further complication arises from the fact that any alteration in the weight of the machine modifies the set of curves for the required horsepower. In order to avoid this undue multiplication of graphs, we will substitute for the actual thrust horsepower and actual speed of advance two quantities which we shall call the "reduced thrust horsepower" and the "reduced speed of advance." When this is done the whole set of curves for the thrust horsepower required at different altitudes by similar machines of different weights collapses into one. A simple slide rule calculation suffices for the determination of the actual maximum horizontal flight speed under any given conditions when the reduced maximum horizontal flight speed is known.

Let U, V = speed of advance (relative to air) in miles per hour and feet per second, respectively;
A = wing area in square feet;
α = angle of attack;
θ = angle of climb;
ρ = density of air in pounds per cubic foot;
g = acceleration of gravity = 32.16 feet per second per second;
Y = lift of machine in pounds;
X = drag of machine in pounds;
$K_y(\alpha)$ = lift coefficient for entire airplane;
$K_x(\alpha)$ = drag coefficient for entire airplane.

Then from the definition of the lift and drag coefficients we have

$$Y=\frac{\rho V^2 A}{g} K_y(\alpha) \tag{2}$$

$$X=\frac{\rho V^2 A}{g} K_x(\alpha). \tag{3}$$

For our present purpose it will be sufficiently accurate to assume that the propeller thrust is opposite in direction to the speed of advance.

Let W = total weight of airplane in pounds;
T = propeller thrust in pounds.

Then

$$Y= W \cos \theta. \tag{4}$$

$$T=X+Y \tan \theta. \tag{5}$$

Substituting $Y\ K_x(\alpha)/K_y(\alpha)$ for X in (5), we reduce it to the form:

$$T= Y[K_x(\alpha)/K_y(\alpha)+\tan \theta]. \tag{6}$$

The elimination of Y between (4) and (6) then gives

$$T= W\left[\frac{K_x(\alpha)}{K_y(\alpha)} \cos \theta+\sin \theta\right]. \tag{7}$$

Let H_t = thrust horsepower *required* for steady flight with the angle of attack α and the angle of climb θ.

Then

$$H_t= TV/550=\frac{WV}{550}\left[\frac{K_x(\alpha)}{K_y(\alpha)} \cos \theta+\sin \theta\right]. \tag{8}$$

This is the first of the two fundamental equations of our theory. The second is obtained by combining (2) amd (4). It is

$$V=\sqrt{\frac{Wg \cos \theta}{\rho A K_y(\alpha)}}. \tag{9}$$

The above equations in the parameter α can be used to plot the curve showing the relationship between the thrust horsepower required and the speed of advance for any specified angle of climb.

Substituting from (9) into (8) and rearranging, we obtain

$$\frac{H_t}{W}\sqrt{\frac{\rho A}{W}}=\frac{1}{550}\sqrt{\frac{g\cos\theta}{K_y(\alpha)}}\left[\frac{K_x(\alpha)}{K_y(\alpha)}\cos\theta+\sin\theta\right]. \tag{10}$$

Rearrangement of (9) itself yields

$$V\sqrt{\frac{\rho A}{W}}=\sqrt{\frac{g\cos\theta}{K_y(\alpha)}}. \tag{11}$$

The right hand side of each of the equations (10) and (11) is a function of α and θ independent of the density of the air and the weight of the machine. This fact suggests the definitions of the "reduced thrust horsepower" and the "reduced speed of advance" which we shall adopt.

Let w denote the wing loading (W/A) in pounds per square foot. We define the "reduced thrust horsepower required" h_t and the "reduced speed of advance" u by means of the equations

$$h_t=\frac{H_t}{W}\sqrt{\frac{\rho}{w}}, \tag{12}$$

$$u=U\sqrt{\frac{\rho}{w}}=0.6818\,V\sqrt{\frac{\rho}{w}}. \tag{13}$$

Here H_t=horsepower required for propulsion at the altitude and speed of advance under consideration

Then equations (10) and (11) yield

$$h_t=\frac{1}{550}\sqrt{\frac{g\cos\theta}{K_y(\alpha)}}\left[\frac{K_x(\alpha)}{K(\alpha)_y}\cos\theta+\sin\theta\right]. \tag{14}$$

$$u=0.6818\sqrt{\frac{g\cos\theta}{K_y(\alpha)}}. \tag{15}$$

(14) and (15) might be used as they stand for the determination of the maximum speed of advance at different climbing angles. We shall be interested, however, only in the case of horizontal flight for which they reduce to

$$h_t=\frac{0.0103\,K_x(\alpha)}{[K_y(\alpha)]^{3/2}} \tag{16}$$

$$u=\frac{3.867}{\sqrt{K_y(\alpha)}}. \tag{17}$$

It will readily be seen from (16) and (17) that the relationship between h and u for horizontal flight depends only on the lift and drag coefficients of the airplane, and is independent of its size and weight, and also of the density of the air in which it flies.

The h_t, u curve for any given machine or family of machines may be plotted from the parametric equations (16) and (17), or it may be derived from the H_t, U curve for any given set of conditions if one is obtainable. For example, figure 14 shows the thrust horsepower required for the propulsion of the LePere two-seater fighter in horizontal flight at sea level as a function of the speed of advance U. Inserting in (12) and (13) the numerical values of ρ, W, and w applicable to this particular case, we obtain

$$h_t=2.48\times10^{-5}H_t,$$

and

$$u=0.0905\,U.$$

Figure 15 shows the curve for the reduced thrust horsepower required derived from figure 14 by means of the foregoing equations.

Let H_t' denote the maximum thrust horsepower *available* at the speed of advance U when the density of the air is ρ, and let h_t' denote its reduced value, i. e., let

$$h_t' = \frac{H_t'}{W}\sqrt{\frac{\rho}{w}}. \tag{18}$$

h_t' differs from h_t in that it depends on the density of the air, the weight of the machine, etc. Its value for any given airplane, propelling plant, and air density can be calculated, however, and a series of curves can be laid out showing h_t' as a function of u for various altitudes. Such a set of curves for the Liberty engine with turbine-driven supercharging compressor on the Le Pere plane is shown on figure 15. The method of plotting these curves will be discussed in the next article. For the present we will concern ourselves only with their use.

The maximum speed of horizontal flight is that speed for which the maximum thrust horsepower available is equal to the thrust horsepower required. But when H_t equals H_t' it is

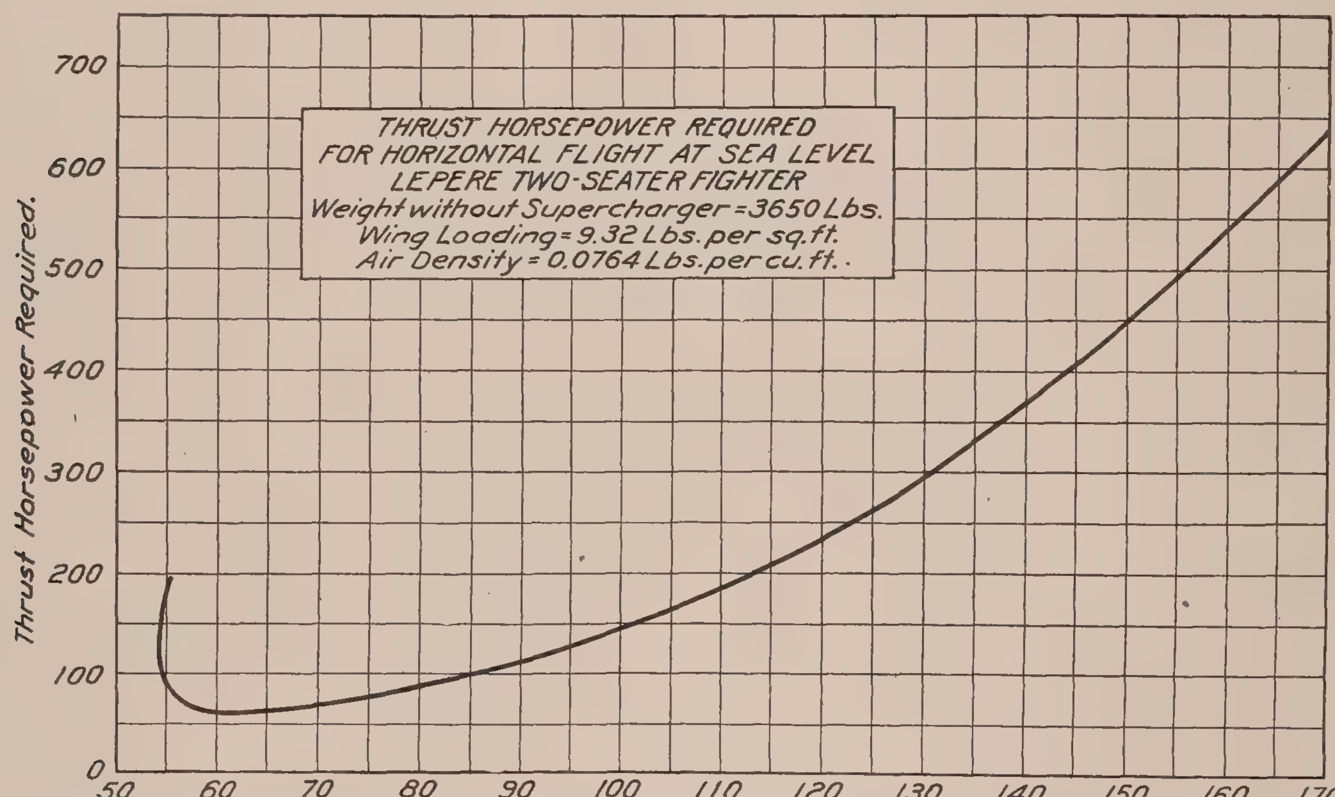


Fig. 14.

obvious that h_t must equal h_t'. Hence the reduced value of the maximum horizontal flight speed at any altitude is the value of u for the point where the curve, $y = h_t'(u)$, for the given altitude crosses the curve, $y = h_t(u)$. To get the actual horizontal flight speed from its reduced value, we make use of (13).

For example, the reduced value of the maximum horizontal flight speed at 30,000 feet given by figure 15 is 8.5. The weight of the machine as modified by the addition of the compressor is 3,770 pounds. The wing loading is 9.625 pounds per square foot. The density of the air at 30,000 feet is 0.02866 pounds per cubic foot. Hence the actual maximum horizontal flight speed at 30,000 feet is

$$U = 8.5\sqrt{\frac{9.625}{0.02866}} = 155.5 \text{ mi./hr.}$$

The maximum climbing speed is also easily calculated with the aid of the curves for the reduced thrust horsepower available and required for horizontal flight. In accordance with common practice we make the approximate assumption that the maximum horsepower available for climbing is equal to the maximum difference between the ordinates of the curves for

the thrust horsepower available and required for horizontal flight. Let H_c denote the horsepower available for climbing, and let V_c denote the maximum climbing speed in feet per minute. Then

$$H_c = W\sqrt{\frac{w}{\rho}}(h_t' - h_t)_{\max} = WV_c/33{,}000.$$

Hence

$$V_c = 33{,}000\sqrt{\frac{w}{\rho}}(h_t' - h_t)_{\max}. \qquad (19)$$

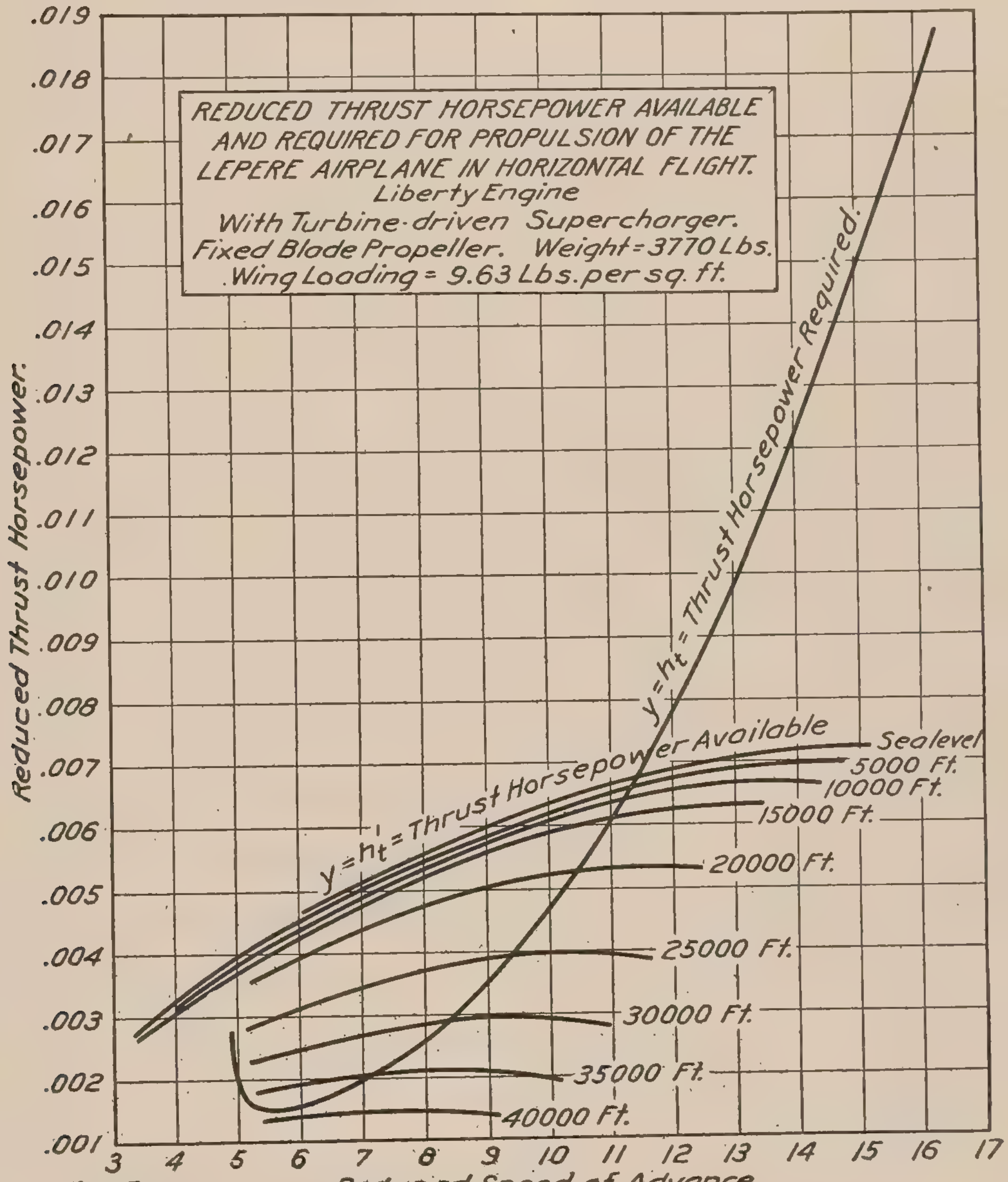


Fig. 15.

As an example of the application of (19) let us again consider the airplane and power plant of figure 15, at 30,000 feet. The maximum value of $(h_t' - h_t)$ is 0.00099. Hence

$$V_c = 33{,}000 \times 0.00099\sqrt{\frac{9.625}{0.02866}} = 599 \text{ feet/minute.}$$

5. METHOD OF PLOTTING CURVES FOR THE THRUST HORSEPOWER AVAILABLE: 5. FIXED PROPELLER BLADES.

The form of the curves showing the reduced thrust horsepower available as a function of the reduced speed of advance is conveniently calculated by a method described by Bairstow and Coales.[3]

[3] "Notes on the Prediction and Analysis of Aeroplane Performance," by L. Bairstow and Lieut J. D. Coales, British Advisory Committee or Aeronautics, Reports and Memoranda No. 474, May, 1918.

Let Q = propeller torque in pounds-feet;
n = propeller speed in revolutions per minute;
D = propeller diameter in feet;
P = experimental mean pitch in feet.

The torque coefficient q_c is defined by the following equation:[4]

$$q_c = -\frac{gQ}{\rho n^2 D^5}. \tag{20}$$

The dimensionless quantity q_c is a function of V/nD, or of V/nP, which is to a first approximation independent of the propeller size for a family of similar propellers. The propeller efficiency, which we denote by η, is also a function of V/nP. In order to make the computation by the method here described, it is necessary to have curves showing q_c and η as functions of V/nP for the propeller employed. (In the future the quantity V/nP will be denoted by the single symbol σ).

The power absorbed by the propeller, i. e., the brake horsepower of the engine H, is related to the torque coefficient by the following simple equation:[5]

$$q_c = \frac{550gH}{2\pi \rho n^3 D^5}. \tag{21}$$

It is easy to compute from this equation, and the curves for q_c and η as functions of σ, the propeller efficiency and the speed of advance corresponding to any given set of values for n, H, ρ, and D. To do this we first calculate the value of q_c from (21). The q_c, σ curve gives value of σ, and the speed of advance is then computed from

$$V = nP\sigma. \tag{22}$$

The propeller efficiency for the same value of σ is taken from the η, σ curve. The thrust horsepower available at the speed V is then

$$H_t' = \eta H. \tag{23}$$

We are interested in the computation of the reduced speed of advance and the reduced thrust horsepower available. In order to get these reduced values directly, we substitute from (22) and (23) into (12) and (13). The resulting expressions are

$$u = 0.682 nP\sigma \sqrt{\frac{\rho}{w}}, \tag{24}$$

$$h_t' = \frac{\eta H}{W} \sqrt{\frac{\rho}{w}}. \tag{25}$$

Assuming a series of different values for n, it is easy to determine the form of the u, h_t' curve from equations (21), (24), and (25).

Example.—Let us compute the h_t', u curve for the Liberty engine with turbine-driven supercharger, as installed on the Le Pere two-seater fighter, for a propeller with fixed blades at an altitude of 30,000 feet. We assume the propeller efficiency and torque coefficient curves shown on figure 16.[6] The ratio of the experimental mean pitch (in flight) to the diameter

[4] Note that this definition differs from that adopted by Durand in Reports 14 and 30 (National Advisory Committee for Aeronautics, 1917 and 1919). The relation between q_c and the Durand coefficient is shown by equation (26), q. v.

[5] Cf. Bairstow and Coales, loc. cit., equation 14.

[6] These curves were not derived from experimental tests on any definite propeller. They were drawn up from the Bairstow-Coales-Betts empirical formulas for the torque and thrust coefficients (British Advisory Committee for Aeronautics, Reports and Memoranda No. 474, Appendix) with the aid of the meager data available on the propeller used in the test of the Le Pere two-seater fighter on Aug. 15, 1918. The assumption of a maximum efficiency of a little over 80 per cent is pure guesswork.

The curves are intended to show the *effective* values of the torque coefficient and of the propeller efficiency in flight. The determination of these effective values involves corrections for body interference and for the additional resistance of the body due to the slip stream. The first of these corrections consists in a simultaneous increase in effective pitch and in the efficiency of the propeller. The correction for the slip stream consists in a scaling down of the thrust coefficient curve in a manner described in the appendix to this report.

for the family of propellers specified by these curves is assumed to be 1.06. In order that a propeller of the family specified by these curves shall hold the speed of the supercharging engine down to 1,800 revolutions per minute at 18,000 feet altitude with a speed of advance of 162 miles per hour (cf. fig. 22) it is necessary that its diameter shall be 10.54. This is large for practical purposes, but that fact need not concern us in the present theoretical discussion.

The density of the air at 30,000 feet being 0.02866 pounds per cubic foot, equation (21) reduces to

$$q_c(\sigma) = 0.756 H/n^3. \tag{21'}$$

The pitch of the propeller is 11.17 feet, and equations (24) and (25) become

$$u = 0.4165 n\sigma, \tag{24'}$$

$$h_t' = 1.446 \times 10^{-5} \eta H. \tag{25'}$$

The computation of the u, h'_t curve for this altitude is summarized in the accompanying table.

TABLE 3.

N_e. R. P. M.	n. R. P. S.	H. (Fig. 10.)	q_c. (Equation 21'.)	σ (Fig. 16.)	u. (Equation 24'.)	η (Fig. 16.)	h'_t (Equation 25'.)
1,600	26.67	234	.00933	0.519	5.76	0.712	24.1
1,700	28.33	246	.00818	.66	7.78	.795	28.3
1,800	30.00	253	.00708	.752	9.4	.805	29.4
1,900	31.67	252	.0060	.825	10.9	.761	27.7

6. THRUST HORSEPOWER AVAILABLE WITH VARIABLE PITCH PROPELLER.

There are two methods of drawing up curves for the reduced horsepower available with a variable pitch propeller. The first and more accurate method requires experimental curves for the torque coefficient and efficiency at various blade settings, such as those given by Durand for propeller 96 in Report No. 30 of the Fourth Annual Report of the National Advisory Committee for Aeronautics. The second method makes use of approximate empirical formulas due to (Miss) Betts, Mettam, Bairstow, and Coales, and requires only a minimum of data regarding the characteristics of the propeller at the particular blade setting which gives maximum efficiency.

Method I. A complete set of curves for the efficiency and torque coefficients at various blade settings available.—We assume that the propeller is to be adjusted so that under all conditions the engine is permitted to revolve at the speed of maximum power. The diameter being chosen, let it be required to draw up a graph of the reduced horsepower available as a function of the reduced speed of advance for some definite altitude. Equations (21), (24), and (25) still hold, but the propeller setting and pitch for any given speed of advance is unknown.

The method is as follows: The torque coefficient q_c is computed from (21). The torque coefficient Q_c, as defined by Durand, is related to q_c by the equation

$$Q_c = \frac{1000 q_c}{g(V/nD)^2}. \tag{26}$$

(We assume that the available data is in the form of curves for Q_c and η as functions of $V/(nD)$). Q_c is calculated from (26). Since V/nD is a known quantity, the Q_c curves (see Plate XIX, Report No. 30, Fourth Annual Report of the National Advisory Committee for Aeronautics) together with the value of Q_c for the given conditions fix the blade setting. The propeller efficiency can then be determined by interpolation from the η, $\frac{V}{nD}$ curves. (See Plate XIII, Report No. 30.) Finally the reduced thrust horsepower available and the reduced speed of advance can be computed from (25) and (13), respectively.

Method II. Use of empirical formulas for thrust and torque coefficients.—The method of computation here briefly described is due to Miss A. D. Betts and H. A. Mettam.[7] It has the advantage over the method described above that it requires a minimum of information regarding the propeller. The only data needed are the absolute maximum of efficiency for the propeller, the experimental mean pitch corresponding to the maximum efficiency and the torque coefficient corresponding to the maximum efficiency.

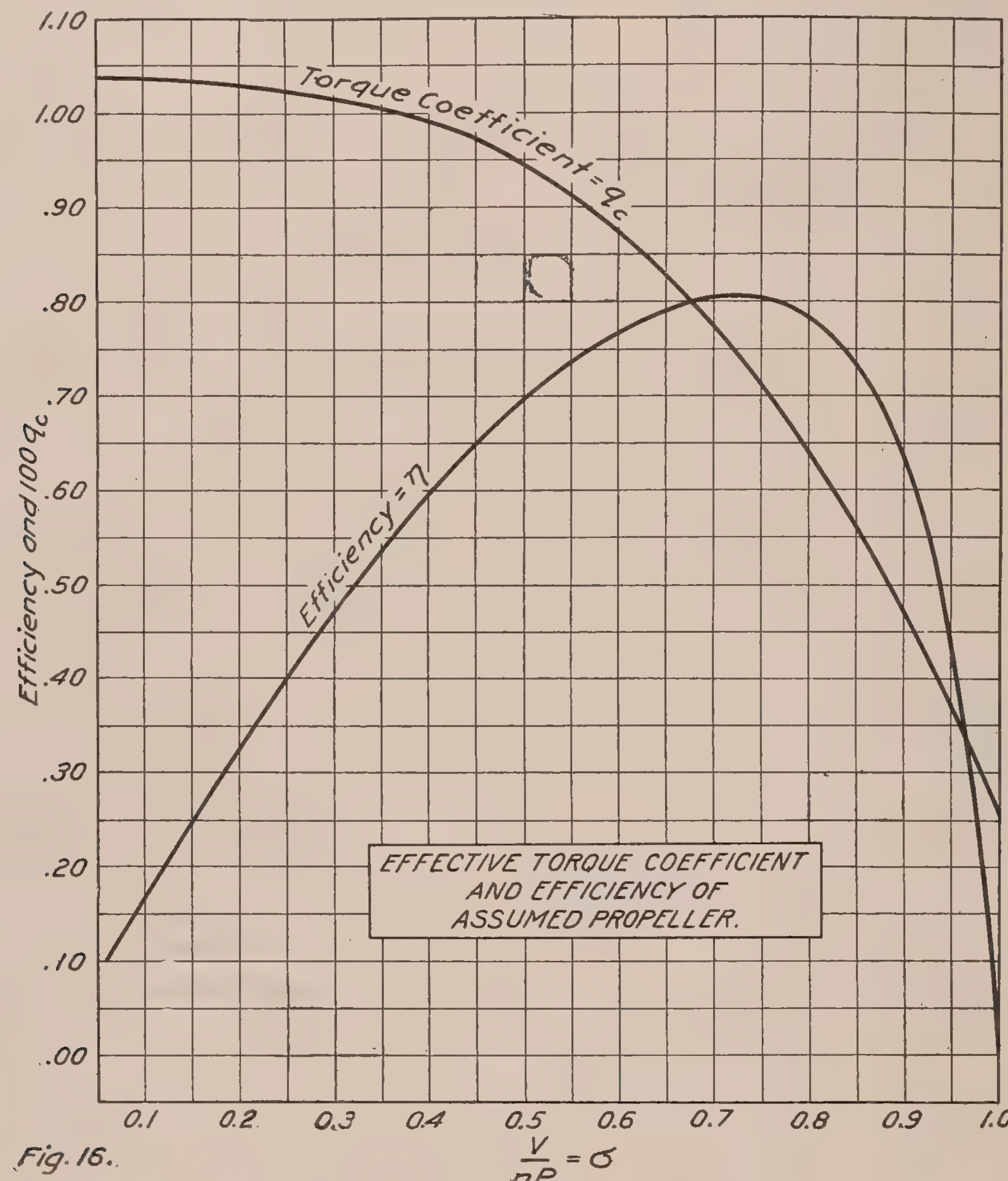


Fig. 16.

The method is based on the following empirical formulas for the thrust coefficient, the torque coefficient, and the efficiency of any fixed blade propeller:

$$K_t t_c = \frac{4}{3}(1-\sigma^2), \tag{27}$$

$$K_q q_c = 1.1042 - 0.833\sigma^3, \tag{28}$$

$$\eta = \frac{1}{2\pi}\frac{P}{D}\frac{K_q}{K_t}F(\sigma). \tag{29}$$

Here t_c is the thrust coefficient as defined by

$$t_c = \frac{gT}{n^2 D^4}, \tag{30}$$

[7] Miss A. D. Betts and H. A. Mettam, "Empirical Formulæ for a Variable Pitch Airscrew, with Applications to the Prediction of Aeroplane performance," British Advisory Committee for Aeronautics, Reports and Memoranda, No. 474, February, 1919.

K_q and K_t are constants, while $F\ (\sigma)$ is defined by the equation

$$F(\sigma)=\frac{\sigma-\sigma^3}{0.828-0.625\sigma^3}. \tag{31}$$

Figure 17 shows curves of $F\ (\sigma)$ and $K_q q_c$.

The above formulas hold for variable pitch propellers, if it be understood that K_t and K_q depend on the pitch. Let η_m, P_m, and K_{q_m} denote, respectively, the absolute maximum of efficiency, the corresponding pitch, and the corresponding value of K_q. Betts and Mettam have established approximate empirical relationships between P/P_m and the ratios K_q/K_{q_m} and η/η_m, which we indicate by the following equations:

$$K_q=K_{q_m}\phi(P/P_m) \tag{32}$$

$$\eta=\eta_m\zeta(P/P_m)\,F(\sigma). \qquad (33$$

Graphs of the empiracally determined functions ϕ and ζ are shown on figure (18).

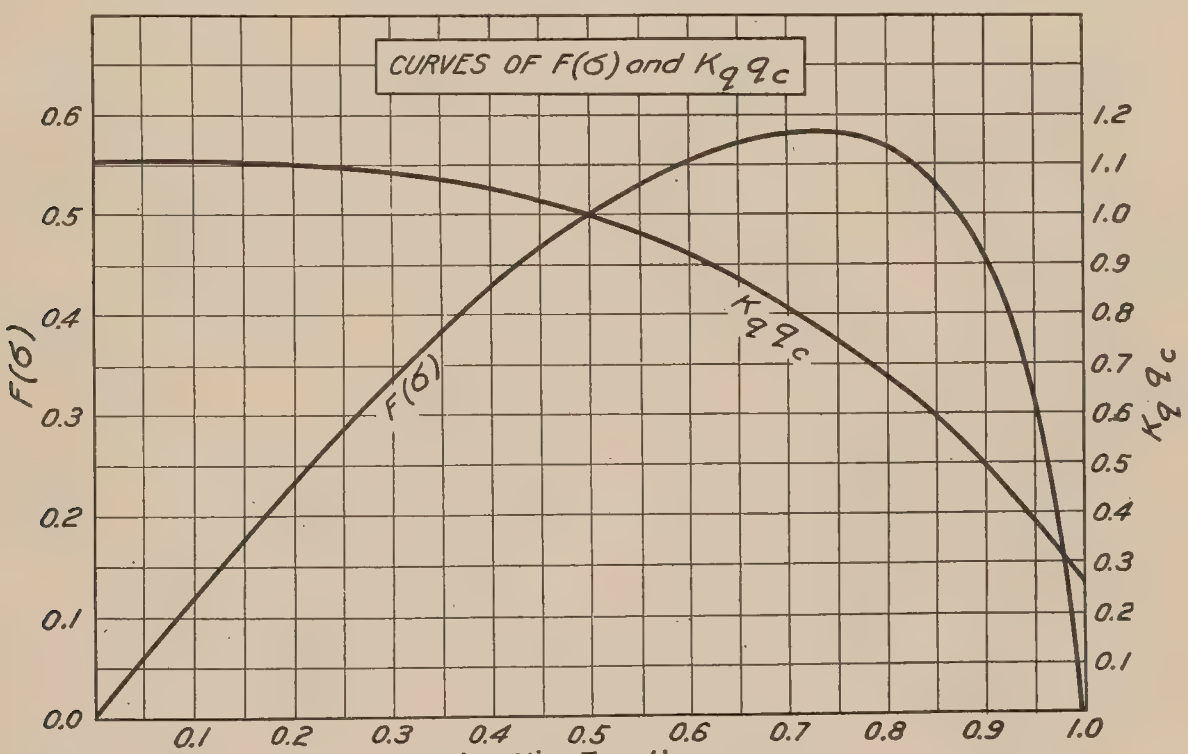


Fig. 17.

The function ζ may be expressed by the formula

$$\zeta=\frac{1}{0.583}\frac{P}{P_m}^{1-P/P_m}. \tag{34}$$

Let q_{c_m} denote the torque coefficient corresponding to the maximum efficiency. Since the maximum efficiency occurs when $F(\sigma)$ is a maximum, or when σ is 0.725, it is easy to calculate K_{q_m} from q_{c_m}. The substitution of 0.725 for σ in (28) yields

$$K_{q_m}=0.787/q_{c_m}.^{8} \tag{35}$$

When η_m, P_m, and K_{q_m} are known, the determination of the curve for the reduced horsepower available is comparatively straightforward. Assume a number of values of P. For each, calculate the value of P/P_m and determine K_q with the aid of (32) and figure 18. The torque coefficients q_c are calculated from equation (21), and the values of $F(\sigma)$ are taken from the $K_q q_c$ curve of figure 17. Equation (33) in conjunction with the graphs of ζ and $F(\sigma)$

[8] An error in the paper of Betts and Mettam, loc. cit., p. 11, states that $K_{q_m}q_{c_m}=1$. As a matter of fact, the product $K_q q_c$ is equal to unity when σ is 0.5, which is not the point of maximum efficiency. See figure 1 of the Betts and Mettam report.

yields the propeller efficiencies. The computation is completed with the application of (24) and (25) to the determination of the values of u and h'_t corresponding to the several assumed values of P.

Example.—Let us compute a set of u, h'_t curves for the Liberty engine with supercharger as installed on the Le Pere, the propeller being of variable blade angle, but otherwise similar to the propeller of the problem of article 5.

Our first task is to fix the values of P_m, η_m, and K_{q_m}. This can be done with the aid of the information we already possess regarding the torque coefficient and efficiency of the propeller of article 5, together with one additional assumption. Referring to the efficiency curves for

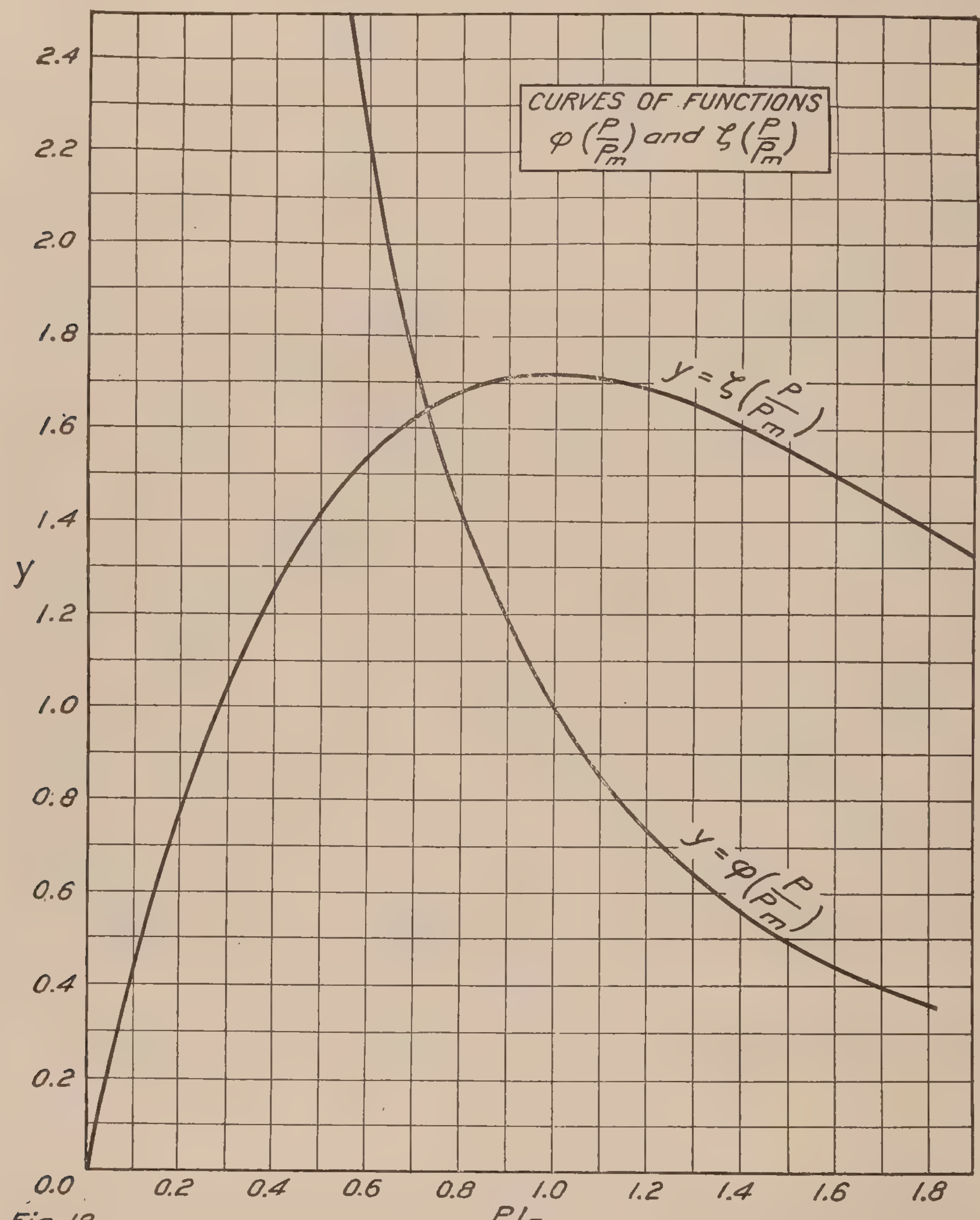


Fig. 18.

propeller No. 96 (Plate XIII, Report No. 30), it will be observed that the maximum efficiency of the propeller increases as the blade setting is advanced from its normal position until the pitch/diameter ratio (given by the intersection of the efficiency curve with the $V/(nD)$ axis) reaches the value 1.3. This is in accord with the general observation that the efficiency of fixed blade propellers tends to increase with the pitch/diameter ratio at least up to values as great as 1.2. We therefore assume that the efficiency of the variable pitch propeller now under consideration reaches its absolute maximum when the pitch/diameter ratio is 1.3.

Since the propeller diameter is 10.54 feet, the above assumption fixes the value of P_m as 13.7 feet. The maximum efficiency for the normal pitch (11.17 feet) is 80.7 per cent. The corresponding values of P/P_m and $\zeta\,(P/P_m)$ are 0.815 and 1.683, respectively. The value of

$F(\sigma)$ for maximum efficiency is 0.583. Hence equation (33) can be solved for the absolute maximum efficiency η_m. It yields

$$\eta_m = \frac{0.807}{0.583 \times 1.683} = 0.823.$$

The constant K_{q_m} can be evaluated in similar fashion. The torque coefficient q_c for the maximum efficiency consistent with the normal pitch of 11.17 feet is 0.00743. (Cf. fig. 16.) The corresponding value of $K_q q_c$ (fig. 17) is 0.785. Hence K_q is 105.6 when the pitch is 11.17 feet. Equation (32) and figure 18 now yield the desired value of K_{q_m}.

$$K_{q_m} = \frac{105.6}{1.385} = 76.3.$$

The remainder of the computation will be summarized for a single altitude only, viz, 30,000 feet. Let it be assumed that the propeller is adjusted at all speeds of advance to allow the engine to turn up to 1,800 revolutions per minute. The horsepower absorbed at 18,000 feet will then be 253. (Fig. 10.) The density of the air is 0.02866 pounds per cubic foot. Hence the torque coefficient q_c for this altitude is 0.00708 (equation (21)). (24) reduces to the form

$$u = 1.116\ P\sigma. \tag{24''}$$

while (25) unites with (33) to give

$$h'_t = 0.00301\ \zeta\ F(\sigma). \tag{25''}$$

The rest of the calculation is condensed into Table 4.

TABLE 4.

P.	P/P_m.	φ. (Fig. 18.)	ζ. (Fig. 18.)	K_q. Equation 32.	$K_q q_c$.	σ. (Fig. 17.)	$F(\sigma)$. (Fig. 17.)	h'_t. Equation 25''.	u. Equation 24''.
9	0.657	1.933	1.588	147.5	1.044	0.413	0.438	0.00209	4.15
10	.73	1.649	1.649	125.8	.891	.633	.566	.00281	7.06
11	.803	1.415	1.677	108.0	.765	.740	.583	.00294	9.1
12	.876	1.239	1.702	94.6	.670	.804	.566	.00290	10.78
13	.949	1.089	1.713	83.1	.589	.851	.528	.00272	12.36

7. THEORETICAL PERFORMANCE CURVES FOR AIRPLANE WITH SUPERCHARGING COMPRESSOR.

Figure 15 shows a complete set of curves for the reduced thrust horsepower available and required for the propulsion of the Le Pere two-seater fighter in horizontal flight when equipped with a turbine-driven supercharger and a fixed blade propeller. Figure 19 shows a similar set of curves for the case where the propeller is of variable pitch. Figures 20 and 21 show curves for the reduced thrust horsepower available and required for the same plane with fixed and variable propeller blades, but without the supercharger. The blade form of the airscrew is the same for all four sets of curves.

The resulting horizontal flight speed and maximum climbing speed curves, calculated in the manner described at the end of article 4, are shown on figures 22 and 23. The very great gain in ceiling, and in the horizontal flight speed and rate of climb at considerable altitudes, due to supercharging even without the variable pitch propeller, is the outstanding feature of these charts. At sea level, to be sure, the horizontal flight speed and the maximum climbing speed are reduced considerably by the use of the supercharger with a fixed blade propeller, but at an altitude of a little over 4,000 feet the supercharging plane is on even terms with the nonsupercharging plane, and at altitudes above 15,000 feet the gain due to supercharging is enormous.

The increase in the height of the ceiling due to the use of the variable pitch propeller is small, particularly in the case of the supercharging job. It should be remembered in this connection that the gain in ceiling due to the use of a variable pitch propeller depends essentially on the particular fixed pitch propeller with which the comparison is made. A suitably chosen fixed pitch propeller (i. e., an altitude propeller) would give as high a ceiling as one of variable pitch, but at the cost of sea level performance.

The chief advantage of the variable pitch airscrew for the supercharging military airplane is in the great increase in climbing speed which it gives, especially for the first 18,000 feet.

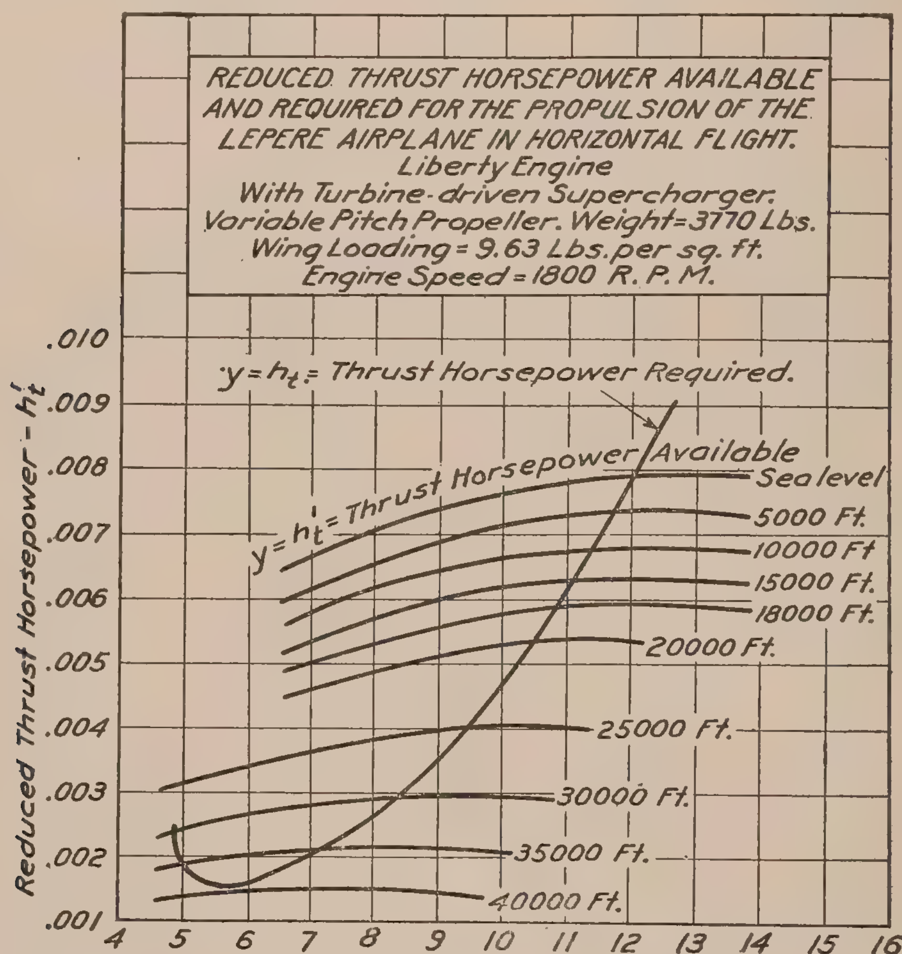


Fig. 19.

This gain in ability to climb is best studied, however, with the aid of altitude-time curves, which will be discussed in the next article.

8. ALTITUDE-TIME CURVES FOR MAXIMUM RATE OF CLIMB.

Let z denote the altitude in feet, and let τ denote the time in minutes.

The theoretical prediction of an altitude-time curve for maximum climb requires the integration of the right-hand member of the equation

$$\tau - \tau_0 = \int_{z_0}^{z} \frac{dz}{V_c(z)}. \tag{36}$$

In the absence of a simple mathematical formula for V_c as a function of z, some method of approximation must be resorted to in order to evaluate the above expression. It so happens that the maximum climbing speed curves are usually approximately rectilinear. Consequently a close approximation to the true curve for climbing speed can usually be obtained by means of a broken line of two or three segments, drawn in by eye with a straight edge. In replacing the computed curve for maximum climbing speed by such a broken line it should be observed that the smaller the value of V_c, the more important a small discrepancy in its value becomes.

The problem of determining the time-altitude curve is thus reduced to the rather simple one of evaluating the right-hand member of (36) when the relation between V_c and z is given

by a broken line. Let the segments be numbered (fig. 23*a*) 0, 1, 2, etc., and let the coordinates of the end points of the r^{th} segment be z_r, $(V_c)_r$ and z_{r+1}, $(V_c)_{r+1}$. The equation of this segment will then be

$$V_c(z) = (V_c)_r + m_r(z - z_r), \tag{37}$$

where

$$m = \frac{(V_c)_{r+1} - (V_c)_r}{z_{r+1} - z_r}. \tag{38}$$

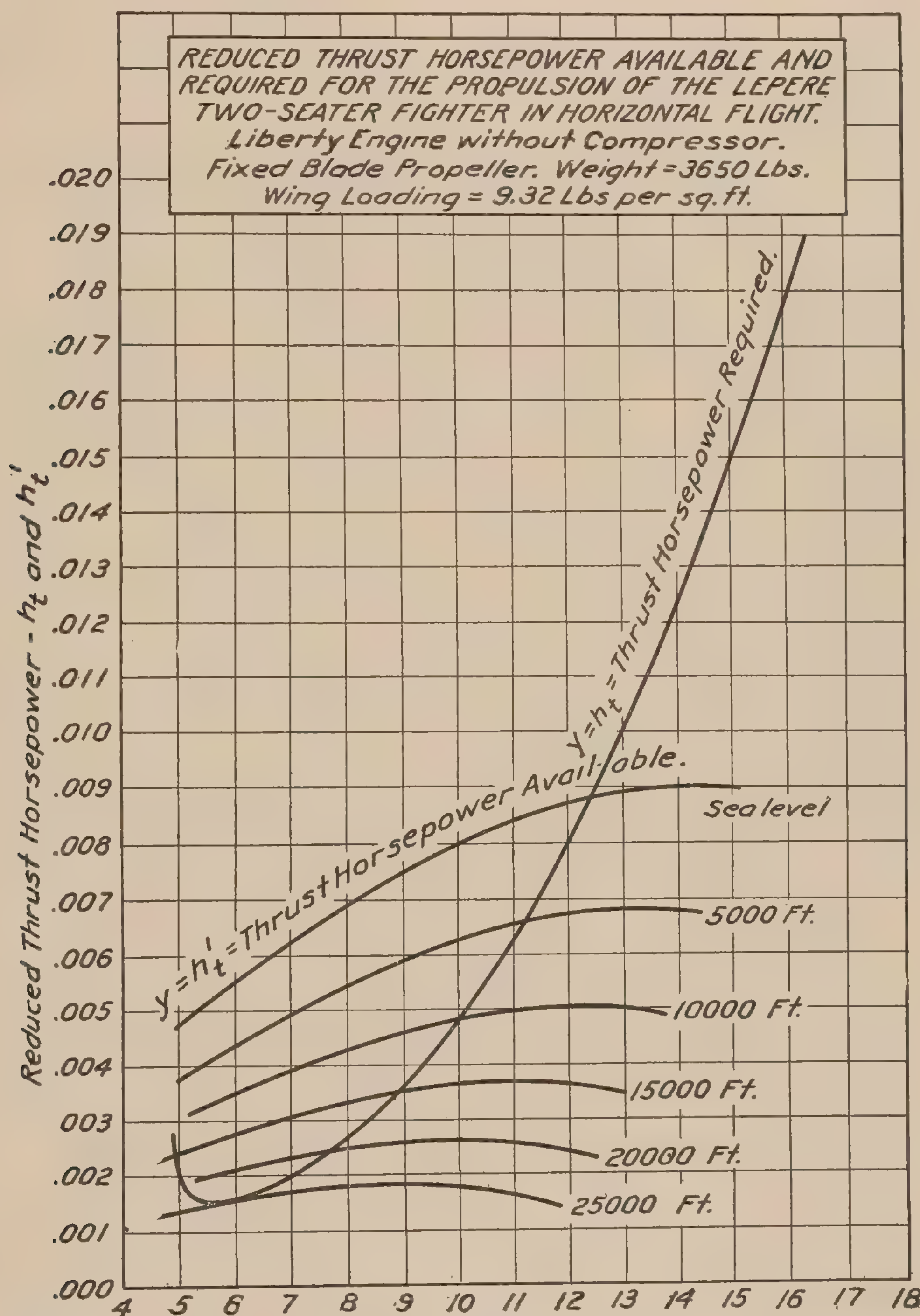


Fig. 20

Let τ_r denote the time corresponding to the altitude z_r. Then (36) yields

$$\tau - \tau_r = \frac{2.3026}{m_r} \log_{10} \left[\frac{V_c(z)}{(V_c)_r} \right]. \tag{39}$$

But evidently,

$$\tau_r - \tau_0 = 2.3026 \sum_{p=0}^{p=r-1} \frac{1}{m_p} \log_{10} \left[\frac{(V_c)_{p+1}}{(V_c)_p} \right]. \tag{40}$$

The addition of equations (39) and (40) gives the following simple expression for the time required to reach an altitude z in the r^{th} segment.

$$\tau - \tau_0 = 2.3026 \left\{ \frac{1}{m_r} \log_{10} \left[\frac{V_c(z)}{(V_c)_r} \right] + \sum_{p=0}^{p=r-1} \frac{1}{m_p} \log_{10} \left[\frac{(V_c)_{p+1}}{(V_c)_p} \right] \right\}. \tag{41}$$

When the values of V_c are taken from the graph, it is easy to evaluate the right-hand member of (41) if the number of segments is small.

Figure 24 shows approximate altitude-time curves obtained in the manner described above from the four maximum climbing speed curves of figure 23. In the evaluation of the curves for the supercharging airplane, broken lines of two segments only were used, while single straight lines were employed for the other two cases.

Comparison of the two altitude-time curves for the supercharging airplane shows that the plane with the variable pitch airscrew climbs 24,000 feet while that with the fixed blade propeller climbs 20,000. Since the former plane would also be able to outmaneuver the latter

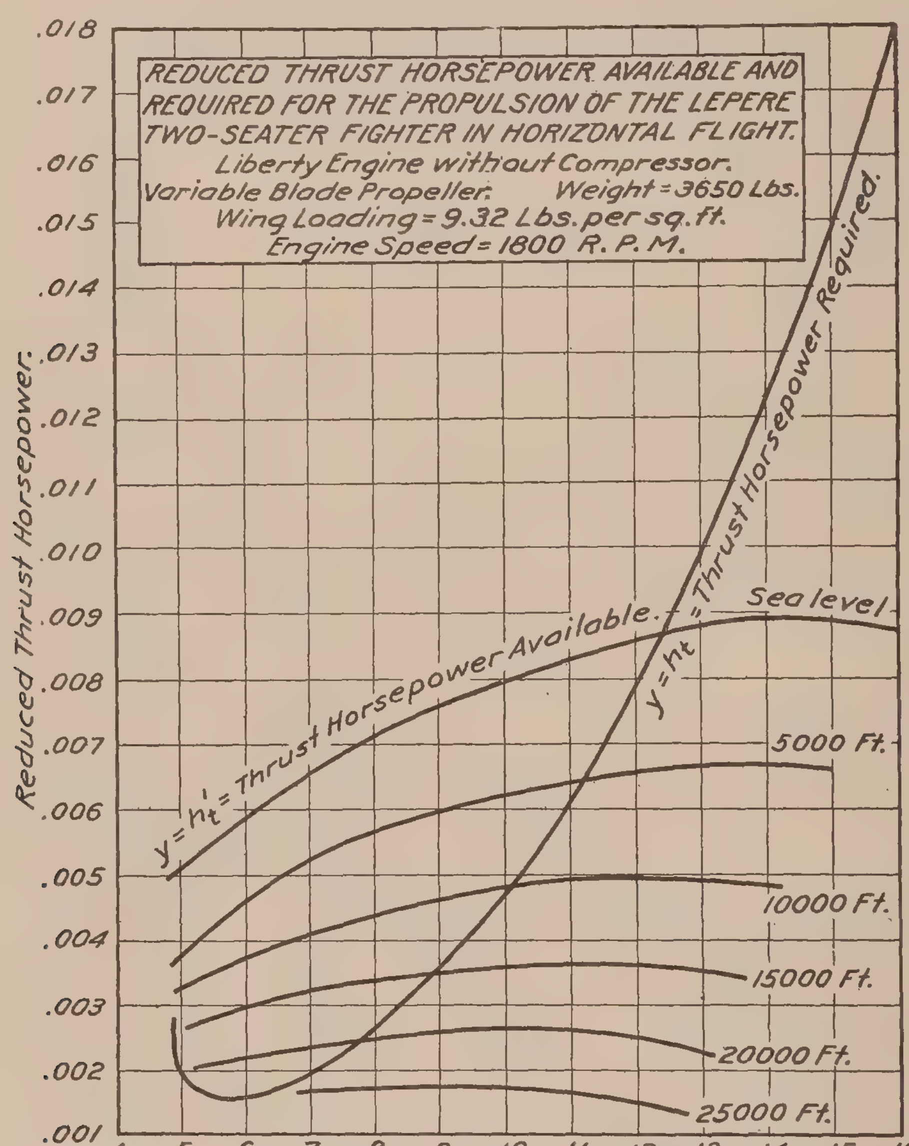


Fig. 21.

completely at all altitudes below 15,000 feet, the prime importance of the variable pitch airscrew for fighting military planes is evident.

9. FUEL ECONOMY: COMMERCIAL APPLICATIONS.

Airplane transportation will always be high speed transportation, and the commercial aeronautical engineer will always be interested in horizontal flight speeds, but he will always have to consider the question of fuel economy at the same time. It is therefore important to discover to what extent the high speeds which the supercharger offers are to be obtained at the cost of fuel waste.

Figure 25 shows theoretical curves for the relative fuel economy (relative distance traversed per pound of fuel) of the LePere plane equipped with the four different propelling plants discussed in the preceding articles. These curves were worked out with the aid of the graphs of figures 13 and 22. The computation was based on the assumption that the engine is wide open at all altitudes, and that the carburetor is adjusted for maximum power. The small variation in the fuel consumption of the engine with speed at sea level was neglected.

The graphs on figure 25 emphasize the fact that the prime controlling factor in determining the fuel economy is the angle of attack of the plane. The fuel economy is proportional to the product of the lift over drag ratio of the airplane, the efficiency of the propeller, and the recip-

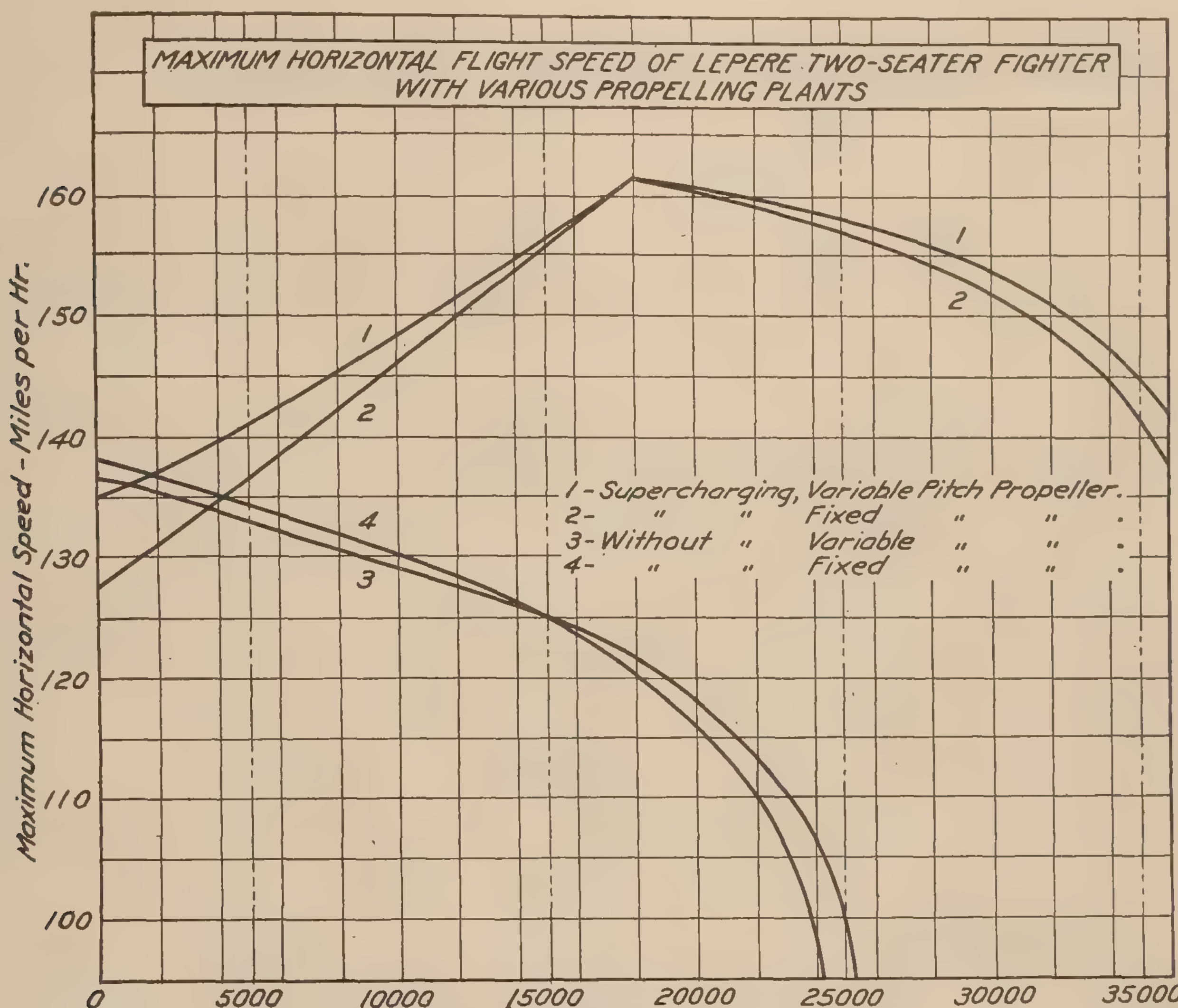
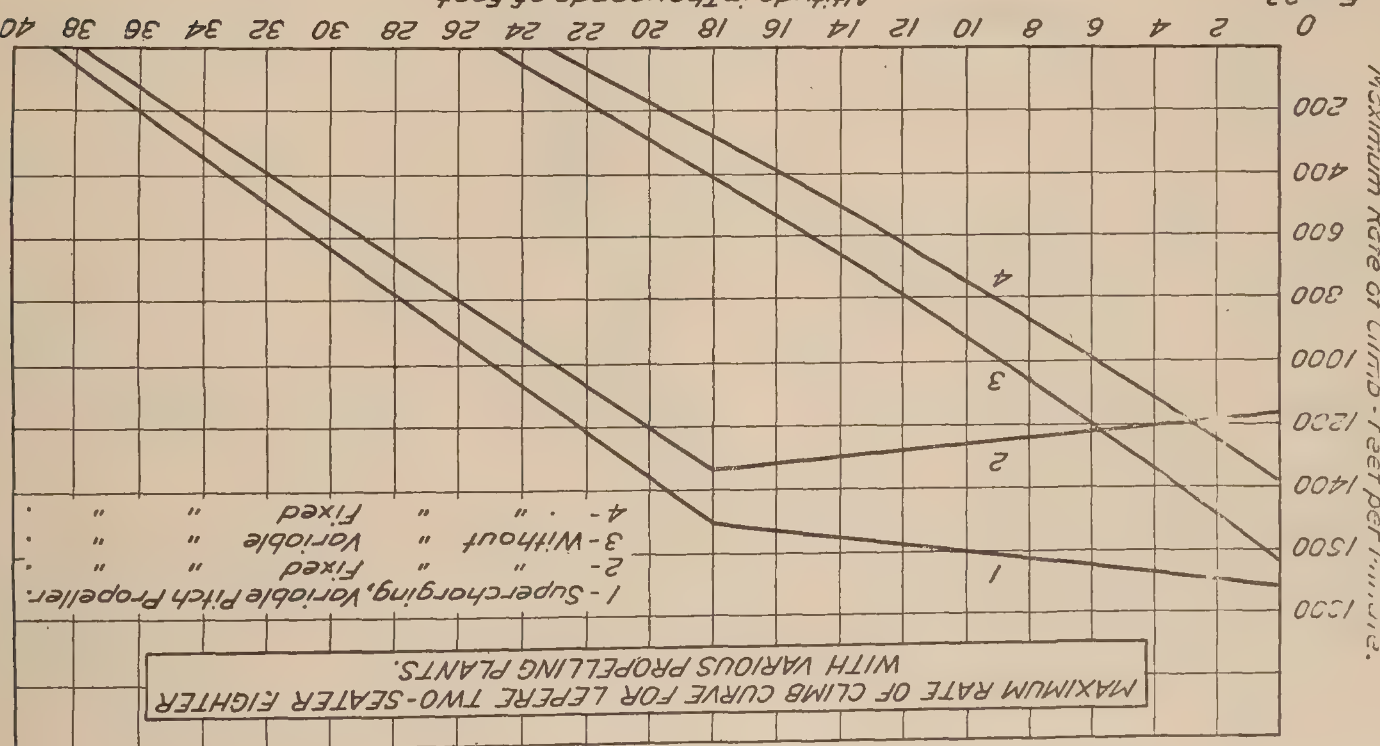


Fig. 22.

Fig. 23.



rocal of the specific fuel consumption of the engine. As the airplane climbs to greater and greater altitudes, the angle of attack becomes larger and larger, and the increase in the lift over drag ratio causes a decided increase in the fuel economy in spite of the steadily increasing

fuel consumption of the engine per brake horsepower hour. The variation in the propeller efficiency plays a relatively small part in the variation of the over-all fuel economy.

The lift over drag ratio has practically the same value at the ceiling for all cases, and consequently the difference between the maximum fuel economy for the plane without the supercharger, and with the supercharger, is due entirely to the differences in the mechanical efficiencies of the engines and in the propellers for the two cases.[9] The somewhat higher maximum fuel economy which the graphs indicate for the supercharging arrangement is due

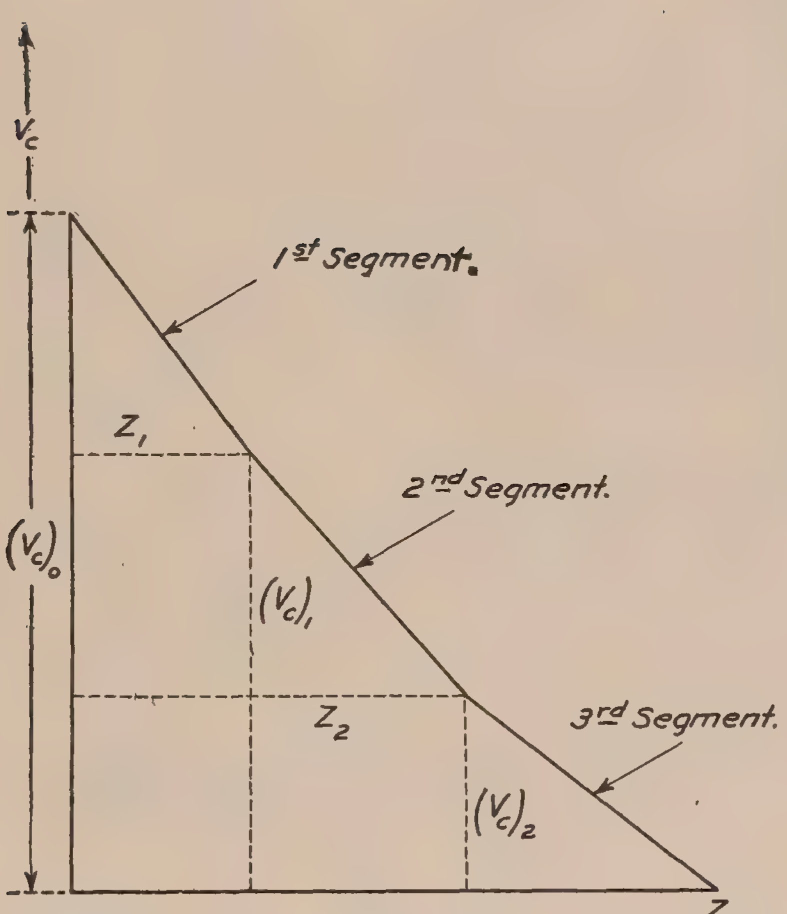


Fig. 23a.

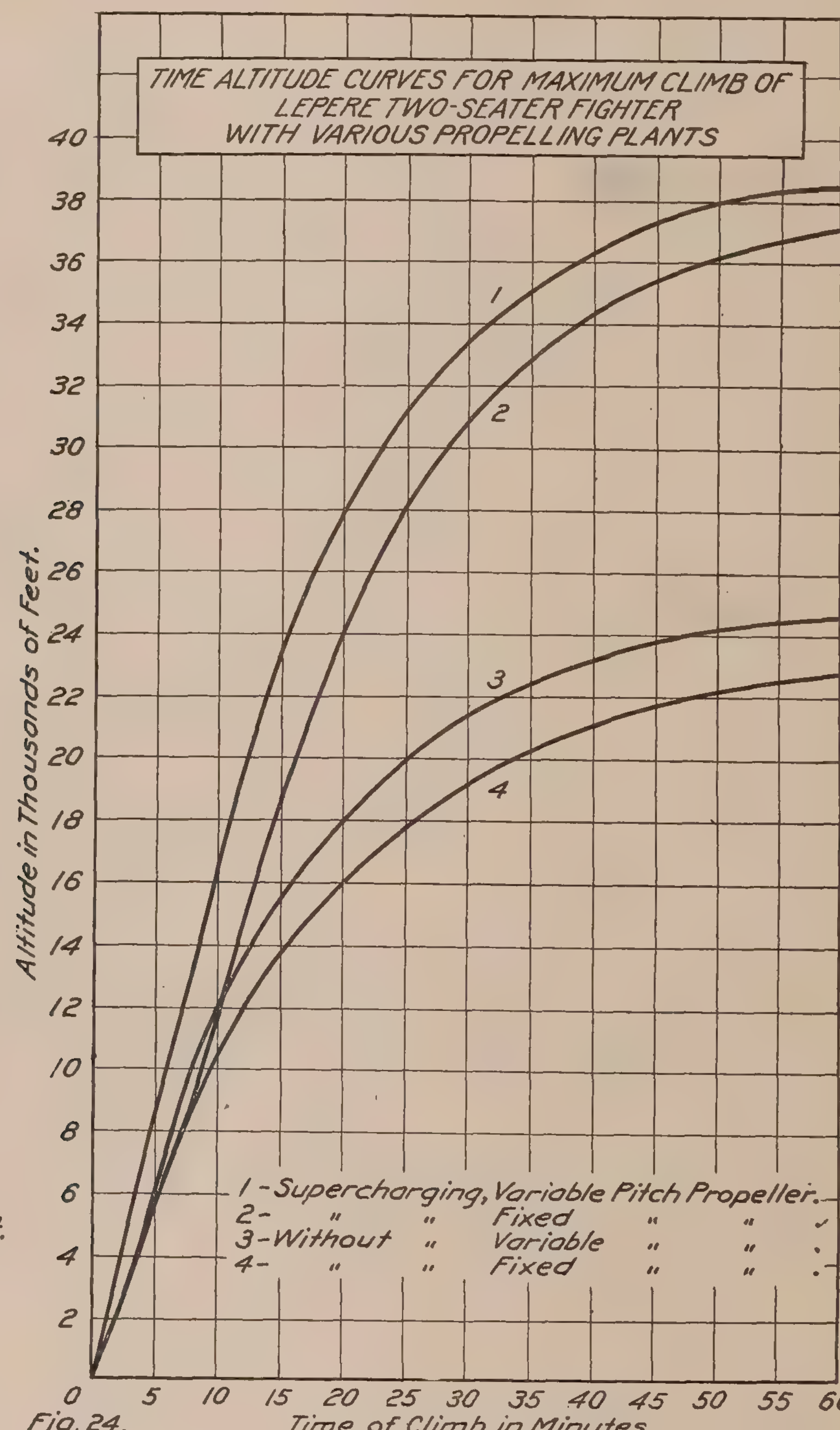


Fig. 24.

primarily to the fact that the power output near the ceiling, and hence the mechanical efficiency, is greater when the compressor is used.

As already stated, the curves of figure 25 are based on the assumption that the engine is wide open at all times. It is to be understood that fuel economy can be gained at any altitude below that of the ceiling at the expense of speed by throttling the engine or slowing down the compressor.

While the supercharging installation considered thus far is excellent for a military fighting plane, it very much overpowers the machine for commercial or military transportation. It is

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[9] Here we neglect the fuel losses due to poor carburetion and distribution which must commonly occur without the compressor as a result of the very low intake temperatures at great altitudes.

therefore desirable, in conclusion, to consider an installation adapted to the transportation of a load at a moderate speed with as great a fuel economy as possible.

In order to obtain an absolute maximum of fuel economy, the engine should drive the airplane at its most economical angle of attack while developing as large a percentage as possible of its sea level power, or mean effective pressure. Obviously this means that the most economical way to fly is near the ground with an engine which is barely able to lift the plane, but to obtain maximum economy in this manner would involve a large loss of speed, for the horizontal flight speed at any given angle of attack is inversely proportional to the square root of the air density.[10] To obtain maximum economy with a given plane and wing loading without sacrificing speed, the plane should operate at as great an altitude as is practicable.

The device of feeding warm compressed air from the supercharger to the aviators will in all probability make it possible to operate airplanes in the future at much greater altitudes

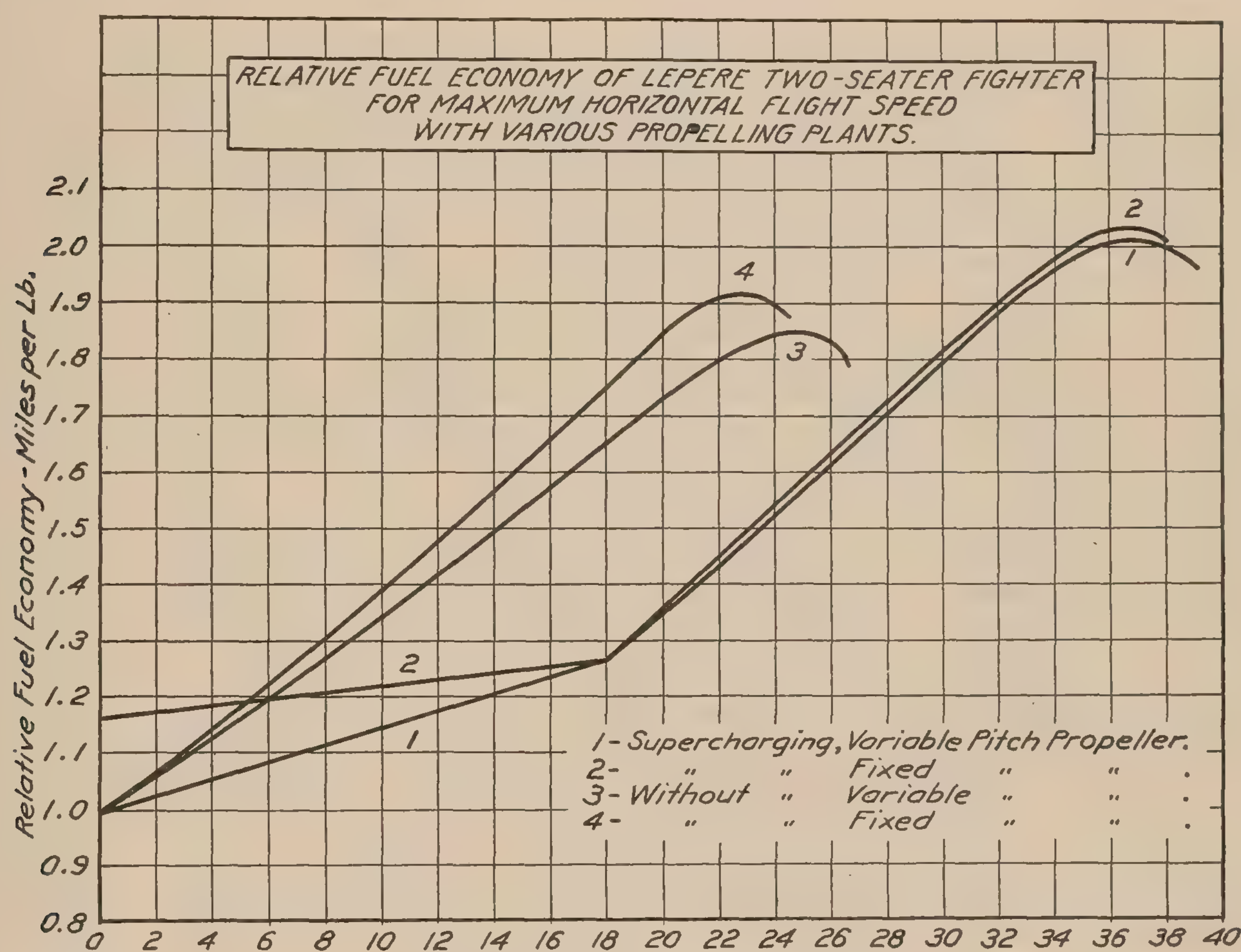

Fig. 25.

that at present. There will be a practical upper limit in any case, however, and we may, for the purposes of argument, set it at 25,000 feet. In order to see what the real commercial advantage of the supercharging compressor is, we therefore compare the weights and fuel economies of two engines developing equal power at 25,000 feet, one with the supercharger, the other without.

Let us assume that an airplane A, having the same lift and drag coefficients and the same wing loading as the Le Pere two-seater fighter, but larger and heavier, is to be driven with a horizontal flight speed of 120 miles an hour at 25,000 feet by the Liberty engine with supercharging compressor "all out."[11] The curve for the reduced thrust horsepower required (fig. 26) is the same as for the Le Pere. The ordinates of the new curves for the reduced thrust horsepower available are to be obtained from those of figure 18 (assuming a fixed blade propeller) through multiplication by the ratio of the weight of the Le Pere to the weight of A. The reduced

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[10] Compare equations (13) and (15), which combine to give $V=\sqrt{\frac{w\, g \cos\theta}{\rho\, K_y(\alpha)}}$. The speed of advance for any given plane and angle of attack can not be increased by increasing the wing loading on account of the necessity for preserving a moderate landing speed.

[11] The comparison would be essentially the same if the engines were assumed to operate partially throttled at 25,000 feet.

speed of advance corresponding to 120 miles an hour at 25,000 feet is 7.45. The corresponding reduced horsepower required is 0.0227. But the reduced horsepower available for the same propelling plant when installed on the Le Pere is 0.0455. (Fig. 18.) Hence the ratio of the weight of A to the weight of the Le Pere is 0.0455/0.0277, or 2.005. The weight of the Le Pere with the supercharger being 3,770 pounds, the weight of A works out to be 7,560 pounds. The reduced thrust horsepower available for the nonsupercharging Liberty engine when installed on a plane weighing 3,650 pounds is 0.0175 pounds at the speed and altitude in question. (Cf.

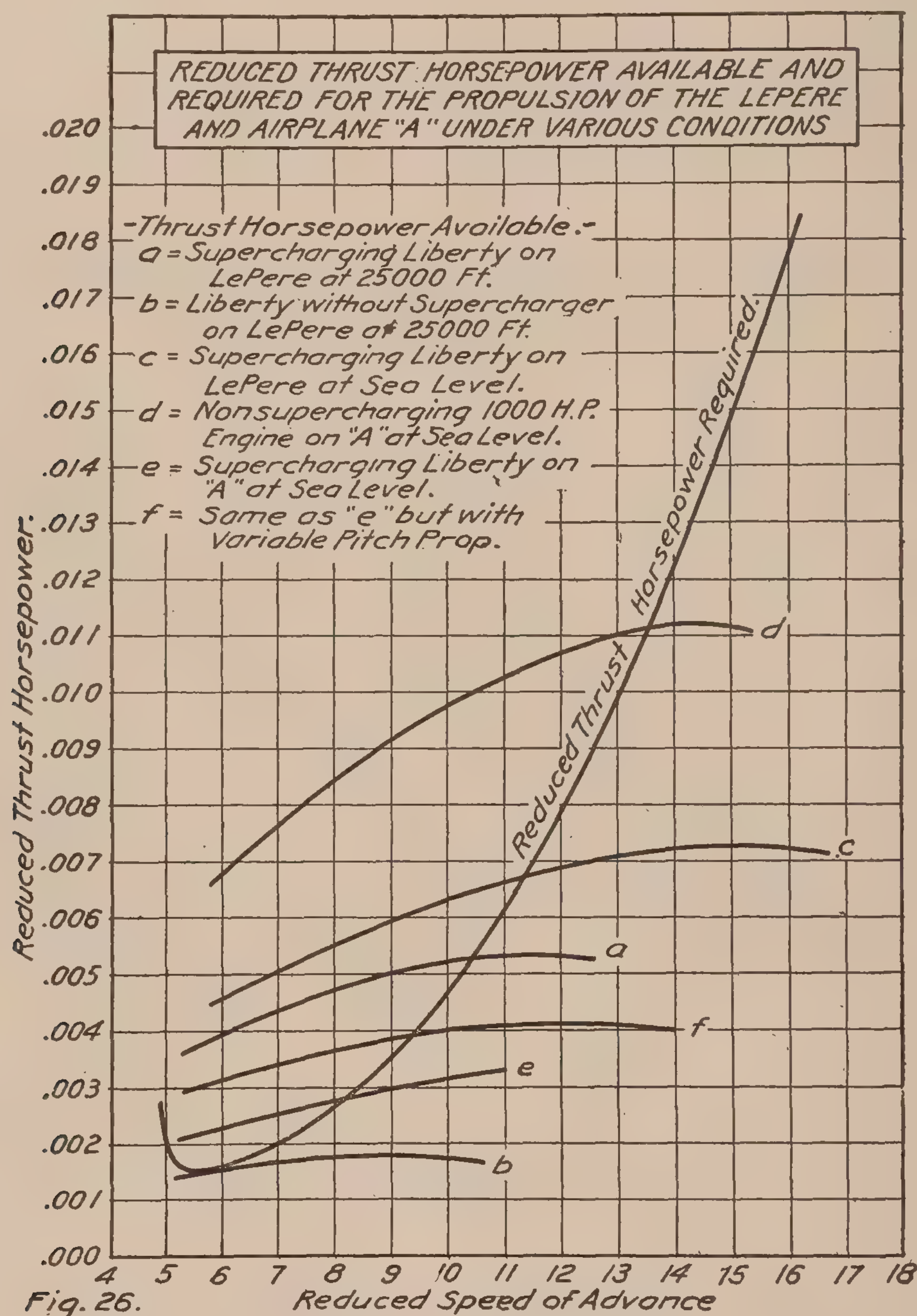

Fig. 26.

fig. 20.) Hence the reduced horsepower available for the nonsupercharging engine on plane A would be $0.0175 \times 3650/7560$, or 0.00844. This is 1/2.69 times the reduced horsepower required. Calling the nominal power of the Liberty engine 400, it is evident that the nominal power of a supercharging engine, capable of driving this plane at the assumed speed of 120 miles an hour at 25,000 feet, would be 400×2.69, or 1,075 horsepower. This comparison is somewhat unfair to the nonsupercharging engine, however, since the propellers assumed would make the engine speeds 1,670 revolutions per minute and 1,570 revolutions per minute for the supercharging and nonsupercharging cases, respectively. Assuming the same speed for both (at 25,000 feet), the nominal horsepower of the required nonsupercharging engine would be a trifle over 1,025. Thus the use of the compressor would increase the carrying capacity of plane A by an amount equal to the difference between the weight of a 1,000 horsepower engine and that of a 400 horsepower engine, minus 100 pounds, the weight of the turbine and compressor. This may be roughly estimated at from 900 to 1,000 pounds.

At the same time the curves of figure 13 show that the nonsupercharging engine will use 33 per cent more fuel per brake horsepower hour, and per mile. The one drawback of the small engine and supercharger as compared with the large engine would be in the excessively low climbing speed at sea level. This works out to be 257 feet per minute as compared with 2,145 feet per minute for the 1,000 horsepower engine. A variable pitch propeller would increase the sea-level climbing speed for the smaller engine and compressor to 563 feet per minute (a gain of 119 per cent) and the horizontal flight speed at sea level from 92 miles an hour to 105 miles an hour. This ability to more than double the sea-level climbing speed of a heavy plane with a high-power loading would be of great use in getting the machine off the ground and points to an important commercial application of the variable pitch propeller.

The fuel consumption per mile for the same airplane operating at sea level with a speed of 120 miles an hour would be 62 per cent greater than at 25,000 feet with the supercharging compressor.

It may be observed in conclusion that on account of the meagerness of the data available, the probable error involved in the present estimate of the performance of an airplane equipped with an engine and supercharging compressor is considerable. The calculated gains are so large, however, that there can be little doubt of the great value of the compressor both for military and commercial purposes.

REPORT No. 101.

APPENDIX.

NOTE ON THE CORRECTION OF THE PROPELLER THRUST COEFFICIENT CURVE FOR THE SLIP STREAM RESISTANCE.

Bairstow and Coales (British Advisory Committee for Aeronautics, Reports and Memoranda 474) have shown on the basis of an empirical formula for the resistance of the parts of an airplane in the slip stream, that it is possible to correct for the extra head resistance due to the slip stream effect by merely scaling down the thrust coefficient curve by a constant factor. In the following treatment of the slip stream effect (cf. note 6, part 2 of this report) the writer employs Dr. Warner's theoretical expression for the slip stream velocity [1] to derive a theoretical expression for the effective thrust coefficient. It turns out that the correction factor is not quite constant, but is nearly so for a consierable range of values of V/np.

Let R_s = resistance of portion of machine in slip stream;

R = resistance of portion outside slip stream;

R' = resistance which the entire machine would have if the slip stream velocity were equal to the speed of advance;

V_s = slip stream velocity in feet per second.

Substituting $R+R_s$ for X in equation (5) (part 2), we obtain

$$T=R+R_s+Y\tan\theta. \tag{a}$$

But

$$R_s=k_s\rho V^2{}_s/g, \tag{b}$$

where k_s is an easily calculable coefficient, and

$$R'=R+k_s\rho V^2/g. \tag{c}$$

Combining (b) and (c) with (a), we obtain

$$T-\frac{k_s\rho V^2}{g}\left[\left(\frac{V_s}{V}\right)^2-1\right]=R'+Y\tan\theta. \tag{d}$$

The left-hand member is equal to the thrust which would be required if there were no slip stream effect, and can properly be called the *effective* thrust. We denote it by the symbol T' thus:

$$T'=T-\frac{k_s\rho V^2}{g}\left[\left(\frac{V_s}{V}\right)^2-1\right]. \tag{e}$$

The substitution of the value of T in terms of the torque coefficient (equation 30, part 2) yields

$$T'=\frac{\rho n^2D^4}{g}\left\{t_c-\frac{k_s}{D^2}\left(\frac{V}{nD}\right)^2\left[\left(\frac{V_s}{V}\right)^2-1\right]\right\} \tag{f}$$

Let

$$t'_c=t_c-\frac{k_s}{D^2}\left(\frac{V}{nD}\right)^2\left[\left(\frac{V_s}{V}\right)^2-1\right]=t_c-\frac{k_s\sigma^2}{D^2}\left[\left(\frac{V_s}{V}\right)^2-1\right]\left(\frac{P}{D}\right)^2. \tag{g}$$

[1] "Slip Stream Velocities," by E. P. Warner, Report No. 71 of the National Advisory Committee for Aeronautics.

By definition, the torque coefficient is a quantity which, when multiplied by $\rho n^2D^4/g$, gives the true thrust. But when t'_c is multiplied by $\rho n^2D^4/g$, it gives the effective thrust. Hence t'_c plays the part of an effective thrust coefficient. It remains to show that t'_c like t_c is a function of σ only, for a given propeller and airplane.

Warner's momentum formula for the slip stream velocity is

$$T=0.636\frac{\rho}{g}D^2V_s(V_s-V),$$

or

$$t_cn^2D^2=0.636\ V_s\ (V_s-V). \qquad \text{(h)}$$

(h) is easily thrown into the form

$$\frac{V_s}{V}\left(\frac{V_s}{V}-1\right)=\frac{t_c}{0.636\,\sigma^2}\left(\frac{D}{P}\right)^2. \qquad \text{(i)}$$

Solving for V_s/V, we obtain

$$\frac{V_s}{V}=\frac{1}{2}\left[1+\sqrt{1+\frac{4t_c}{0.636}\left(\frac{D}{P}\right)^2\cdot\frac{1}{\sigma^2}}\right] \qquad \text{(j)}$$

Since t_c is a function of σ, it is evident that V_s/V, and therefore t'_c, is a function of σ. Equations (j) and (g) can be used for the evaluation of t'_c when the relationship between t_c and σ is known.

If the effective value of the thrust coefficient T_c as defined by the equation

$$T_c=\frac{100\ T}{\rho V^2D^2} \qquad \text{(k)}$$

is desired, equation (g) should be replaced by

$$T'_c=T_c-\frac{100k_s}{g\ D^2}\left[\left(\frac{V_s}{V}\right)^2-1\right]. \qquad \text{(l)}$$

Through the range of values of σ which are used in practice, the velocity ratio V_s/V is generally less than 1.5, and consequently the following approximation should be useful. Treating $(V_s/V-1)/2$ as a quantity small in comparison with unity, we can write:

$$\left(\frac{V_s}{V}\right)^2-1=2\frac{V_s}{V}\left(\frac{V_s}{V}-1\right) \qquad \text{(Approx.)}$$

Substituting this value for $\left(\frac{V_s}{V}\right)^2-1$ into (g), we obtain

$$t'_c=t_c\ (1-3.15\ k_s\ /D^2). \qquad \text{(m)}$$

Thus to a first rough approximation, the effective torque coefficient can be obtained from the true torque coefficient through multiplication by a constant correction factor.

Since the above approximate expression for $(V_s/V)^2-1$ is somewhat too large for all values of V_s/V, better results are obtained by reducing the coefficient of k_s/D^2 in (m) to the value 2.9. Thus

$$t_c'=t_c\ (1-2.9\ k_s/D^2). \qquad \text{(m')}$$

This equation also holds if T_c' and T_c are substituted for t_c' and t_c, respectively.

The accompanying table shows the percentage error in the correction to the thrust and thrust coefficient for various values of σ due to the above approximation, as computed for the propeller of figure 16. The percentage error in the thrust coefficient itself would, of course, be much smaller.

$\sigma=\frac{V}{nP}$	V_s/V.	Per cent error in $(V_s/V)^2-1$.	Propeller efficiency.
0.3	1.836	19.5	0.47
.4	1.523	11.1	.59
.5	1.340	5.6	.70
.6	1.221	1.4	.77
.7	1.139	−1.7	.805
.8	1.080	−3.9	.78
.9	1.035	−5.8	.63

The effective efficiency of the propeller for any value of σ, or of $V/(nD)$, is decreased in the same ratio as the corresponding thrust coefficient.

SUMMARY OF NOTATION FOR PART 2.

H = brake horsepower of engine.
H_t = thrust horsepower of propeller.
U, V = speed of advance in miles per hour and feet per second, respectively
V_c = maximum climbing speed in feet per minute.
ρ = density of air in pounds per cubic foot.
d = relative density of air.
S = radiator area.
t = atmospheric temperature (Fahrenheit).
t' = mean temperature of water in radiator.
α = angle of attack.
θ = angle of climb.
g = acceleration of gravity.
Y = total lift of airplane in pounds.
X = total drag of airplane in pounds.
T = propeller thrust in pounds.
W = weight of airplane in pounds.
A = wing area in square feet.
w = wing loading in pounds per square foot.
$K_y(\alpha)$ = lift coefficient for entire machine.
$K_x(\alpha)$ = drag coefficient for entire machine.
h_t = reduced thrust horsepower required. (Cf. Equation (12).)
h_t' = reduced thrust horsepower available. (Cf. Equation (18).)
u = reduced speed of advance. (Cf. Equation (13).)
n = propeller speed in revolutions per second.
D = propeller diameter in feet.
P = propeller experimental mean pitch in feet.
η = propeller efficiency.
Q = propeller torque in pounds-feet.
q_c = torque coefficient as defined by equation (20).
Q_c = torque coefficient as defined by equation (26).
t_c = thrust coefficient as defined by equation (30).
$\sigma = V/(nP)$.
K_q, K_t = constants defined by equations (27) and (28).
$F(\sigma)$ = function defined by equation (31).

$\varphi(P/P_m)$, $\zeta(P/P_m)$ = functions defined by the graphs of figure (18).

τ = time of climb in minutes.

z = altitude in feet.

V_s = slip stream velocity, in feet per second.

R = resistance of portion of plane outside slip stream.

R_s = resistance of portion of plane in slip stream.

R' = resistance which entire machine would have if slip stream velocity were equal to V.

k_s = constant defined by (c) (Appendix).

T' = effective thrust.

t_c' = effective thrust coefficient.

REPORT No. 102

PERFORMANCE OF A LIBERTY 12 AIRPLANE ENGINE

By S. W. SPARROW and H. S. WHITE
Bureau of Standards

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REPORT No. 102.

PERFORMANCE OF A LIBERTY 12 AIRPLANE ENGINE.

By S. W. Sparrow and H. S. White.

Bureau of Standards.

RÉSUMÉ.

This report, on the complete performance test of the Liberty 12 airplane engine, was submitted for publication to the National Advisory Committee for Aeronautics by the Bureau of Standards. The tests described were conducted in the altitude chamber of the dynamometer laboratory of the Bureau of Standards. The program of tests was planned in cooperation with the Engineering Division of the Air Service of the United States Army, so as to yield that information which is considered of most importance in determining the value of an engine for aviation. The particular engine used in these tests was assembled by the Engineering Division at McCook Field and subjected to the standard dynamometer test for operation at ground level, then shipped to the Bureau of Standards, and mounted in the altitude chamber without overhaul. After the altitude tests it was then returned to McCook Field for such flight tests as might be desired. Though the question of durability is of vital importance, it can be determined with much less equipment. Since an aviation engine is comparatively short lived, the tests were purposely made as short as was consistent with the securing of the above information so that the engine might be left in condition for many hours of flight tests.

The following tests were made:

1. A full power run at ground altitude at speeds from 1,200 to 2,000 revolutions per minute.
2. An altitude power run at full throttle and at speeds of 1,600 and 1,700 revolutions per minute from ground altitude to 25,000 feet (7,620 meters) in steps of 5,000 feet (1,520 meters).
3. Propeller load runs, in which the dynamometer load and engine throttle were so adjusted as to produce approximately the same engine load as would be imposed by a propeller at speeds from 1,200 revolutions per minute to the normal full load propeller speed of 1,700 revolutions per minute. These were taken at altitudes of 5,000, 10,000, and 15,000 feet (1,520, 3,050, 4,570 meters).
4. Friction horsepower runs at ground altitude and at 15,000 feet (4,570 meters).

RESULTS.

Some of the outstanding results are given in the accompanying tables. Correcting the results to the standard barometric pressure of 29.9 inches (76 centimeters) of mercury gives a maximum brake horsepower of about 440 (446 metric horsepower) at a speed of 1,900 revolutions per minute, and a maximum brake mean effective pressure of 124 pounds per square inch (8.7 kilograms per square centimeter) at 1,200 revolutions per minute. The mechanical efficiency varies from 90 per cent to 84 per cent from speeds of 1,200 revolutions per minute to 1,900 revolutions per minute, while the brake thermal efficiency, based on the lower calorific value of the fuel maintains a constant value of 25 per cent over the same speed range.

Above 15,000 feet (4,570 meters) altitude the carburetor altitude control is inadequate to maintain the proper mixture ratio. The effect of the rich mixture resulting is to cause a

decrease in power below that which would be expected from the very nearly linear relation which brake horsepower and mean effective pressure bear to density as long as the mixture ratio can be adjusted for maximum power. The volumetric efficiency at 1,600 revolutions per minute decreases steadily with altitude from a value of 86 per cent at the ground to a value of 78 per cent at 20,000 feet altitude.

Under the conditions of this test at an air density of 0.0405 pound per cubic foot or 0.65 kilogram per cubic meter, the brake horsepower is about 42 per cent of that at the ground, and the indicated horsepower is approximately 47 per cent of that at the ground.

CONCLUSIONS.

Such information as is contained in a report of this kind is of most value when compared with similar tests of other engines. It then furnishes not only a basis for comparing the relative value of two engines, but also a means for explaining the causes of the superiority of one engine over another in any particular phase of performance. This test in itself, however, yields two conclusions that seem of primary importance:

1. The provision on the carburetors for adjusting the mixture ratio to suit altitude conditions is inadequate for altitudes above 15,000 feet (4,570 meters).

2. In making any changes in this engine to improve its altitude performance—that is, to decrease the rate at which the brake horsepower falls off with altitude—the two methods which offer most hope of success are to increase the mechanical efficiency by decreasing friction horsepower and to make such changes as will prevent the present decrease of volumetric efficiency with increase of altitude.

TABLE A.—*English units.*

Ground runs. Full power.

Approximate altitude (feet).	Revolutions per minute.	Brake mean effective pressure (pounds per square inch).	Brake horsepower.	Pounds of fuel per brake horsepower hour.	Carburetor air temperature (°F.).	Air density (pounds per cubic foot).	Volumetric efficiency (per cent).	Thermal efficiency (per cent).	Pounds air per pound fuel, ±0.2.
1,000	1,220	118.3	302	0.53	59	0.073	85	25	13.3
1,000	1,410	118.0	345	.53	59	.073	87	26	14.0
1,000	1,600	114.2	380	.54	54	.073	84	25	13.6
1,000	1,800	107.6	404	.54	59	.072	83	25	13.9
1,000	1,790	109.4	408	.52	59	.073	84	25	14.7
1,000	1,890	103.8	409	.55	60	.072	82	24	14.1
1,000	1,800	110.4	414	.53	59	.073	83	25	14.1
1,000	1,900	106.3	419	.54	59	.073	81	25	14.1
1,000	2,000	98.2	409	.61	59	.073	78	22	13.0

TABLE B.—*English units.*

Altitude runs. Full power.

Approximate altitude (feet).	Revolutions per minute.	Brake mean effective pressure (pounds per square inch).	Brake horsepower.	Pounds of fuel per brake-horsepower hour.	Carburetor air temperature (°F.).	Air density (pounds per cubic foot).	Volumetric efficiency (per cent).	Thermal efficiency (per cent).	Pounds air per pound fuel, ±0.2.
Ground.	1,620	116.5	393	0.53	59	0.075	86	25	14.3
Ground.	1,700	117.0	414	.51	59	.075	85	26	14.5
5,000	1,690	100.0	352	.53	41	.066	83	25	14.3
5,000	1,610	101.2	339	.58	41	.066	85	23	13.0
10,000	1,600	82.3	273	.58	26	.057	83	23	13.5
10,000	1,690	74.6	262	.69	29	.056	81	20	12.2
10,000	1,710	79.2	282	.54	26	.057	81	25	14.5
15,000	1,690	59.8	209	.67	14	.048	76	20	12.5
15,000	1,590	65.2	216	.57	12	.048	81	23	14.3
20,000	1,570	50.9	166	.67	8	.040	76	20	12.2
20,000	1,700	44.3	157	.81	32	.038	77	16	11.1
25,000	1,700	24.8	88	1.41	19	.032	73	9	9.1
25,000	1,590	30.4	101	1.11	11	.033	74	12	10.0

TABLE C.—*English units.*

Ground runs.

Revolutions per minute.	Brake horsepower.	Friction horsepower.	Indicated horsepower.	Mechanical efficiency (per cent).	Air density (pounds per cubic foot).
1,200	295	33	328	90	0.073
1,400	344	43	387	89	.073
1,600	385	55	440	87	.073
1,800	415	69	484	86	.073
1,900	419	77	496	84	.073
2,000	410	90	500	82	.073

TABLE D.—*English units.*

Altitude runs.

Air density (pounds per cubic foot).	Brake horsepower.	Friction horsepower.	Indicated horsepower.	Mechanical efficiency (per cent).	Revolutions per minute.	B.h.p. ÷(b.h.p. at 0.076 density).
0.076	403	55	458	88	1,600	1.00
.066	336	54	390	86	1,600	.83
.057	276	52	328	84	1,600	.68
.048	216	51	267	81	1,600	.53
.040	163	50	213	77	1,600	.40
.033	96	49	145	66	1,600	.24

TABLE A.—*Metric units.*

Ground runs. Full power.

Approximate altitude (meters).	Revolutions per minute.	Brake mean effective pressure (kilograms per square centimeter).	Brake horsepower.	Kilograms of fuel per brake horsepower hour.	Carburetor air temperature (°C.).	Air density (kilograms per cubic meter).	Volumetric efficiency (per cent).	Thermal efficiency (per cent).	Kilograms air per kilogram fuel, ±0.2.
305	1,220	8.3	306	0.24	15	1.17	85	25	13.3
305	1,410	8.3	350	.24	15	1.17	87	26	14.0
305	1,600	8.0	385	.24	12	1.17	84	25	13.6
305	1,800	7.6	419	.24	15	1.15	83	25	13.9
305	1,790	7.7	414	.23	15	1.16	84	25	14.2
305	1,890	7.3	415	.25	16	1.16	82	24	14.1
305	1,800	7.8	420	.23	15	1.17	83	25	14.1
305	1,900	7.5	425	.24	15	1.17	81	25	14.1
305	2,000	6.9	415	.27	15	1.16	78	22	13.0

TABLE B.—*Metric units.*

Altitude runs. Full power.

Approximate altitude (meters).	Revolutions per minute.	Brake mean effective pressure (kilograms per square centimeter).	Brake horsepower.	Kilograms of fuel per brake horsepower hour.	Carburetor air temperature (°C.).	Air density (kilograms per cubic meter).	Volumetric efficiency (per cent).	Thermal efficiency (per cent).	Kilograms air per kilogram fuel, ±0.2.
Ground.	1,620	8.2	399	94	15	1.20	86	25	14.3
Ground.	1,700	8.2	420	96	15	1.20	85	26	14.5
1,520	1,690	7.0	357	85	4	1.06	83	25	14.3
1,520	1,610	7.1	344	90	5	1.06	85	23	13.0
3,050	1,600	5.8	277	72	− 3	.91	83	23	13.5
3,050	1,690	5.2	266	82	− 2	.90	81	20	12.2
3,050	1,710	5.1	286	69	− 3	.91	81	25	14.5
4,570	1,690	4.2	212	63	−10	.77	76	20	12.5
4,570	1,590	4.6	219	56	−11	.77	81	23	14.3
6,040	1,570	3.6	168	51	13	.64	76	20	12.2
6,040	1,700	3.1	159	58	0	.61	77	16	11.1
7,620	1,700	1.7	89	56	− 7	.52	73	9	9.1
7,620	1,590	2.1	102	51	−12	.53	74	12	10.0

TABLE C.—*Metric units.*

Ground runs.

Revolutions per minute.	Brake horsepower.	Friction horsepower.	Indicated horsepower.	Mechanical efficiency (per cent).	Air density (kilograms per cubic meter).
1,200	299	33	332	90	1.17
1,400	349	44	393	89	1.17
1,600	391	56	447	87	1.17
1,800	421	70	491	86	1.17
1,900	425	78	503	84	1.17
2,000	416	91	507	82	1.17

TABLE D.—*Metric units.*

Altitude runs.

Air density (kilograms per cubic meter).	Brake horsepower.	Friction horsepower.	Indicated horsepower.	Mechanical efficiency (per cent).	Revolutions per minute.	B.h.p. ÷(b.h.p. at 1.22 density).
1.22	409	56	465	88	1,600	1.00
1.06	341	54	395	86	1,600	.83
.91	280	53	333	84	1,600	.68
.77	219	52	271	81	1,600	.53
.64	165	51	216	77	1,600	.40
.53	97	50	147	66	1,600	.24

OBJECT OF TEST.

This test was made to determine the performance of a Liberty 12 airplane engine. The engine was operated under such conditions as would yield sufficient information to determine its value for aviation use. The test was typical of that ordinarily made on a new type of engine in that the completeness of the tests was sacrificed for the sake of reducing the actual running time of the engine. Such a procedure does not materially reduce the life of the engine, and leaves it in good condition for actual flight work.

DESCRIPTION OF ENGINE AND APPARATUS.

(*a*) *Engine.*—The engine used was a Liberty 12, U. S., No. 22, 519 standard in every respect. This is a V-type motor with 12 water-cooled cylinders. It has a bore of 5 inches (12.7 centimeters), stroke of 7 inches (17.8 centimeters), and compression ratio of 5.4. The Zenith carburetor used is provided with a manually operated valve for decreasing the fuel flow to maintain the correct mixture under altitude conditions. Mobile B oil was used on the full-power run on the ground and Liberty aero oil No. 3501 thereafter. Both oils were satisfactory, the reason for the change being of no importance in this test. The X gasoline used conforms to the Aircraft Production Board's Specification 3512 for the American Expeditionary Forces, 1918. A distillation curve of the fuel according to the standard Bureau of Mines method is given on curve sheet 15.

(*b*) *Apparatus.*—The engine was tested in the altitude chamber of the dynamometer laboratory of the Bureau of Standards. A complete description of this apparatus is given in report No. 44 of the National Advisory Committee for Aeronautics (No. 52 of the Bureau of Standards Automatic Power Plant Series). The air from this chamber can be exhausted until its pressure is reduced to that of the altitude desired, while at the same time the temperature of the air supplied to the engine can be reduced to approximately that prevailing at the given altitude. Outside the chamber, apparatus is provided for measuring power, fuel consumption, and all necessary temperatures and pressures.

PROGRAM OF TESTS.

1. A run was made with wide-open throttle at ground altitude at speeds from 1,200 to 2,000 revolutions per minute. At each speed the spark advance was adjusted for maximum power. The carburetor was adjusted at each speed to give the least fuel consumption possible with maximum power. This was accomplished by adjusting the carburetor until the speed and torque showed that the maximum power had been obtained. The mixture was then made so lean that the torque dropped appreciably and then enriched just enough to restore the maximum power.

2. A run was made with wide-open throttle at speeds of 1,600 revolutions per minute and 1,700 revolutions per minute at the ground, and at altitudes of 5,000, 10,000, 15,000, 20,000, and 25,000 feet (1,520, 3,050, 4,570, 6,040, and 7,620 meters). At each speed and altitude the spark advance and carburetor were adjusted as for the ground run with the exception that carburetor limitations prevented the desired adjustment at the latter two altitudes.

3. A series of runs was made at altitudes of 5,000, 10,000, and 15,000 feet (1,520, 3,050, and 4,570 meters) at speeds of 1,700, 1,600, 1,500, 1,400, 1,300, and 1,200 revolutions per minute. In these runs the dynamometer and throttle were so adjusted as to put a load on the engine at each speed equal to that which would be imposed by a propeller which would absorb the full power of the engine at 1,700 revolutions per minute. In runs of this type it is assumed that the horsepower of a propeller varies as the cube of the speed. Thus if 1,700 be the normal revolutions per minute of the propeller, the horsepower at 1,400 revolutions per minute will be $\frac{1400^3}{1700^3}$ times the horsepower at 1,700 revolutions per minute. In these runs the spark and carburetor were adjusted at 1,700 revolutions per minute at each altitude as in the preceding runs, but were not altered for the other speeds.

4. A series of friction horsepower runs was made at speeds from 1,200 to 2,000 revolutions per minute both at the ground and at an altitude of 15,000 feet (4,570 meters). In these runs the engine was operated under power until oil and water temperatures were normal, and then it was driven by the dynamometer and the power input measured.

METHOD OF OBTAINING RESULTS.

The results of the tests are given in Tables 1 to 9. A detailed record of the complete test procedure of the laboratory, both in securing data and computing results, is in preparation, so that a brief explanation here will suffice. The run numbers are those that were used on the original sheets to designate the different runs.

Altitude was determined by means of the pressure altitude relation adopted by the Aeronautical Instruments Section of the Bureau of Standards and given on curve sheet 16. The pressures used were measured at the carburetor entrance. The engine torque was measured on a 21-inch (53-centimeter) arm on the dynamometer, and from this value the torque in pounds feet, brake mean effective pressure, and brake horsepower are calculated. The brake horsepower calculation requires the speed which was obtained with a revolution counter and stop watch. Temperatures were all measured with thermo couples and pressures with U-type manometers.

The volume of air used was measured by a Venturi meter calibrated in place with a carefully tested Thomas meter. From measurements of temperature and pressure, air density, and then weight of air used per unit, time is computed.

The volumetric efficiency is the ratio of the volume of air which the engine actually takes in per cycle of two revolutions to the total piston displacement of the engine per stroke. The volume of the air used is determined for the pressure and temperature measured at the carburetor entrance.

The thermal efficiency is the ratio of the heat equivalent of brake horsepower to the heat equivalent of fuel supplied. The lower heating value of the fuel is used, which for gasoline is 18,940 British thermal units per pound (34,100 calories per gram).

In calculating the heat distribution on Table 2, the higher heating value of the fuel (20,320 B. t. u. per pound; 36,600 calories per gram) is used.

Heat in the exhaust is obtained by an exhaust calorimeter method and the residual heat by difference. The latter includes the heat equivalent of the unburned fuel that goes out the exhaust. Power developed by the burning of the lubricating oil has been neglected in heat balances chiefly because of the difficulty in determining just how much of the oil consumed is actually burned on the power stroke.

The brake horsepower and brake mean effective pressure obtained on the ground run are converted to standard barometric pressure by multiplying the values actually obtained by the ratio of 29.9 to the actual barometric pressure in inches of mercury.

The results shown in Table 9 are taken from the curves at given speeds. The indicated horsepower is obtained by adding the brake horsepower to the friction horsepower. The mechanical efficiency is obtained by dividing the brake horsepower by the indicated horsepower. In obtaining the friction horsepower at different densities its value at 1,600 revolutions per minute at the ground and at 15,000 feet (4,570 meters) was taken, and it was assumed to vary linearly with air density between these points. Previous tests show this to be close enough to the true relation to justify this assumption.

RESULTS.

The more important results of the ground tests are shown on curves 1 to 5, inclusive. The maximum brake mean effective pressure of about 118 pounds per square inch (8.3 kilograms per square centimeter) was attained at 1,200 revolutions per minute. The maximum brake horsepower occurs at about 1,900 revolutions per minute, which is the peak of the curve, the power falling off rapidly thereafter. The atmospheric pressure was such as would be equivalent to an altitude of about 1,000 feet (305 meters), and the results that would be expected under standard barometric pressure are shown on sheet 2. This shows a maximum brake mean effective pressure of 124 pounds per square inch (8.7 kilograms per square centimeter) and maximum brake horsepower of 440 (446 metric) horsepower. The reason for the "flattening out" or peaking of the brake horsepower speed curve is due usually to two major causes—the increase in friction horsepower with speed and the decrease in volumetric efficiency. A study of the indicated horsepower speed curve on sheet No. 3 shows a maximum at about 2,000 revolutions per minute. This curve was obtained by adding the friction horsepower to the brake horsepower at the different speeds, hence the flattening of this curve is due to decrease in volumetric efficiency. The curve at the bottom of the sheet shows how closely the power developed is related to the volumetric efficiency of the engine. On this curve the ratio of the indicated horsepower at each speed to the indicated horsepower at 2,000 revolutions per minute has been plotted, and also the ratio of pounds of air used per hour to the pounds of air used per hour at 2,000 revolutions per minute. The curves are practically identical. In studying the mixture-ratio curve plotted from values taken from curves of pounds of air per hour (sheet 4), it must be remembered that the mixture and pounds of fuel per hour were adjusted by hand at each speed, so that the shape of the curve and the values themselves in no way represent a carburetor characteristic. It is not clear at the present time why these values should be the ones to give maximum power and minimum fuel consumption, but the accumulation of data of this sort, together with further research based definitely on this subject, should furnish an explanation. On sheet 5 is shown the heat distribution. At 1,700 revolutions per minute, the normal speed of the engine, the heat in the fuel supplied is about 4.3 times that realized in brake horsepower, and the heat in the jacket is about half that developed in brake horsepower; under the same conditions the heat in the exhaust is nearly 1.7 times and residual is about 1.1 times that realized in brake horsepower. It should be remembered that the residual heat is the difference between the heat in the fuel and that which appears in brake horsepower, in the jacket, and in the exhaust. Hence the residual heat includes the heat value of the unburned fuel which goes out of the exhaust.

The curves from 6 to 10, inclusive, show the effect of change in altitude on engine performance. Since it is the change in air density with change in altitude which is the fundamental cause of these changes, it is against air density that these curves are plotted. For convenience in interpretation of the results, from the standpoint of barometric pressure, vertical lines have been drawn through the plotted points and the approximate barometric pressure noted.

For a proper understanding of the altitude curves, the curve of pounds of fuel per brake horsepower hour on curve sheet 6 and pounds of air per pound of fuel curve on sheet 9 should be noted. It will be observed that the mixture became extremely rich as the altitude was increased above 15,000 feet (4,570 meters), due to the fact that the adjustment was at its leanest position at this altitude, and it was impossible to secure the desired mixture beyond this point. The result of this richness of mixture of course manifested itself in extreme fluctuations of speed and torque, and excessive fouling of spark plugs. It will be noted on curve sheet 6 that the brake horsepower and brake mean effective pressure vary linearly with density up to the point where the abnormal mixture ratio results in their decrease. On curve sheet 7 is plotted the ratio of the indicated horsepower, pounds of air per hour and brake horsepower at a given density to their corresponding values at a density of 0.076 (approximately ground altitude). It is seen here, again, that as long as the mixture ratio was held within reasonable limits the percentage change in indicated horsepower was the same as the percentage change in weight of air received by the engine.

The rate of decrease of brake horsepower over indicated horsepower is more rapid with increase of altitude as the friction horsepower does not decrease nearly as rapidly as the brake horsepower and therefore becomes a greater and greater per cent of the brake horsepower. It is of interest to note that on this engine the volumetric efficiency drops steadily with altitude to the extent of a 12 per cent decrease at 25,000 feet (7,620 meters) over what it was at the ground, this, of course, being a vital factor in causing the decrease of power. In studying the curves on sheet 10, it must be remembered that it is the carburetor that is directly responsible for the high "heat in fuel over heat in brake horsepower" and "residual heat over heat in brake horsepower" values, and that indirectly it is responsible for the final high values of "heat in the exhaust over heat in brake horsepower" and "heat in jacket over heat in brake horsepower," through the resulting low power.

Comparison of the "Propeller load" curves on sheets 11 and 12 will show that fuel-consumption curve is influenced strongly by the carburetor characteristics as shown by pounds of air over pounds of fuel curve, it being remembered that the carburetor was only adjusted for 1,700 revolutions per minute in these cases.

CONCLUSIONS.

The greatest value of such data as is contained herein can only be realized by its comparison with a number of results obtained on other engines. This not only enables the engines to be judged as to their relative value for a given type of work but also indicates reasons for superiority in performance of one type over another.

There are two outstanding conclusions, however, to be drawn from the test itself—namely, that the carburetor control is inadequate above 15,000 feet (4,570 meters); that the most promising line of development for improved altitude performance lies in increased mechanical efficiency through decreased friction horsepower, and in such changes as will prevent the decrease in volumetric efficiency with increase in altitude.

TABLE I.—*English units.*

Ground runs. Full power.

Run No.	Approximate altitude (feet).	Revolutions per minute.	Torque (pound feet).	Brake mean effective pressure.	Brake horsepower.	Pounds of fuel per hour.	Pounds of fuel per brake horsepower hour.	Temperature (° F.). Oil. Inlet.	Oil. Outlet.	Jacket water. Inlet.	Jacket water. Outlet.	Carburetor air.	Oil pressure (pounds per square inch).	Manifold suction (inches hg., cyl. 4-5-6 R.).	Barometric pressure (inches hg.).
5 A...	1,000	1,220	1,292	118.3	302	162	0.53	81	142	147	167	59	35	0.6	28.6
6 A...	1,000	1,410	1,289	118.0	345	182	.53	84	156	137	155	59	36	.8	28.6
7 A...	1,000	1,600	1,248	114.2	380	206	.54	88	164	150	165	54	40	1.0	28.3
8 A...	1,000	1,800	1,176	107.6	404	219	.54	88	164	134	151	59	41	1.3	28.1
9 A...	1,000	1,790	1,195	109.4	408	210	.52	88	162	133	151	59	41	1.1	28.3
10 A...	1,000	1,890	1,133	103.8	409	226	.55	98	175	138	152	60	41	1.3	28.3
1 B...	1,000	1,800	1,204	110.4	414	218	.53	80	160	143	162	59	43	1.1	28.5
2 B...	1,000	1,900	1,162	106.3	419	224	.54	91	176	145	163	59	40	1.3	28.4
3 B...	1,000	2,000	1,074	98.2	409	248	.61	98	194	147	165	59	41	1.4	28.3

TABLE II.—*English units.*

Ground runs. Full power.

Run No.	Heat distribution based on brake horsepower. Heat in fuel÷(heat in b.h.p.).	Heat in jacket÷(heat in b.h.p.).	Heat in exhaust÷(heat in b.h.p.).	Residual heat÷(heat in b.h.p.).	Heat distribution based on heat in fuel. Brake horsepower (per cent).	Jacket (per cent).	Exhaust (per cent).	Residual (per cent).	Air density (pound per cubic foot).	Pounds of air per hour.	Volumetric efficiency (per cent).	Thermal efficiency (per cent).	Pounds of air per pound of fuel ±0.2.
5 A....	4.3	0.57	1.8	0.9	23	13	42	22	0.073	2,140	85	25	13.3
6 A....	4.2	.53	1.7	1.0	24	13	40	23	.073	2,530	87	26	14.0
7 A....	4.3	.46	1.7	1.2	23	11	38	28	.073	2,800	84	25	13.6
8 A	4.3	.53	1.8	.9	23	12	43	22	.072	3,030	83	25	13.9
9 A....	4.1	.56	1.9	.6	24	14	46	16	.073	3,100	84	25	14.7
10 A....	4.4	.44	1.8	1.2	23	10	41	26	.072	3,190	82	24	14.1
1 B....	4.2	.56	1.9	.8	24	13	44	19	.073	3,080	83	25	14.1
2 B....	4.3	.55	1.9	.8	23	13	45	19	.073	3,170	81	25	14.1
3 B....	4.8	.64	2.0	1.2	21	13	41	25	.073	3,220	78	22	13.0

TABLE III.—*English units.*

Altitude runs. Full power.

Run No.	Approximate altitude (feet).	Revolutions per minute.	Torque (pound feet).	Brake mean effective pressure (pounds per square inch).	Brake horsepower.	Pounds of fuel per hour.	Pounds of fuel per brake horsepower hour.	Temperature (° F.). Oil. Inlet.	Oil. Outlet.	Jacket water. Inlet.	Jacket water. Outlet.	Carburetor air.	Oil pressure (pounds per square inch).	Manifold suction (inches hg.). Cylinder 4-5-6 R.	Cylinder 1-2-3 L.	Barometric pressure (inches hg.).
5 C..	Ground.	1,620	1,273	116.5	393	208	0.53	82	145	145	166	59	43	1.4	1.1	29.3
6 C..	Ground.	1,700	1,279	117.0	414	212	.51	85	164	143	163	59	41	1.2	1.2	29.2
7 C..	5,000	1,690	1,094	100.0	352	187	.53	91	164	135	153	41	39	1.0	.9	25.1
8 C..	5,000	1,610	1,106	101.2	339	198	.58	95	167	135	153	41	40	1.1	.6	25.0
9 C..	10,000	1,600	900	82.3	273	158	.58	95	166	130	146	26	38	.4	.5	20.8
10 C..	10,000	1,690	815	74.6	262	180	.69	85	156	133	149	29	42	.8	.3	20.8
1 D.	10,000	1,710	866	79.2	282	153	.54	74	142	137	149	26	45	1.0	.9	20.8
2 D.	15,000	1,690	653	59.8	209	13	.67	72	162	156	172	14	36	.9	1.0	17.2
3 D.	15,000	1,590	713	65.2	216	123	.57	74	165	138	155	12	34	.2	.4	17.1
4 D.	20,000	1,570	556	50.9	166	112	.67	93	165	144	154	8	30	.8	.8	14.2
5 D.	20,000	1,700	483	44.3	157	128	.81	83	155	156	170	32	34	.6	.6	14.2
6 D.	25,000	1,700	271	24.8	88	124	1.41	88	158	151	158	19	29	.6	.6	11.7
7 D.	25,000	1,590	332	30.4	101	113	1.11	92	164	149	156	11	28	.6	.5	11.9

TABLE IV.—*English units.*

Altitude runs. Full power.

Run No.	Heat distribution based on brake horsepower.				Heat distribution based on heat in fuel.				Air density (pound per cubic foot).	Pounds of air per hour.	Volumetric efficiency (per cent).	Thermal efficiency (per cent).	Pounds of air per pound of fuel, ±0.2.
	Heat in fuel÷(heat in b. h. p.).	Heat in jacket÷(heat in b. h. p.).	Heat in exhaust÷(heat in b. h. p.).	Residual heat ÷(heat in b. h. p.).	Brake horsepower (per cent).	Jacket (per cent).	Exhaust (per cent).	Residual (per cent).					
5 C	4.2	0.62	2.0	0.6	24	15	47	14	0.075	2,980	86	25	14.3
6 C	4.1	.58	1.9	.6	24	14	46	16	.075	3,070	85	26	14.5
7 C	4.3	.61	1.9	.7	23	14	45	18	.066	2,690	83	25	14.3
8 C	4.6	.60	1.8	.7	21	13	39	27	.066	2,560	85	23	13.0
9 C	4.6	.66	1.9	1.0	22	14	42	22	.057	2,130	83	23	13.5
10 C	5.5	.71	1.9	1.9	18	13	34	35	.056	2,180	81	20	12.2
1 D	4.3	.51	1.8	1.0	23	12	42	23	.057	2,220	81	25	14.5
2 D	5.3	.63	1.9	1.5	19	17	35	29	.048	1,740	76	20	12.5
3 D	4.6	.85	2.1	1.1	22	19	45	14	.048	1,750	81	23	14.3
4 D	5.3	.65	1.8	1.9	19	12	33	36	.040	1,360	76	20	12.2
5 D	6.5	.64	1.9	2.5	15	16	29	40	.038	1,420	77	16	11.1
6 D	11.2	.95	2.6	6.7	9	8	23	60	.032	1,130	73	9	9.1
7 D	8.9	.80	2.3	4.8	11	9	25	55	.033	1,120	74	12	10.0

TABLE V.—*English units.*

Propeller load runs.

Run No.	Approximate altitude (feet).	Revolutions per minute.	Torque (pound feet).	Brake mean effective pressure.	Brake horsepower.	Pounds of fuel per hour.	Pound of fuel per brake horsepower hour.	Barometric pressure (inches hg.).
8 D	15,000	1,710	538	59.3	212	142	0.67	17.2
1 E	15,000	1,640	599	54.7	186	107	.57	17.4
2 E	15,000	1,530	514	47.1	150	77	.51	17.2
3 E	15,000	1,400	447	40.9	119	69	.58	17.3
4 E	15,000	1,300	391	35.7	97	61	.63	16.9
5 E	15,000	1,210	331	30.3	76	54	.71	17.4
6 E	10,000	1,690	870	79.6	280	164	.58	20.7
7 E	10,000	1,610	769	70.3	236	121	.51	20.8
8 E	10,000	1,460	665	60.8	184	92	.50	20.5
9 E	10,000	1,390	580	53.1	153	71	.46	20.6
10 E	10,000	1,300	506	46.3	126	62	.49	20.7
11 E	10,000	1,190	430	39.4	98	53	.55	20.7
12 E	5,000	1,690	1,051	96.1	338	167	.49	25.0
13 E	5,000	1,620	975	89.2	300	146	.49	25.1
14 E	5,000	1,510	865	79.1	249	121	.48	24.9
15 E	5,000	1,410	749	67.7	199	97	.49	25.0
16 E	5,000	1,320	656	60.0	165	82	.49	24.9
17 E	5,000	1,200	544	49.8	124	63	.51	24.9

TABLE VI.—*English units.*

Propeller load runs.

Run No.	Temperature (°F.).					Manifold suction (inches hg.).		Air density (pound per cubic foot).	Pounds of air per hour.	Pounds of air per pound of fuel, ±.02.
	Oil.		Jacket water.		Carburetor air.	Cylinder 4-5-6 R.	Cylinder 1-2-3 L.			
	Inlet.	Outlet.	Inlet.	Outlet.						
8 D	88	158	144	158	30	0.7	0.7	0.047	1,750	12.4
1 E	75	139	144	158	25	1.1	2.1	.047	1,610	15.0
2 E	87	155	142	155	14	2.0	3.3	.048	1,430	18.6
3 E	88	156	147	159	14	2.2	6.5	.048	1,110	16.1
4 E	80	145	150	163	25	5.3	4.9	.046	890	14.5
5 E	82	147	140	151	16	6.0	7.1	.048	776	14.1
6 E	88	162	147	163	26	.8	.8	.056	2,170	13.3
7 E	93	165	137	152	28	1.5	2.1	.056	1,930	16.0
8 E	95	165	143	158	29	3.3	4.8	.056	1,520	16.5
9 E	91	166	153	168	26	4.9	4.5	.056	1,300	18.3
10 E	76	126	129	143	27	6.2	5.3	.057	1,090	17.6
11 E	77	139	156	166	25	6.4	8.1	.057	940	17.7
12 E	85	160	145	162	42	1.0	1.0	.066	2,610	15.7
13 E	89	164	139	156	41	2.1	1.8	.066	2,330	16.0
14 E	91	162	136	152	42	3.0	2.6	.067	1,970	16.3
15 E	92	160	138	154	40	4.4	3.9	.066	1,720	17.7
16 E	90	157	137	152	41	5.9	5.3	.066	1,490	18.2
17 E	88	153	144	158	41	8.4	7.1	.066	1,200	19.0

TABLE VII.—*English units.*

Friction horsepower.

Run No.	Approximate altitude (feet).	Revolutions per minute.	Friction horsepower.	Barometric pressure inches Hg.	Air density (pound per cubic foot).	Temperature, (° F.). Oil. Inlet.	Oil. Outlet.	Jacket water. Inlet.	Jacket water. Outlet.	Carburetor air.
26 E.....	15,000	1,210	30	17.2	0.046	91	170	154	155	32
27 E.....	15,000	1,410	41	17.5	.047	92	156	157	158	29
28 E.....	15,000	1,610	52	17.1	.047	94	156	162	163	28
29 E.....	15,000	1,790	61	17.1	.047	100	162	165	166	27
30 E.....	15,000	1,980	70	17.1	.047	107	173	151	168	28
31 E.....	Ground.	1,200	33	....	.071	99	151	166	167	90
32 E.....	Ground.	1,400	43	....	.071	96	155	167	168	93
33 E.....	Ground.	1,600	55	....	.070	95	157	167	168	97
34 E.....	Ground.	1,800	69	....	.069	97	159	171	172	100

TABLE VIII.—*English units.*

Ground and altitude runs.

Revolutions per minute.	Brake horsepower.	Friction horsepower.	Indicated horsepower.	Lb. air per hour÷(lb. air per hour at 2,000 r.p.m.).	I. h. p. ÷(i.h.p. at 2,000 r.p.m.).	Mechanical efficiency (per cent).	Approximate air density (pound per cubic foot).	Air density (pound per cubic foot).	Brake horsepower.	Friction horsepower.	Indicated horsepower.	Lb. air per hour ÷ (lb. air per hour at 0.076 density).	I.h.p.÷ (i. h. p. at 0.076 density).	Mechanical efficiency (per cent).	Revolutions per minute.	B.h.p. ÷(b.h.p. at 0.076 density).
1,200	295	33	328	0.64	0.66	90	0.073	0.076	403	55	458	1.00	1.00	88	1,600	1.00
1,400	344	43	387	.78	.77	89	.073	.066	336	54	390	.85	.86	86	1,600	.83
1,600	385	55	440	.87	.88	87	.073	.057	276	52	328	.72	.72	84	1,600	.68
1,800	415	69	484	.95	.97	86	.073	.048	216	51	267	.58	.58	81	1,600	.53
1,900	419	77	496	.98	.99	84	.073	.040	163	50	213	.46	.46	77	1,600	.40
2,000	410	90	500	1.00	1.00	82	.073	.033	96	49	145	.36	.32	66	1,600	.24

TABLE I.—*Metric units.*

Ground runs. Full power.

Run No.	Approximate altitude (meters).	Revolutions per minute.	Torque (kilogram meter).	Brake mean effective pressure kilograms per square centimeter.	Brake horsepower.	Kilograms of fuel per hour.	Kilogram of fuel per brake horsepower hours.	Temperature (°C.). Oil. Inlet.	Oil. Outlet.	Jacket water. Inlet.	Jacket water. Outlet.	Carburetor air.	Oil pressure (kilograms per square centimeter).	Manifold suction centimeter Hg. cyl. 4–5–6 R.	Barometric pressure centimeter Hg.
5 A....	305	1,220	179	8.3	306	73	0.24	27	61	64	75	15	2.5	1.4	72.6
6 A....	305	1,410	178	8.3	350	82	.24	29	69	58	68	15	2.5	2.0	72.7
7 A....	305	1,600	173	8.0	385	93	.24	31	73	65	74	12	2.8	2.4	71.9
8 A....	305	1,800	163	7.6	410	99	.24	31	73	57	66	15	2.9	3.2	71.3
9 A....	305	1,790	165	7.7	414	95	.23	31	72	56	66	15	2.9	2.7	72.1
10 A....	305	1,890	157	7.3	415	102	.25	38	90	59	67	16	2.9	3.4	72.0
1 B....	305	1,800	166	7.8	420	99	.23	27	71	62	72	15	3.0	2.7	72.4
2 B....	305	1,900	161	7.5	425	102	.24	33	80	63	73	15	2.8	3.3	72.3
3 B....	305	2,000	148	6.9	415	113	.27	37	90	64	74	15	2.9	3.5	72.0

TABLE II.—*Metric units.*

Ground runs. Full power.

Run No.	Heat distribution based on brake horsepower. Heat in fuel ÷ (heat in b.h.p.).	Heat in jacket ÷ (heat in b.h.p.).	Heat in exhaust ÷ (heat in b.h.p.).	Residual heat ÷ (heat in b.h.p.).	Heat distribution based on heat in fuel. Brake horsepower (per cent).	Jacket (per cent).	Exhaust (per cent).	Residual (per cent).	Air density (kilograms per cubic meter).	Kilogram of air per hour.	Volumetric efficiency (per cent).	Thermal efficiency (per cent).	Kilograms of air per kilogram of fuel ±0.2.
5 A..	4.3	0.57	1.8	0.9	23	13	42	22	1.17	970	85	25	13.3
6 A..	4.2	.53	1.7	1.0	24	13	40	23	1.17	1,150	87	26	14.0
7 A..	4.3	.46	1.7	1.2	23	11	38	28	1.17	1,270	84	25	13.6
8 A..	4.3	.53	1.8	.9	23	12	43	22	1.15	1,380	83	25	13.9
9 A..	4.1	.56	1.9	.6	24	14	46	16	1.16	1,400	84	25	14.7
10 A..	4.4	.44	1.8	1.2	23	10	41	26	1.16	1,450	82	24	14.1
1 B..	4.2	.56	1.9	.8	24	13	44	19	1.17	1,400	83	25	14.1
2 B..	4.3	.55	1.9	.8	23	13	45	19	1.17	1,440	81	25	14.1
3 B..	4.8	.64	2.0	1.2	21	13	41	25	1.16	1,460	78	22	13.0

TABLE III.—*Metric units.*

Altitude runs. Full power.

Run No.	Approximate altitude (meters).	Revolutions per minute.	Torque (kilogram meters).	Brake mean effective pressure (kilograms per square centimeter).	Brake horsepower.	Kilograms of fuel per hour.	Kilogram of fuel per brake horsepower. hour.	Temperature (°C.) Oil. Inlet.	Oil. Outlet.	Jacket water. Inlet.	Jacket water. Outlet.	Carburetor air.	Oil pressure (kilograms per square centimeter).	Manifold suction centimeter Hg. Cylinder 4–5–6 R.	Cylinder 1–2–3 L.	Barometric pressure (centimeter Hg).
5 C..	Ground.	1,620	176	8.2	399	94	0.24	28	63	63	74	15	3.0	3.5	2.9	74.5
6 C..	Ground.	1,700	177	8.2	420	96	.23	29	73	61	73	15	2.9	3.1	3.1	74.3
7 C..	1,520	1,690	151	7.0	357	85	.24	33	74	57	67	4	2.7	2.6	2.5	63.8
8 C..	1,520	1,610	153	7.1	344	90	.26	35	75	57	67	5	2.8	2.8	1.5	63.6
9 C..	3,050	1,600	124	5.8	277	72	.26	35	74	54	63	− 3	2.7	1.1	1.2	52.9
10 C..	3,050	1,690	113	5.2	266	82	.31	29	69	56	65	− 2	2.9	1.9	0.8	52.8
1 D..	3,050	1,710	120	5.6	286	69	.24	23	61	58	65	− 3	3.2	2.7	2.3	52.8
2 D..	4,570	1,690	90	4.2	212	63	.30	31	88	69	78	−10	2.5	2.2	2.6	43.6
3 D..	4,570	1,590	99	4.6	219	56	.26	34	92	59	68	−11	2.4	0.5	1.0	43.5
4 D..	6,040	1,570	77	3.6	168	51	.30	34	74	62	67	13	2.1	1.9	2.1	36.1
5 D..	6,040	1,700	67	3.1	159	58	.36	28	68	69	76	0	2.4	1.5	1.4	36.0
6 D..	7,620	1,700	37	1.7	89	56	.63	31	70	66	70	− 7	2.0	1.5	1.6	29.7
7 D..	7,620	1,590	46	2.1	102	51	.50	33	73	65	69	−12	2.0	1.5	1.4	30.1

TABLE IV.—*Metric units.*

Altitude runs. Full power.

Run No.	Heat distribution based on brake horsepower. Heat in fuel ÷ (heat in b.h.p.).	Heat in jacket ÷ (heat in b.h.p.).	Heat in exhaust ÷ (heat in b.h.p.).	Residual heat ÷ (heat in b.h.p.).	Heat distribution based on heat in fuel. Brake horsepower (per cent).	Jacket (per cent).	Exhaust (per cent).	Residual (per cent).	Air density (kilograms per square centimeter).	Kilograms of air per hour.	Volumetric efficiency (per cent).	Thermal efficiency (per cent).	Kilograms of air per kilogram of fuel ±0.2.
5 C....	4.2	0.62	2.0	0.6	24	15	47	14	1.20	1,350	86	25	14.3
6 C....	4.1	.58	1.9	.6	24	14	46	16	1.20	1,390	85	26	14.5
7 C....	4.3	.61	1.9	.7	23	14	45	18	1.06	1,220	83	25	14.3
8 C....	4.6	.60	1.8	.7	21	13	39	27	1.06	1,160	85	23	13.0
9 C....	4.6	.66	1.9	1.0	22	14	42	22	.91	970	83	23	13.5
10 C....	5.5	.71	1.9	1.9	18	13	34	35	.90	990	81	20	12.2
1 D....	4.3	.51	1.8	1.0	23	12	42	23	.91	1,010	81	25	14.5
2 D....	5.3	.63	1.9	1.5	19	17	35	29	.77	790	76	20	12.5
3 D....	4.6	.85	2.1	1.1	22	19	45	14	.77	790	81	23	14.3
4 D....	5.3	.65	1.8	1.9	19	12	33	36	.64	620	76	20	12 2
5 D....	6.5	.64	1.9	2.5	15	16	29	40	.61	640	77	16	11.1
6 D....	11.2	.95	2.6	6.7	9	8	23	60	.52	510	73	9	9.1
7 D....	8.9	.80	2.3	4.8	11	9	25	55	.53	510	74	12	10.0

TABLE V.—*Metric units.*

Propeller load runs.

Run No.	Approximate altitude (meters).	Revolutions per minute.	Torque (kilogram meter).	Brake mean effective pressure (kilograms per square centimeter.	Brake horsepower.	Kilograms of fuel per hour.	Kilograms of fuel per brake horsepower hour.	Barometric pressure (centimeters Hg.).
8 D	4,570	1,710	74	4.2	215	64	0.30	43.7
1 E	4,570	1,640	83	3.8	189	48	.26	44.1
2 E	4,570	1,530	71	3.3	152	35	.23	43.7
3 E	4,570	1,400	62	2.9	121	31	.26	43.8
4 E	4,570	1,300	54	2.5	98	28	.28	43.0
5 E	4,570	1,210	46	2.1	78	25	.32	44.1
6 E	3,050	1,690	120	5.6	284	74	.26	52.5
7 E	3,050	1,610	106	4.9	239	55	.23	52.7
8 E	3,050	1,460	92	4.3	187	42	.22	52.1
9 E	3,050	1,390	80	3.7	155	32	.21	52.2
10 E	3,050	1,300	70	3.2	128	28	.22	52.7
11 E	3,050	1,190	59	2.8	99	24	.24	52.7
12 E	1,520	1,690	145	6.7	343	76	.22	63.5
13 E	1,520	1,620	135	6.3	304	66	.22	63.7
14 E	1,520	1,510	120	5.6	253	55	.22	63.2
15 E	1,520	1,410	102	4.7	202	44	.22	63.6
16 E	1,520	1,320	91	4.2	167	37	.22	63.3
17 E	1,520	1,200	75	3.5	126	29	.23	63.4

TABLE VI.—*Metric units.*

Propeller load runs.

Run No.	Temperature (° C.).					Manifold suction (centimeters Hg.).		Air density (kilograms per cubic meter).	Kilograms of air per hour.	Kilograms of air per kilograms fuel, ±0.2.
	Oil.		Jacket water.		Carburetor air.	Cylinder 4–5–6 R.	Cylinder 1–2–3 L.			
	Inlet.	Outlet.	Inlet.	Outlet.						
8 D	31	70	62	70	−1	1.7	1.9	0.75	790	12.4
1 E	24	60	62	70	−4	2.8	5.2	.76	730	15.0
2 E	31	68	61	68	−10	5.0	8.3	.77	650	18.6
3 E	31	69	64	71	−10	5.5	16.4	.77	510	16.1
4 E	27	63	65	73	−4	13.5	12.4	.74	400	14.5
5 E	28	64	60	66	−9	15.3	17.9	.78	350	14.1
6 E	31	72	64	73	−3	2.1	2.1	.90	990	13.3
7 E	34	74	58	67	−2	3.7	5.3	.90	870	16.0
8 E	34	74	62	70	−2	8.3	12.2	.89	690	16.5
9 E	33	74	67	76	−3	12.5	11.5	.90	590	18.3
10 E	24	52	54	62	−3	15.7	13.4	.91	490	17.6
11 E	25	59	69	74	−4	16.3	20.5	.91	430	17.7
12 E	30	71	63	72	5	2.4	2.5	1.06	1,190	15.7
13 E	32	73	60	69	5	5.2	4.7	1.06	1,060	16.0
14 E	33	72	58	67	6	7.6	6.7	1.07	890	16.3
15 E	33	71	59	67	5	11.1	10.0	1.06	780	17.7
16 E	32	69	59	67	5	15.1	13.5	1.06	670	18.2
17 E	31	67	62	70	5	21.4	18.0	1.06	540	19.0

TABLE VII.—*Metric units.*

Friction horsepower.

Run No.	Approximate altitude (meters).	Revolutions per minute.	Friction horsepower.	Barometric pressure (centimeters Hg.).	Air density (kilograms per cubic meter).	Temperature (° C.).				
						Oil.		Jacket water.		Carburetor air.
						Inlet.	Outlet.	Inlet.	Outlet.	
26 E	4,570	1,210	30	43.7	0.75	33	76	68	68	0
27 E	4,570	1,410	41	43.9	.75	33	69	70	70	−2
28 E	4,570	1,610	52	43.5	.75	34	69	72	73	−2
29 E	4,570	1,790	61	43.4	.75	38	72	74	74	−3
30 E	4,570	1,980	70	43.5	.75	42	78	66	76	−2
31 E	[1]	1,200	33	....	1.14	37	66	74	75	32
32 E	[1]	1,400	43	....	1.13	35	68	75	76	34
33 E	[1]	1,600	55	....	1.12	35	69	75	76	36
34 E	[1]	1,800	69	....	1.11	36	71	77	78	38

[1] Ground.

TABLE VIII.—*Metric units.*

Ground and altitude runs.

Revolutions per minute.	Brake horsepower.	Friction horsepower.	Indicated horsepower.	Kg. air per hour ÷ (kg. air per hour at 2,000 r.p.m.).	I.h.p. ÷ (i.h.p. at 2,000 r.p.m.).	Mechanical efficiency (per cent).	Approximate air density (kilograms per cubic meter).
1,200	299	33	333	0.64	0.66	90	1.17
1,400	349	44	393	.78	.77	89	1.17
1,600	391	56	447	.87	.88	87	1.17
1,800	421	70	491	.95	.97	86	1.17
1,900	425	78	503	.98	.99	84	1.17
2,000	416	91	507	1.00	1.00	82	1.17

Air density (kilograms per cubic meter).	Brake horsepower.	Friction horsepower.	Indicated horsepower	Kg. air per hour ÷ (kg. air per hour at 1.22 density).	I.h.p. ÷ (i.h.p. at 1.22 density).	Mechanical efficiency (per cent).	Revolutions per minute.	B.h.p. ÷ (b.h.p. at 1.22 density).
1.22	409	56	465	1.00	1.00	88	1,600	1.00
1.06	341	54	395	.85	.86	86	1,600	.83
.91	280	53	333	.72	.72	84	1,600	.68
.77	219	52	271	.58	.58	81	1,600	.53
.64	165	51	216	.46	.46	77	1,600	.40
.53	97	50	147	.36	.32	66	1,600	.24

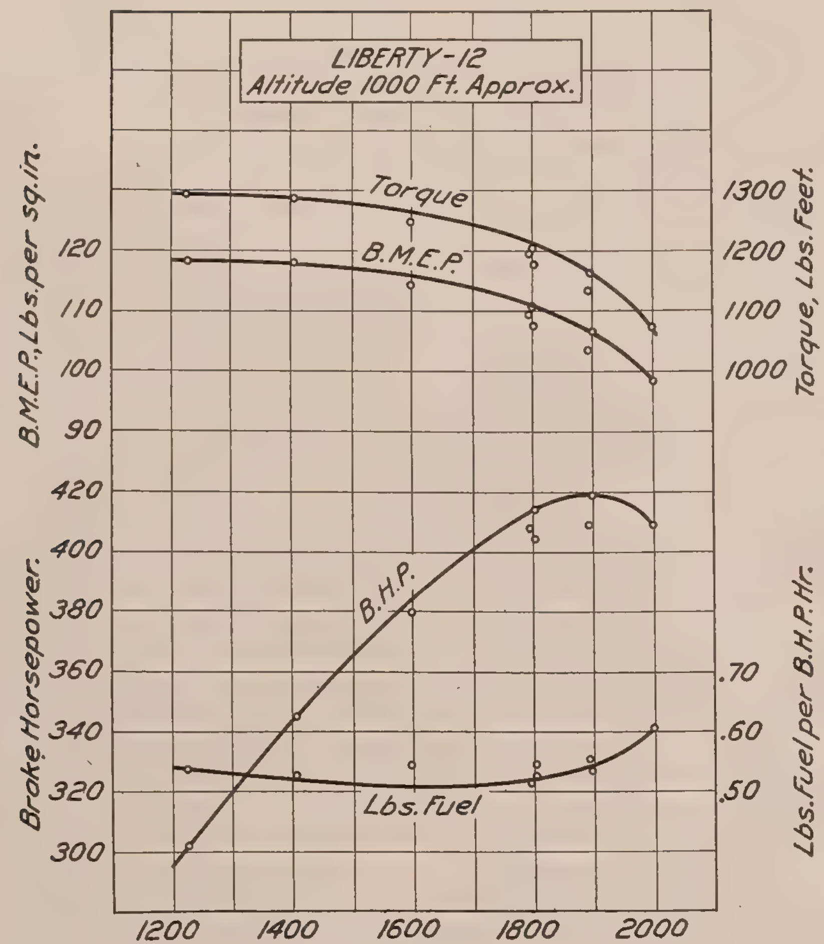

Fig. 1.

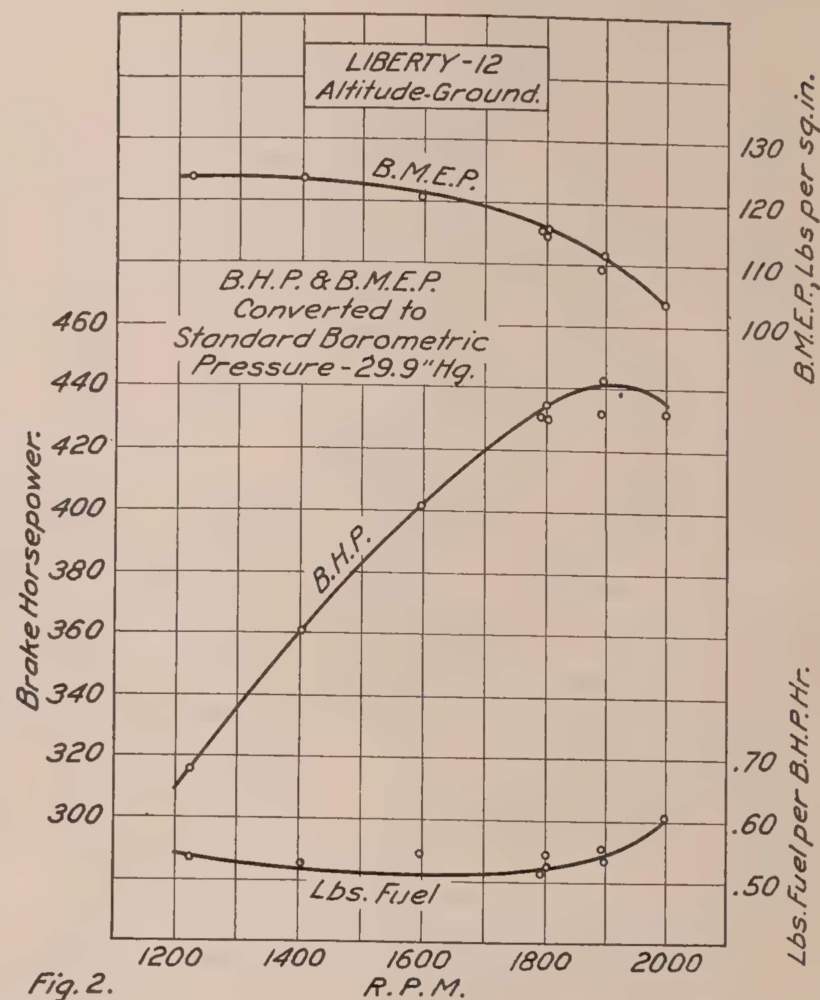

Fig. 2.

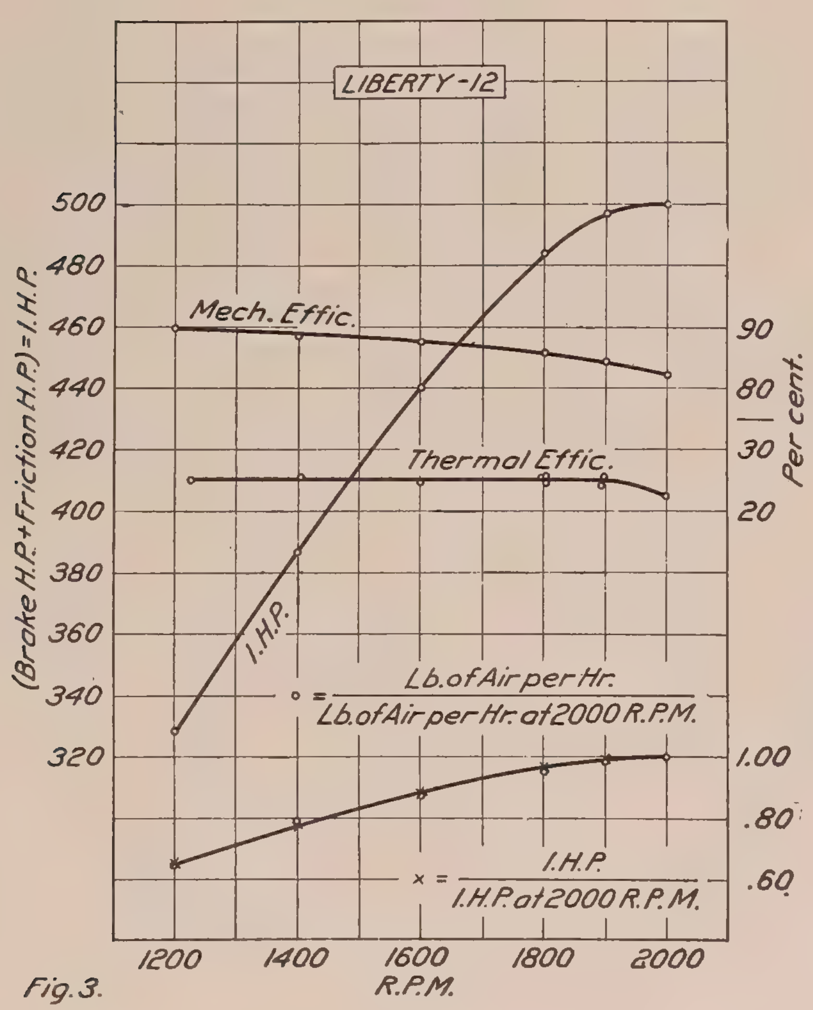

Fig. 3.

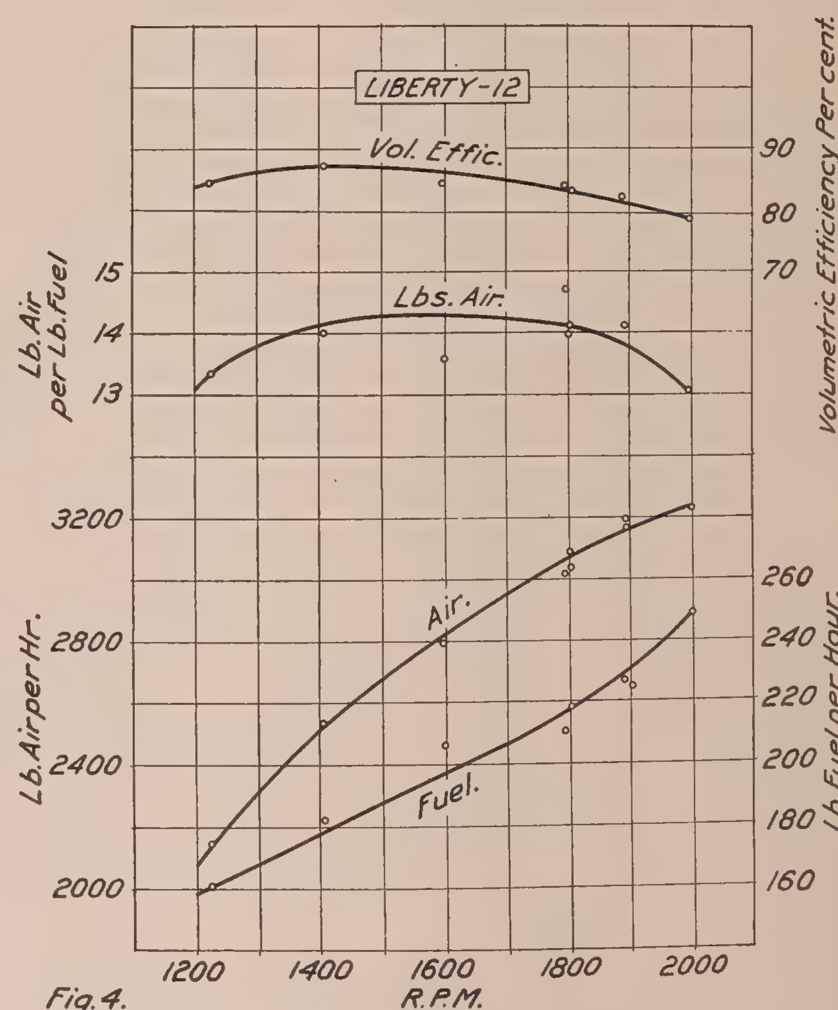

Fig. 4.

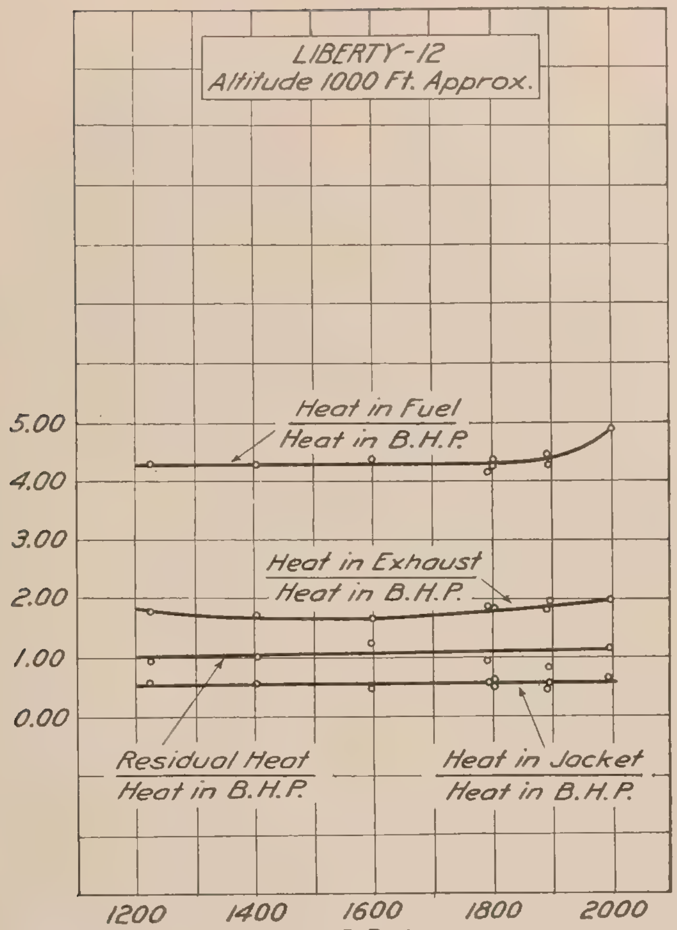

Fig. 5.

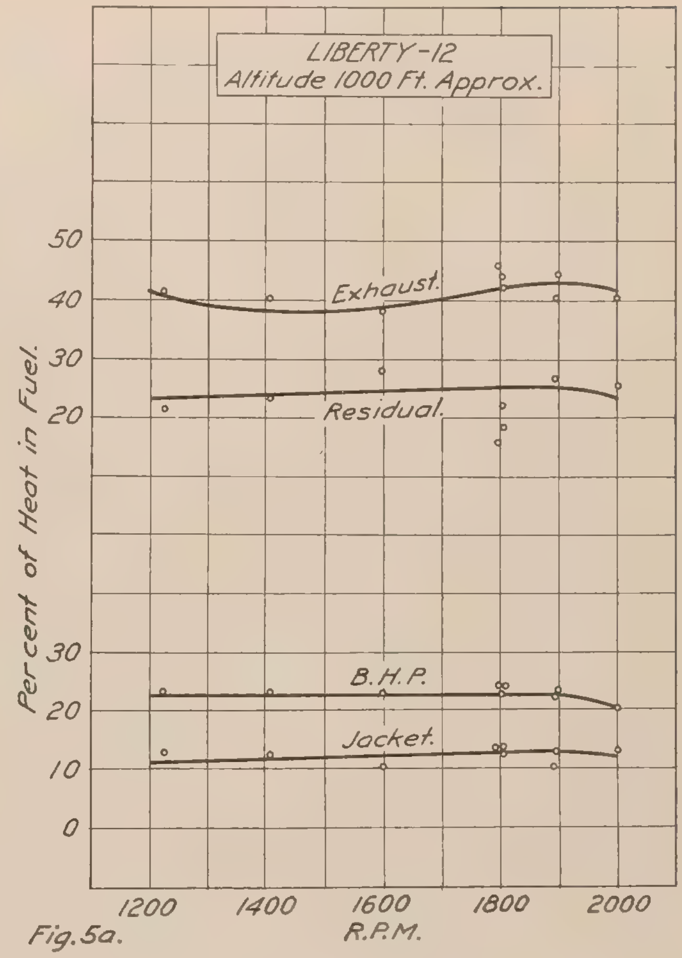

Fig. 5a.

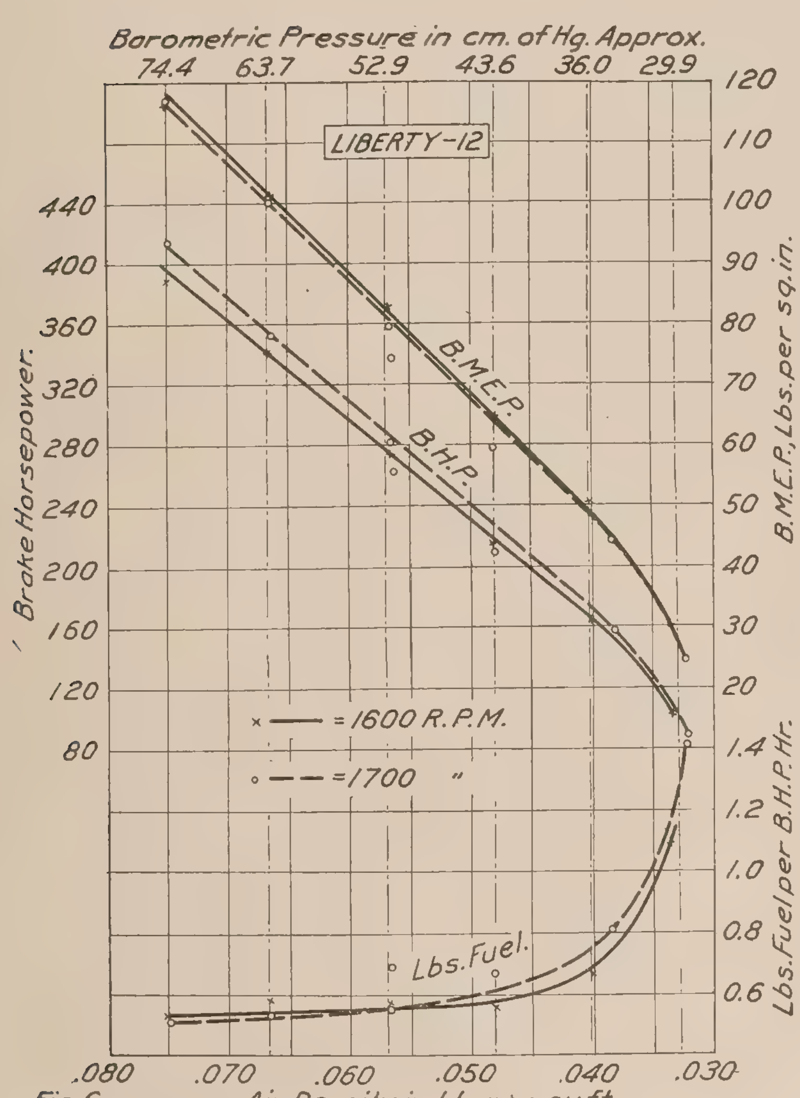

Fig. 6.

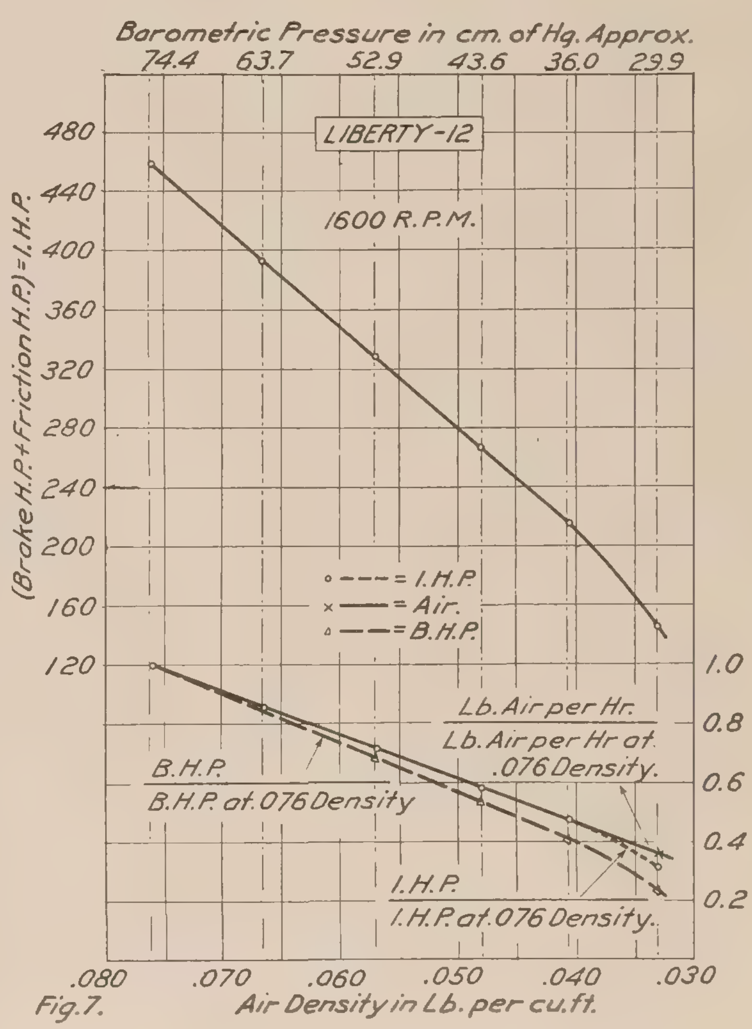

Fig. 7.

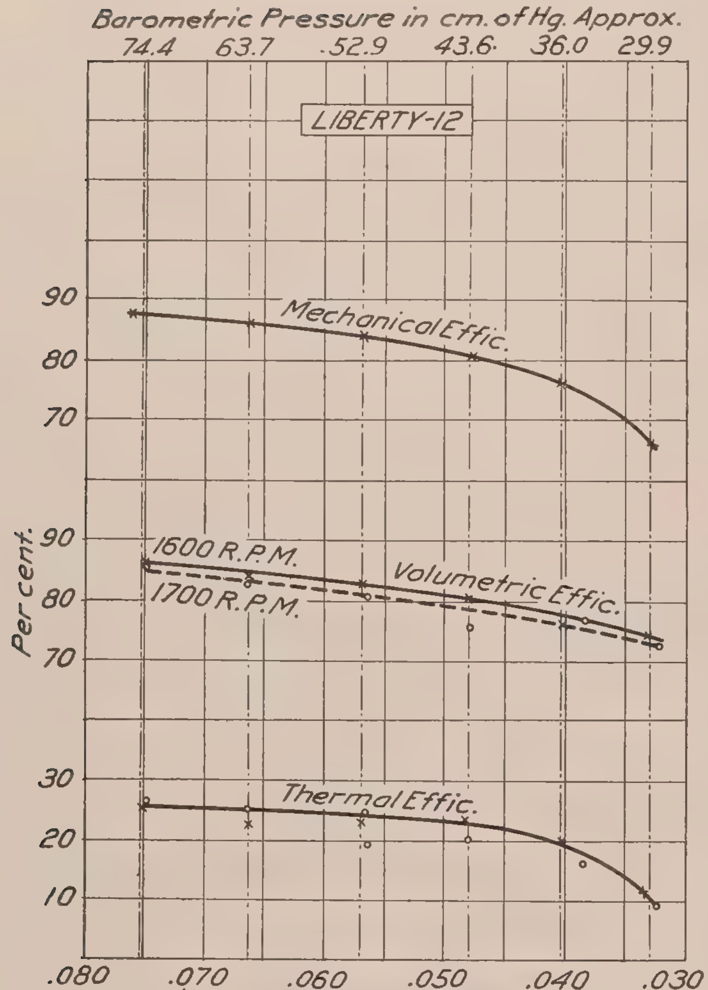

Fig. 8.

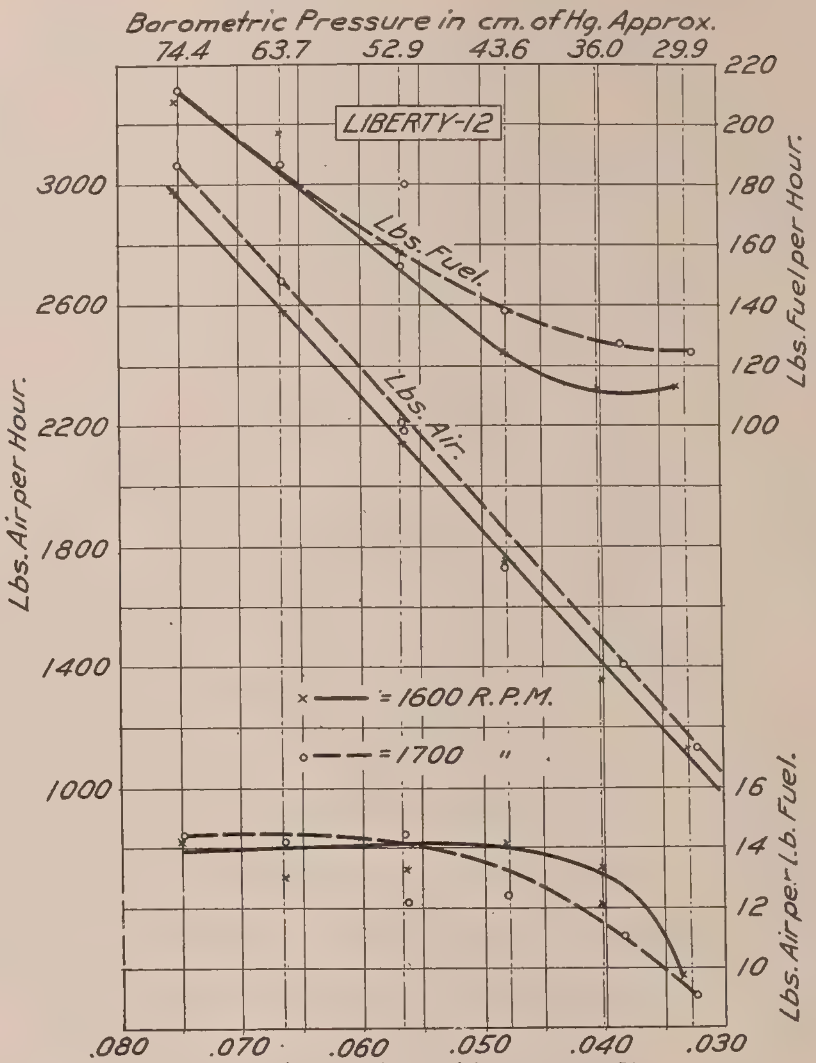

Fig. 9.

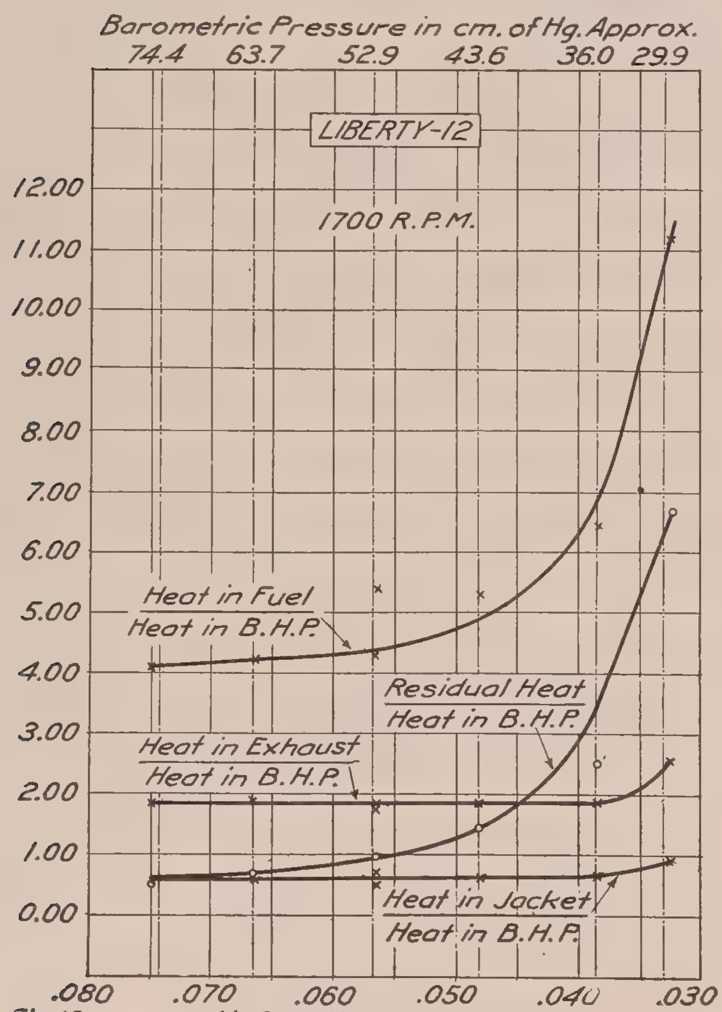

Fig. 10.

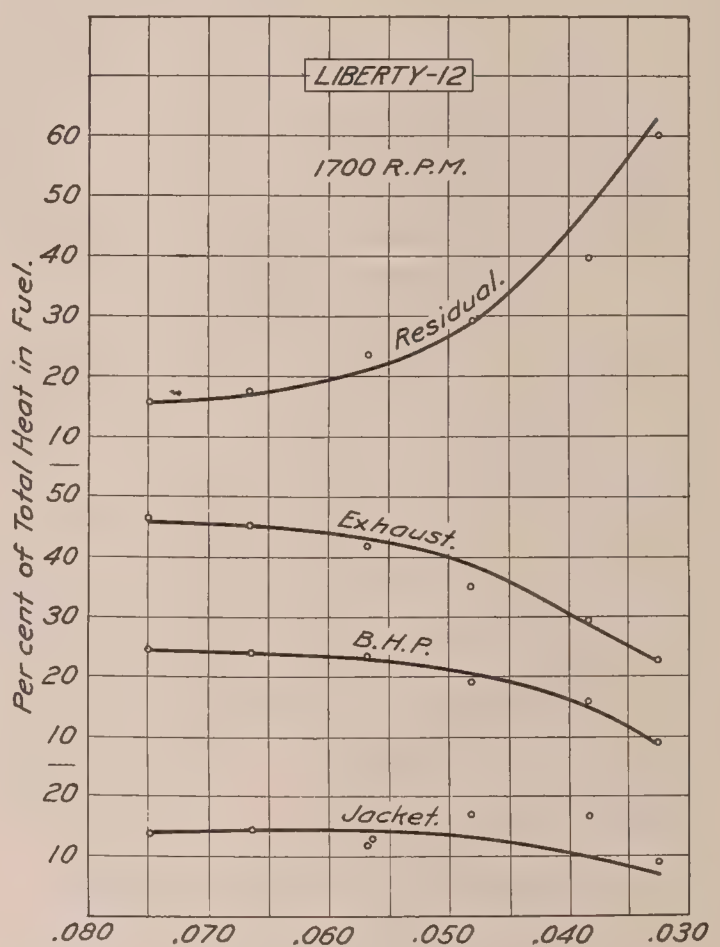

Fig. 10a.

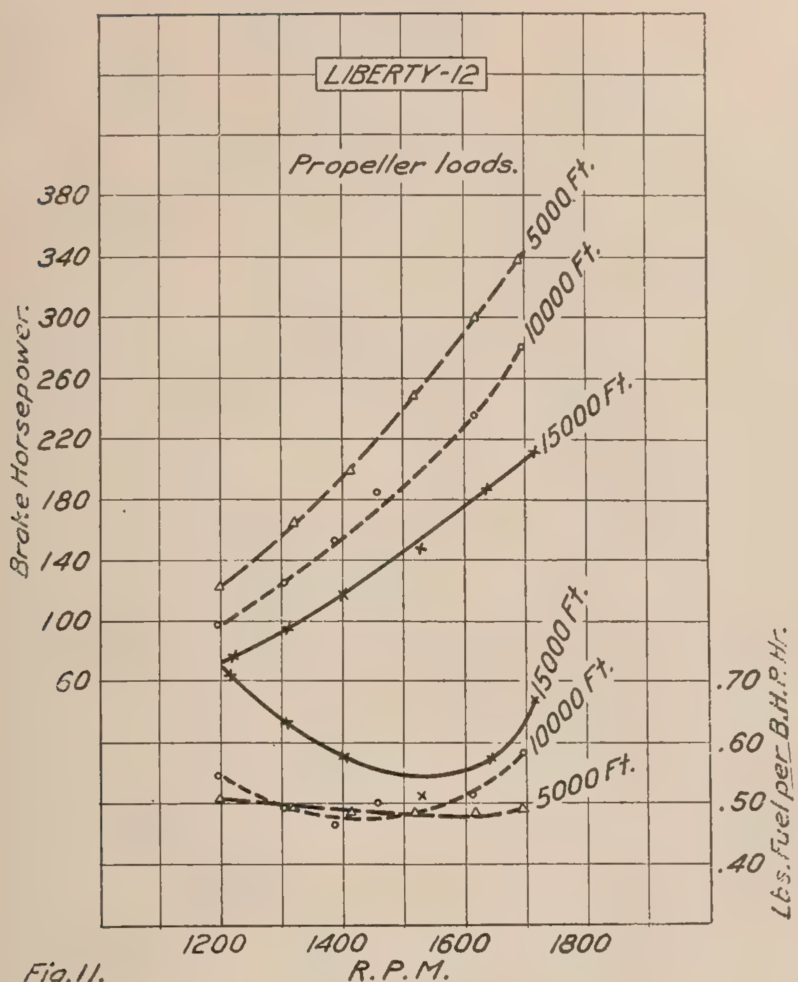

Fig. 11.

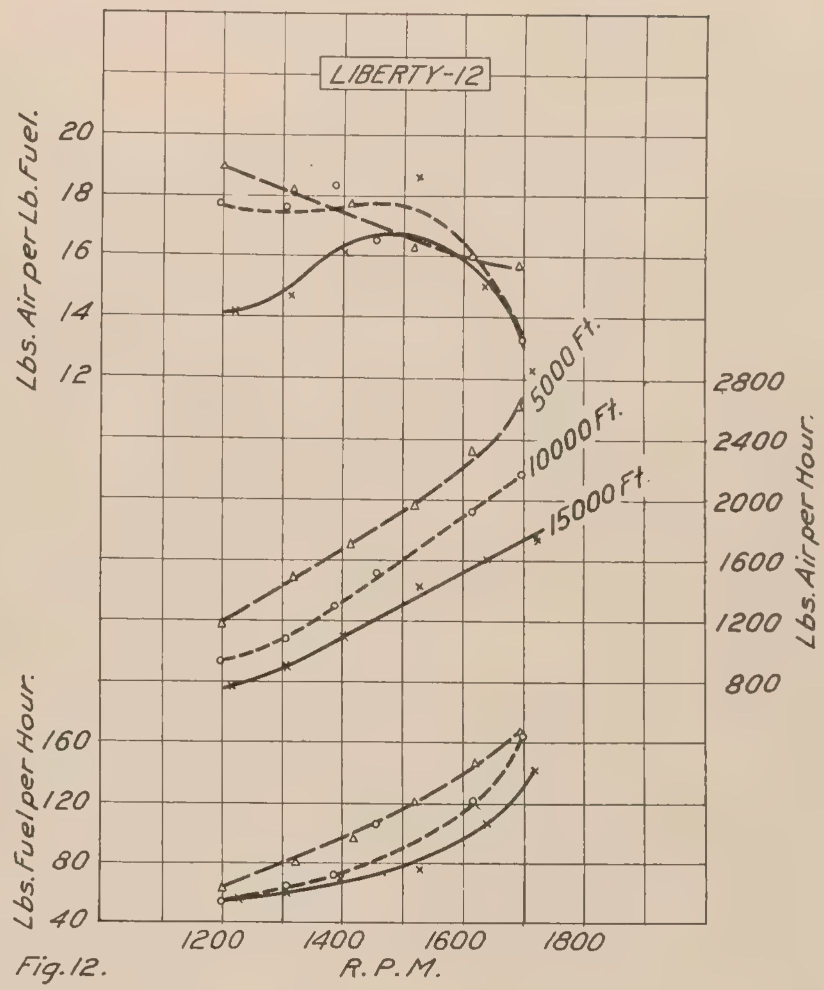

Fig. 12.

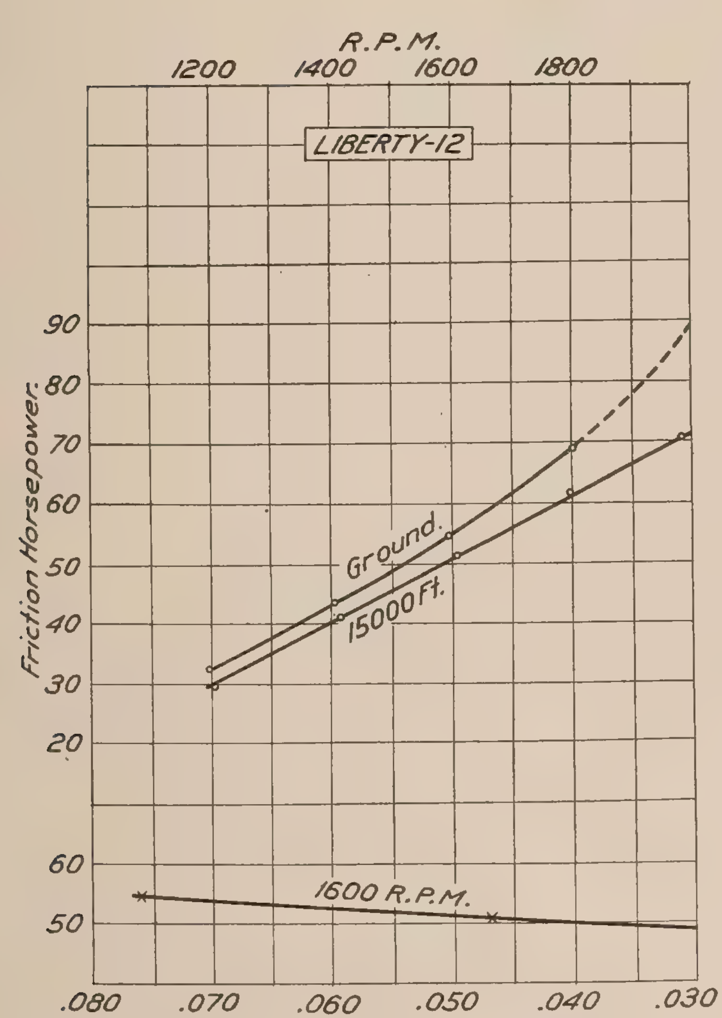

Fig. 13.

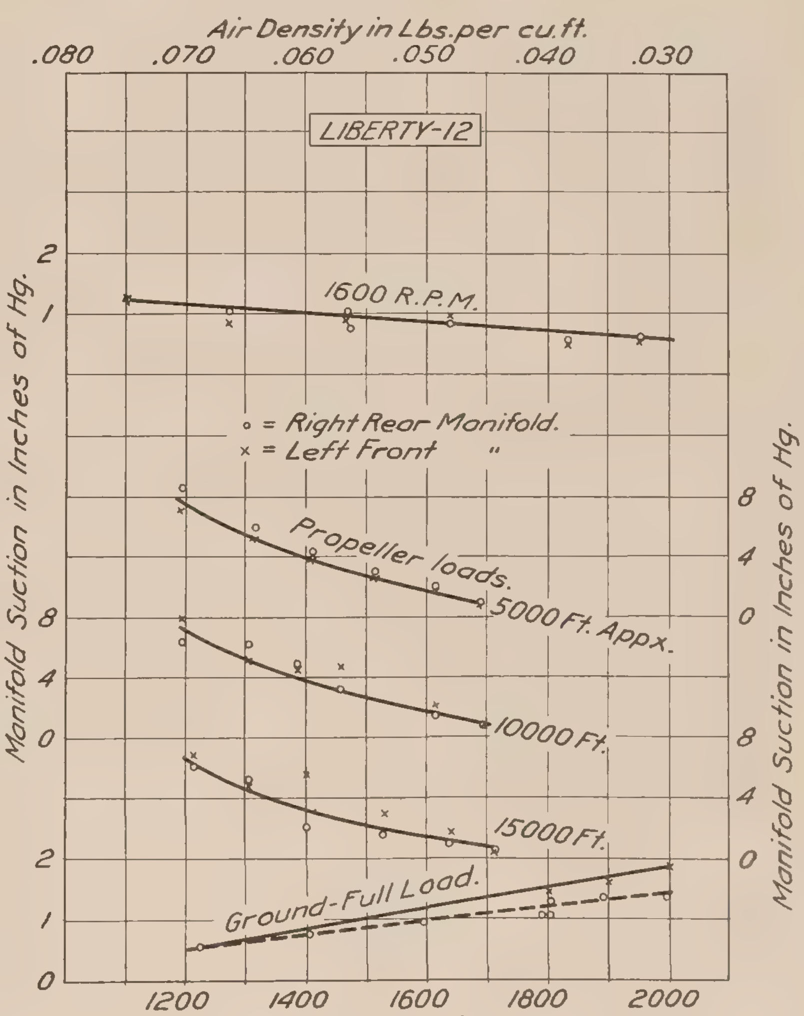

Fig. 14.

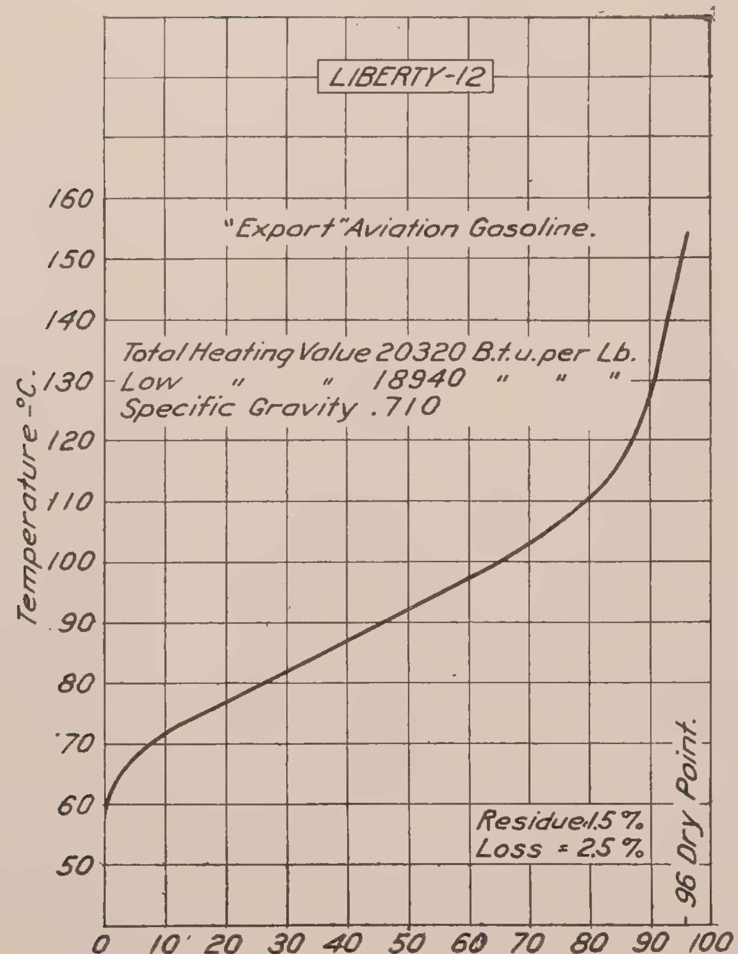
LIBERTY-12
"Export" Aviation Gasoline.
Total Heating Value 20320 B.t.u. per Lb.
Low " " 18940 " " "
Specific Gravity .710
Residue 1.5 %
Loss = 2.5 %
96 Dry Point.
Temperature °C.
160
150
140
130
120
110
100
90
80
70
60
50
0 10 20 30 40 50 60 70 80 90 100
Per cent Distilled.

Fig. 15.

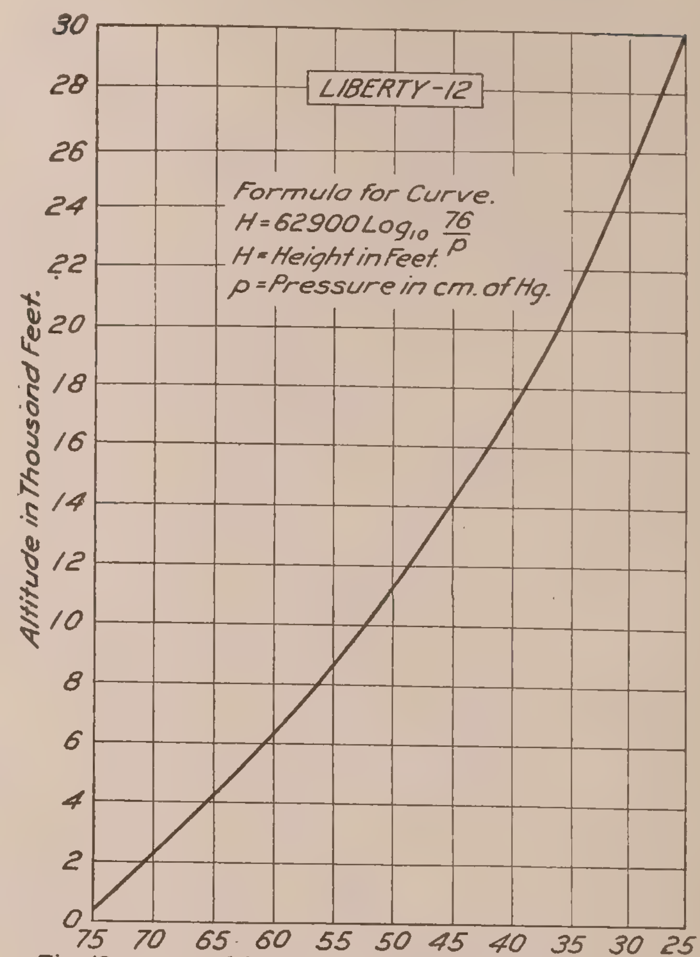
LIBERTY-12
Formula for Curve.
$H = 62900 \log_{10} \frac{76}{p}$
H = Height in Feet.
p = Pressure in cm. of Hg.
Altitude in Thousand Feet.
30
28
26
24
22
20
18
16
14
12
10
8
6
4
2
0
75 70 65 60 55 50 45 40 35 30 25
Pressure in cm. of Hg.

Fig. 16.

REPORT No. 103

PERFORMANCE OF A 300-HORSEPOWER HISPANO-SUIZA AIRPLANE ENGINE

By S. W. SPARROW and H. S. WHITE
Bureau of Standards

REPORT No. 103.

PERFORMANCE OF A 300-HORSEPOWER HISPANO-SUIZA AIRPLANE ENGINE.

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Bureau of Standards.

RÉSUMÉ.

The following report of a complete performance test of a 300-horsepower Hispano-Suiza engine was submitted for publication to the National Advisory Committee for Aeronautics by the Bureau of Standards. The test described in the report was conducted in the altitude chamber of the Bureau of Standards under the joint supervision of the technical staff of the Bureau of Standards and the Engineering Division of the Air Service. The program of tests was planned in cooperation with the Engineering Division of the Air Service of the United States Army so as to yield enough data to determine adequately the characteristics of the engine for aviation purposes without operating it for so long a time as to prevent extensive flying tests from being carried out with the same engine later. The particular engine used in these tests was assembled by the Engineering Division at McCook Field and subjected to the standard dynamometer test for operation at ground level, then shipped to the Bureau of Standards and mounted in the altitude chamber without overhaul. After the altitude test it was returned to McCook Field for such flight tests as might be desired.

A prime requisite of the aviation engine is durability, but it is evident that the long runs necessary to determine this are more properly made with less costly and elaborate equipment than that of the altitude chamber.

The following tests were made:

1. A full power run at ground altitude at speeds from 1,400 to 2,200 r. p. m.
2. An altitude-power run at full throttle and at speeds of 1,600 and 1,800 r. p. m. from the ground to 25,000 feet (7,620 meters) in steps of 5,000 feet (1,520 meters).
3. Propeller load runs, in which the dynamometer load was so adjusted as to produce approximately the same engine load as would be imposed by the propeller at speeds from 1,400 r. p. m. to the normal propeller speed of 1,800 r. p. m. These were taken at altitudes of 5,000, 10,000, and 15,000 feet. (1,520, 3,050, 4,570 meters.)
4. Friction horsepower runs at the ground and at 15,000 feet. (4,570 meters.)

RESULTS.

Some of the outstanding results are given in the tables accompanying this résumé. Correcting the results to a standard barometric pressure of 29.9 inches (76.0 cm.) of mercury gives a brake horsepower at 2,200 r. p. m. of 352 (357 metric horsepower), and a maximum brake mean effective pressure of 128 pounds per square inch (9 kg. per sq. cm.) at about 1,600 r. p. m. The mechanical efficiency varies from 88 per cent to 83 per cent from speeds of 1,400 r. p. m. to 2,200 r. p. m., while the brake thermal efficiency, based on the lower calorific value of the fuel maintains a constant value of 26 per cent over the same range.

Due to lack of an adequate altitude control on the carburetor, the mixture became extremely rich at altitudes of 20,000 feet (6,040 meters) and higher. Below this altitude, where the air fuel ratio could be adjusted to give minimum fuel consumption consistent with maximum brake

horsepower, the brake horsepower and brake mean effective pressure were found to bear a straight line relation to carburetor air density. At 1,800 r. p. m. and at a density of 0.040 pounds per cubic foot (0.64 kg. per cu. m.), the brake horsepower is about 42 per cent of that at the ground and the indicated horsepower is about 47 per cent of that at the ground.

CONCLUSIONS.

The information in such a report as this will be of most value when compared with results of similar tests on other engines. It then serves as a basis for comparing the relative merits of the two engines and as a means of explaining the superiority of one engine to another in any particular phase of performance.

The test shows the inadequacy of the carburetor altitude control of air-fuel ratio for heights above 20,000 feet (6,040 meters). It also shows how the relative importance of high mechanical efficiency increases with altitude.

TABLE A.—*English units.*

Ground runs. Full power.

Approximate altitude in feet.	R. P. M.	B. m. e. p., lb./sq. in.	B. H. P.	Lb. of fuel per b. h. p. hr.	Carb. air temp. °F.	Air density, lb./cu. ft.	Volumetric efficiency, per cent.	Thermal efficiency, per cent.	Lb. air/lb. fuel ±0.2.
500	1, 420	122. 6	248	0. 52	59	0. 075	90	26	14. 6
500	1, 640	124. 7	292	. 51	59	. 075	89	26	14. 5
500	1, 840	122. 9	317	. 52	60	. 075	90	26	14. 3
500	1, 980	117. 4	330	. 51	59	. 075	89	26	15. 2
500	2, 190	110. 0	342	. 52	59	. 074	87	26	15. 2

TABLE B.—*English units.*

Altitude runs. Full power.

Approximate altitude in feet.	R. P. M.	B. m. e. p., lb. per sq. in.	B. H. P.	Lb. of fuel/b. h. p. hr.	Carb. air, temp. °F.	Air density, lb./cu. ft.	Volumetric efficiency, per cent.	Thermal efficiency, per cent.	Lb. of air per lb. of fuel ±0.2.
Ground.	1, 600	124. 5	283	0. 53	59	0. 072	91	25	14. 3
Ground.	1, 800	123. 6	316	. 54	60	. 075	91	25	13. 9
5,000	1, 610	105. 1	241	. 53	58	. 064	91	25	14. 2
5,000	1, 790	103. 1	264	. 54	41	. 066	90	25	14. 7
10,000	1, 600	84. 7	193	. 60	26	. 056	89	23	13. 6
10,000	1, 810	84. 0	216	. 56	26	. 056	89	24	14. 6
15,000	1, 590	68. 3	155	. 61	22	. 047	91	22	14. 2
15,000.	1, 790	66. 6	170	. 59	19	. 047	87	23	14. 2
20,000	1, 620	46. 1	107	. 86	13	. 039	88	16	12. 0
20,000	1, 820	51. 4	133	. 69	11	. 040	88	19	13. 5
25,000	1, 780	29. 9	76	1. 18	12	. 033	89	11	11. 5
25,000	1, 600	31. 5	72	1. 12	11	. 033	91	12	11. 6

TABLE C.—*English units.*

Ground runs.

R. P. M.	B. H. P.	F. H. P.	I. H. P.	Mechanical efficiency, per cent.	Air density, lb. per cu. ft.
1,400	243	34	277	88	0. 075
1,600	284	43	328	87	. 075
1,800	315	53	368	85	. 075
2,000	334	62	396	84	. 075
2,200	343	72	415	83	. 075

TABLE D.

Altitude runs.

Air density, lb. per cu. ft.	B. H. P.	F. H. P.	I. H. P.	Mechanical efficiency.	R. P. M.	B.h.p.÷(b.h.p. at 0.075 density).
0.075	318	53	371	86	1,800	1.00
.065	263	30	313	84	1,800	.83
.055	210	46	256	82	1,800	.66
.045	158	43	201	78	1,800	.50
.040	131	42	173	76	1,800	.41
.035	96	40	136	71	1,800	.30

TABLE A.—*Metric units.*

Ground runs. Full power.

Approximate altitude in meters.	R. P. M.	B.m.e.p. kg. per sq. cm.	B. H. P.	Kg. of fuel per b. h. p. hr.	Carb. air, temp. °C.	Air density, kg. per cu. m.	Volumetric efficiency, per cent.	Thermal efficiency, per cent.	Kg. air per kg. fuel ±0.2.
152	1,420	8.6	251	0.23	15	1.20	90	26	14.6
152	1,640	8.8	296	.23	15	1.20	89	26	14.5
152	1,840	8.6	322	.23	15	1.20	90	26	14.3
152	1,980	8.2	335	.23	15	1.20	89	26	15.2
152	2,190	7.2	347	.23	15	1.19	87	26	15.2

TABLE B.—*Metric units.*

Altitude runs. Full power.

Approximate altitude in meters.	R. P. M.	B.m.e.p. kg. per sq. cm.	B. H. P.	Kg. of fuel per b. h. p. hr.	Carb. air, temp. °C.	Air density, kg. per cu. m.	Volumetric efficiency, per cent.	Thermal efficiency, per cent.	Kg. air per kg. of fuel ±0.2.
Ground.	1,600	8.8	287	0.24	15	1.16	91	25	14.3
Ground.	1,800	8.7	320	.24	16	1.20	91	25	13.9
1,520	1,610	7.4	244	.24	14	1.02	91	25	14.2
1,520	1,790	7.2	268	.24	5	1.06	90	25	14.7
3,050	1,600	6.0	195	.27	− 3	.90	89	23	13.6
3,050	1,810	5.9	219	.25	− 3	.90	89	24	14.6
4,570	1,590	4.8	157	.27	− 6	.76	91	22	14.2
4,570	1,790	4.7	173	.26	− 7	.76	87	23	14.2
6,040	1,620	3.2	109	.38	−11	.63	88	16	12.0
6,040	1,820	3.6	135	.31	−12	.64	88	19	13.5
7,620	1,780	2.1	77	.53	−11	.53	89	11	11.5
7,620	1,600	2.2	73	.50	−12	.53	91	12	11.6

TABLE C.—*Metric units.*

Ground runs.

R. P. M.	B. H. P.	F. H. P.	I. H. P.	Mechanical efficiency, per cent.	Air density, kg. per cu. m.
1,400	246	34	280	88	1.20
1,600	288	44	332	87	1.20
1,800	319	54	373	85	1.20
2,000	339	63	402	84	1.20
2,200	348	73	421	83	1.20

TABLE D.—*Metric units.*

Altitude runs.

Air density, kg. per cu. m.	B. H. P.	F. H. P.	I. H. P.	Mechanical efficiency, per cent.	R. P. M.	B.h.p.÷(b.h.p. at 1.20 density).
1.20	322	54	376	86	1,800	1.00
1.04	267	50	317	84	1,800	.83
.88	213	47	260	82	1,800	.66
.72	160	44	204	78	1,800	.50
.64	133	42	175	76	1,800	.41
.56	97	41	138	71	1,800	.30

OBJECT OF TEST.

The test was made to determine the performance of a 300 horsepower Hispano-Suiza engine and was typical of the class of tests usually run on a new type engine in that some completeness was sacrificed in order to restrict the actual running time of the engine to an amount which would leave the engine in good condition for actual flight work.

DESCRIPTION OF ENGINE AND APPARATUS.

(*A*). *Engine and supplies.*

The engine used was a 300 horsepower Hispano-Suiza, S. C. No. 13481. This is a Vee type motor with eight water-cooled cylinders. It has a bore of 140 mm. (5.51 inches), stroke of 150 mm. (5.91 inches), and a compression ratio of 5.3. The Stromberg carburetor used is provided with a manually operated valve for controlling the air-fuel ratio at the different altitudes. Mobile B oil was used for lubrication and X gasoline for fuel. The X gasoline conforms to the Aircraft Production Board's Specification 3512 for Export Aviation Gasoline for the A. E. F., 1918. A distillation curve of the fuel is given on curve sheet 15.

(*B*). *Apparatus.*

The engine was tested in the Altitude Chamber of the Bureau of Standards. This chamber and apparatus is described in report No. 44 of the National Advisory Committee for Aeronautics (Bureau of Standards Automotive Power Plants Report No. 52). Provision is made for reducing the pressure of the air in the chamber to that of the altitude desired, while at the same time its temperature may be reduced to correspond with the temperature that prevails at that altitude. Outside the chamber there is ample equipment for measuring power, fuel consumption, and various temperatures and pressures.

PROGRAM OF TESTS.

(1) A run was made with wide-open throttle at ground altitude at speeds from 1,400 r. p. m. to 2,200 r. p. m. The spark advance was adjusted for maximum power at each speed. The carburetor was adjusted at each speed to give the least fuel consumption possible with maximum power. To secure this result the carburetor was first adjusted for maximum power and then the mixture was leaned until the torque dropped appreciably. The mixture was then again enriched until maximum torque was restored.

(2) A run was made with wide-open throttle at speeds of 1,600 r. p. m. and 1,800 r. p. m. at altitudes of ground, 5,000, 10,000, 15,000, 20,000, and 25,000 feet (1,520, 3,050, 4,570, 6,040, and 7,620 meters). At each speed and altitude the spark and carburetor were adjusted as for the ground run.

(3) A series of runs were made at altitudes of 5,000, 10,000, and 15,000 feet (1,520, 3,050, and 4,570 meters) at speeds of 1,400, 1,500, 1,600, 1,700, and 1,800 r. p. m. In these runs the dynamometer and throttle were so adjusted as to put a load on the engine at each speed equal to that which would be imposed by a propeller whose normal speed was 1,800 r. p. m. In runs of this type it is assumed that the horsepower of a propeller varies as the cube of the speed.

Thus, if 1,800 be the normal r. p. m. of the propeller, that is, the r. p. m. obtained with full power of the engine, then the horsepower at 1,400 r. p. m. will be $\frac{1400^3}{1800^3}$ times the horsepower at 1,800 r. p. m. In these runs the spark and carburetor were adjusted at 1,800 r. p. m. as in the above runs, but these adjustments were not altered for the other loads.

(4) A series of friction horsepower runs were made at speeds from 1,400 r. p. m. to 2,200 r. p. m. at altitudes of ground and 15,000 feet (4,570 meters.) In these runs the engine was operated under power until oil and water temperature became normal. It was then driven by the dynamometer and the power input measured.

METHOD OF OBTAINING RESULTS.

The results of the tests are given in Tables 1 to 9. A detailed record of the complete test procedure of the laboratory, both in securing data and computing results, is in preparation, so that a brief explanation here will suffice. The run numbers are those that were used on the original sheets to designate the different runs.

Altitude was determined from the curve sheet number 16, using the barometric pressure measured at the carburetor entrance. The engine torque was measured on a 21-inch arm on the dynamometer, and from this value the torque in pound-feet, brake mean effective pressure, and brake horsepower were calculated. The brake horsepower calculation, of course, required the speed which was obtained with a revolution counter. Temperatures were all measured with thermocouples and pressures with U type manometers.

The volume of air used per unit time was measured with a Venturi meter calibrated in place against a carefully tested Thomas meter. From measurements of temperature and pressure air density was figured, and then the weight of air used.

The volumetric efficiency is the ratio of the volume of air which the engine actually takes in per cycle of two revolutions to the total piston displacement of the engine. The air volume is computed at the temperature and pressure existing at the entrance to the carburetor.

The brake thermal efficiency is the ratio of the heat equivalent of brake horsepower to the heat equivalent of fuel supplied. Since the temperature in the engine cylinder is so high as to prevent the condensation of water vapor resulting from combustion, the heat that would be liberated in such a case (the difference between the upper and lower heating value of the fuel) can not be used by the engine. Hence in calculating thermal efficiencies the lower heating value is used which for X gasoline is 18,940 B. t. u. per pound (34,100 cal. per gram).

In calculating the heat distribution in Table 2, however, the higher heating value of the fuel (20,320 B. t. u. per pound (36,600 cal. per gram)) is used because in the calorimeter used for obtaining exhaust heat the water vapor resulting from combustion is condensed. Residual heat is obtained by difference. It includes, and in fact its chief element is, the heat equivalent of the unburned fuel which goes out of the exhaust. It will be noted that no consideration has been given to the power developed by the lubricating oil burned. The difficulties in determining just how much of the oil consumed is actually burned on the power stroke, together with the probability that this percentage is not greatly different for engines of similar type, have made it seem best to ignore this factor in heat balances up to the present time.

The brake horsepower and brake mean effective pressure obtained on the ground run are converted to values for standard barometric pressure by multiplying the values actually obtained by the ratio of 29.9 to the actual barometric pressure in inches of mercury.

The results shown in Table 9 are taken from the curves at even speeds. The indicated horsepower is obtained by adding the brake horsepower to the friction horsepower. The mechanical efficiency is obtained by dividing the brake horsepower by the indicated horsepower. In obtaining the value of friction horsepower at different densities, its value at the ground and at 15,000 feet (4,570 meters) was taken and it was assumed to vary linearly between these points. Previous tests justify this assumption.

RESULTS.

The more important results of the ground tests are shown on curve sheets 1 to 5, inclusive. Curve sheet 1 shows the maximum measured brake mean effective pressure to have been 124 pounds per square inch (8.7 kg. per sq. cm.) at a speed of about 1,600 r. p. m. The maximum brake horsepower measured was 343 (348 metric horsepower) at 2,200 r. p. m., with the indication that this would have increased slightly at higher speed. The atmospheric pressure was such as would be equivalent to an altitude of about 500 feet (150 meters) and the slightly higher results that would be expected under standard barometric pressure are given on curve sheet 2. This shows a maximum brake mean effective pressure of 128 pounds per square inch (9 kg. per sq. cm.) and a maximum brake horsepower of 352 (357 metric horsepower). Curve sheet 3 shows indicated horsepower, that is, the horsepower obtained by adding to the brake horsepower the friction horsepower at that speed, plotted against r. p. m. The lower curve shows the dependence of power upon charge weight by presenting at each speed the ratios of the indicated horsepower and pounds of air per hour at that speed to their values at 2,200 r. p. m. The mechanical efficiency is shown to vary from 88 per cent to 83 per cent over the speed range tested, while the brake thermal efficiency, based on the lower calorific value of the fuel, maintains a constant value of 26 per cent over the same range. In studying the curve of pounds of air per pound of fuel on sheet 4, it must be remembered that the carburetor was adjusted for each speed so that the shape of this curve does not indicate a carburetor characteristic. Curve sheet 5 shows the heat distribution. At 1,800 r. p. m., the normal speed of the engine, the heat in the fuel supplied is about 4.1 times that realized in brake horsepower and the heat in the jacket is about half that developed in brake horsepower. Under the same conditions the heat in the exhaust is about 1.7 times and the residual about equal to the heat equivalent of the brake horsepower. It should be remembered that the residual heat is the difference between the heat in the fuel and that which appears in brake horsepower, in the jacket, and as heat in the exhaust. Hence the residual heat includes and is chiefly composed of the heat value of the unburned fuel in the exhaust.

The curve sheets 6 to 8, inclusive, show the effect of change of altitude on engine performance. Since it is the change in density caused by change in altitude that is the fundamental cause of these changes, it is against air density that curves are plotted. That the results may be conveniently interpreted from a pressure standpoint vertical lines have been drawn upon which approximate barometric pressure are noted.

Prior to any careful analysis of the altitude curves the curves of pounds of fuel per brake horsepower hour on curve sheet 6 and pounds of air per pound of fuel on curve sheet 9 should be examined. The mixture will be seen to have been very rich at altitudes of 20,000 and 25,000 ft. (6,040 and 7,620 meters) due to the fact that the carburetor adjustment was not sufficient to permit the necessary decrease in fuel flow at those altitudes. Extreme richness, of course, manifests itself both in a reduction and fluctuation in speed and torque. Curve sheet 6 shows the brake mean effective pressure and brake horsepower to vary linearly with density up to the point where the mixture becomes abnormal. Curve sheet 7 shows that at 1,800 r. p. m. and at a density of 0.040 pounds per cubic foot (.6.4 kg. per cu. m.) the brake horsepower is about 42 per cent of that at the ground. This curve sheet also shows the percentage decrease in indicated horsepower for a reduced density to be considerably greater than the decrease in pounds of air used by the engine. On curve sheet 10 it should be borne in mind that it is the carburetion that is directly repsonsible for the high "heat in fuel over heat in brake horsepower" and "residual heat over heat in brake horsepower" values, and that indirectly it is responsible for the final high values of "heat in jacket over heat in brake horsepower" through the resulting low power.

Those curves on propeller load work on curve sheets 11 and 12 which show mixture ratios or fuel consumption are influenced primarily by carburetor characteristics, since its only adjustment was at the maximum speed, 1,800 r. p. m.

CONCLUSIONS.

The information in such a report as this will be of most value when compared with results of similar tests on other engines. It then serves as a basis for comparing the relative merits of the two engines and as a means of explaining the superiority of one engine over another in any particular phase of performance.

The test shows the inadequacy of the carburetor altitude control of air-fuel ratios for heights above 20,000 feet (6,040 meters). It also shows how the relative importance of high mechanical efficiency increases with altitude.

WASHINGTON, D. C., *May 12, 1920.*

TABLE I.—*English units.*

Ground runs. Full power.

Run No.	Approximate altitude in ft.	R. p. m.	Torque, lb. ft.	B. m. e. p., lb. per sq. inch.	B. h. p.	Lb. of fuel per hour.	Lb. of fuel per b. h. p. hour.
1 A	500	1,420	915	122.6	248	128	0.52
2 A	500	1,640	930	124.7	292	148	.51
3 A	500	1,840	917	122.9	317	166	.52
4 A	500	1,980	877	117.4	330	169	.51
5 A	500	2,190	820	110.0	342	179	.52

Run No.	Temperature, degrees F.					Oil pressure, lb. per sq. inch.	Manifold suction, inches hg.		Barometric pressure, inches hg.
	Oil inlet.	Oil outlet.	Jacket water inlet.	Jacket water outlet.	Carburetor air.		R.	L.	
1 A	96	136	87	110	59	65	1.0	1.0	29.4
2 A	123	159	88	106	59	63	1.2	1.1	29.3
3 A	124	170	88	110	60	63	1.5	1.3	29.2
4 A	....	162	87	110	52	65	1.8	1.5	29.1
5 A	....	166	92	107	59	63	1.9	1.8	29.0

TABLE II.—*English units.*

Ground runs. Full power.

Run No.	Heat distribution based on b. h. p.				Heat distribution based on heat in fuel.			
	Heat in fuel ÷(heat in b. h. p.).	Heat in jacket ÷(heat in b. h. p.).	Heat in exhaust ÷(heat in b. h. p.).	Residual heat ÷(heat in b. h. p.).	B. h. p., per cent.	Jacket, per cent.	Exhaust, per cent.	Residual, per cent.
1 A	4.1	0.41	1.9	0.8	24	10	46	20
2 A	4.0	.40	1.7	.9	25	10	42	23
3 A	4.2	.48	1.7	1.0	24	11	40	25
4 A	4.1	.55	1.9	.6	24	13	47	16
5 A	4.2	.39	2.0	.8	24	9	47	20

Run No.	Air density, lb per cu. ft.	Lb. air per hr.	Volumetric efficiency, per cent.	Thermal efficiency, per cent.	Lb. air per lb. of fuel, ±0.2.
1 A	0.075	1,870	90	26	14.6
2 A	.075	2,150	89	26	14.5
3 A	.075	2,380	90	26	14.3
4 A	.075	2,580	89	26	15.2
5 A	.074	2,770	87	26	15.2

TABLE III.—*English units.*

Altitude runs. Full power.

Run No.	Approximate altitude in ft.	R. p. m.	Torque, lb. ft.	B. m. e. p.	B. h. p.	Lb. of fuel per hr.	Lb. of fuel per b. h. p. hr.
11 A	Ground..	1, 600	930	124. 5	283	150	0. 53
12 A	Ground..	1, 800	938	123. 6	316	171	. 54
13 A	5, 000	1, 610	784	105. 1	241	129	. 53
14 A	5, 000	1, 790	772	103. 1	264	142	. 54
15 A	10, 000	1, 600	632	84. 7	193	115	. 60
16 A	10, 000	1, 810	628	84. 0	216	122	. 56
17 A	15, 000	1, 590	511	68. 3	155	95	. 61
18 A	15, 000	1, 790	499	66. 6	170	101	. 59
19 A	20, 000	1, 620	345	46. 1	107	92	. 86
20 A	20, 000	1, 820	383	51. 4	133	92	. 69
21 A	25, 000	1, 780	224	39. 9	76	90	1. 18
22 A	25, 000	1, 600	235	31. 5	72	80	1. 12

Run No.	Temperature, degrees F.					Oil pressure, lb. per sq. in.	Manifold suction, inches hg.		Barometric pressure, inches hg.
	Oil inlet.	Oil outlet.	Jacket water inlet.	Jacket water outlet.	Carburetor, air.		*R.*	*L.*	
11 A	104	141	91	117	59	66	1. 3	1. 2	29. 4
12 A	124	164	92	111	60	64	1. 5	1. 4	29. 3
13 A	113	151	88	108	58	66	1. 1	1. 1	24. 9
14 A	115	157	91	107	41	66	1. 2	1. 2	24. 9
15 A	121	152	92	111	26	64	. 6	. 8	20. 7
16 A	124	158	94	111	26	64	. 6	1. 0	20. 7
17 A	121	156	95	111	22	64	...	. 6	17. 2
18 A	121	157	97	112	19	64	. 1	. 6	17. 0
19 A	118	151	94	107	13	65	. 1	. 6	14. 0
20 A	120	152	98	111	11	65	...	. 6	14. 2
21 A	118	153	97	107	12	62	. 1	. 6	11. 7
22 A	116	151	94	105	11	63	. 2	. 5	11. 7

TABLE IV.—*English units.*

Altitude runs. Full power.

Run No.	Heat distribution based on b. h. p.				Heat distribution based on heat in fuel.			
	Heat in fuel ÷(heat in b. h. p.)	Heat in jacket÷(heat in b. h. p.)	Heat in exhaust÷(heat in b. h. p.)	Residual heat÷(heat in b. h. p.)	B. h. p., per cent.	Jacket, per cent.	Exhaust, per cent.	Residual, per cent.
11 A	4. 2	0. 59	2. 0	0. 6	24	14	48	14
12 A	4. 3	. 46	1. 8	1. 0	23	11	42	24
13 A	4. 3	. 54	1. 8	. 9	23	13	43	21
14 A	4. 3	. 45	1. 8	1. 0	23	10	43	24
15 A	4. 8	. 65	1. 8	1. 3	21	13	38	28
16 A	4. 5	. 57	1. 8	1. 1	22	13	41	24
17 A	4. 9	. 65	1. 9	1. 3	20	13	39	28
18 A	4. 7	. 64	1. 9	1. 2	21	14	40	25
19 A	6. 8	. 76	1. 9	3. 2	15	11	28	46
20 A	5. 6	. 67	2. 1	1. 8	18	12	38	32
21 A	9. 4	. 96	2. 3	5. 2	11	10	24	55
22 A	8. 9	. 98	2. 0	4. 9	11	11	22	56

Run No.	Air density, lb. per cu. ft.	Lb. air per hr.	Volumetric efficiency, per cent.	Thermal efficiency, per cent.	Lb. air per lb. of fuel, ±0.2.
11 A	0. 072	2, 150	91	25	14. 3
12 A	. 075	2, 380	91	25	13. 9
13 A	. 064	1, 840	91	25	14. 2
14 A	. 066	2, 090	90	25	14. 7
15 A	. 056	1, 570	89	23	13. 6
16 A	. 056	1, 790	89	24	14. 6
17 A	. 047	1, 340	91	22	14. 2
18 A	. 047	1, 440	87	23	14. 2
19 A	. 039	1, 100	88	16	12. 0
20 A	. 040	1, 240	88	19	13. 5
21 A	. 033	1, 030	89	11	11. 5
22 A	. 033	930	91	12	11. 6

TABLE V.—*English units.*

Propeller load runs.

Run No.	Approximate altitude in ft.	R. p. m.	Torque, lb. ft.	B. m. e. p., lb. per sq. in.	B. h. p.	Lb. of fuel per hr.	Lb. of fuel per b. h. p. hr.	Barometric pressure, inches hg.
1 B	15, 000	1, 790	283	66. 2	168	100	0. 59	17. 2
2 B	15, 000	1, 680	253	58. 9	142	86	. 60	17. 2
3 B	15, 000	1, 580	223	52. 1	117	74	. 63	17. 1
4 B	15, 000	1, 500	199	46. 3	100	72	. 72	17. 2
5 B	15, 000	1, 390	170	39. 5	78	66	. 84	17. 2
6 B	10, 000	1, 790	358	83. 7	214	117	. 55	20. 7
7 B	10, 000	1, 690	323	75. 3	183	93	. 51	20. 8
8 B	10, 000	1, 620	288	67. 5	155	89	. 57	20. 7
9 B	10, 000	1, 490	253	59. 3	126	72	. 57	20. 7
10 B	10, 000	1, 420	221	51. 7	104	71	. 68	20. 7
11 B	5, 000	1, 780	437	102. 2	259	131	. 51	24. 9
12 B	5, 000	1, 700	394	92. 1	223	110	. 49	25. 0
13 B	5, 000	1, 610	348	81. 4	187	92	. 49	25. 0
14 B	5, 000	1, 510	318	74. 4	160	85	. 53	25. 1
15 B	5, 000	1, 410	266	62. 2	126	78	. 62	25. 0

TABLE VI.—*English units.*

Propeller load runs.

Run No.	Temperature, degrees F.					Manifold suction, inches hg.		Air density, lb. per cu. ft.	Lb. of air per hr.	Lb. of air per lb. of fuel, ±0.2.
	Oil inlet.	Oil outlet.	Jacket water inlet.	Jacket water outlet.	Carburetor, air.	*R.*	*L.*			
1 B	113	152	95	112	20	0. 9	0. 8	0. 048	1, 450	14. 5
2 B	124	157	97	112	19	2. 7	2. 7	. 048	1, 180	13. 7
3 B	123	155	95	110	19	3. 7	4. 0	. 047	990	13. 4
4 B	120	150	94	108	22	4. 8	4. 7	. 047	870	12. 1
5 B	116	146	95	110	14	5. 5	5. 4	. 048	770	11. 7
6 B	119	157	98	114	27	1. 1	1. 1	. 057	1, 690	14. 4
7 B	122	158	97	113	27	2. 7	2. 9	. 057	1, 420	15. 3
8 B	122	157	95	113	27	4. 6	4. 5	. 056	1, 230	13. 9
9 B	117	151	94	108	27	5. 9	5. 8	. 056	1, 030	14. 3
10 B	114	147	92	106	27	7. 1	6. 5	. 056	900	12. 7
11 B	120	158	97	118	43	1. 3	1. 2	. 066	1, 990	15. 2
12 B	122	162	93	111	41	3. 3	3. 4	. 066	1, 680	15. 2
13 B	119	157	92	111	41	5. 7	5. 5	. 066	1, 450	16. 7
14 B	111	150	91	108	40	7. 0	6. 7	. 066	1, 250	14. 8
15 B	108	144	89	105	41	8. 2	8. 4	. 066	1, 030	13. 1

TABLE VII.—*English units.*

Friction horsepower.

Run No.	Approximate altitude, feet.	R. p. m.	Friction h. p.	Barometric pressure, inches hg.	Air density, lb. per cu. ft.
29 B	15, 000	1, 420	29	17. 0	0. 044
30 B	15, 000	1, 600	35	17. 0	. 044
31 B	15, 000	1, 800	42	17. 1	. 044
32 B	15, 000	1, 990	50	17. 0	. 044
33 B	15, 000	2, 170	58	17. 1	. 044
34 B	Ground	1, 390	33	29. 3	. 076
35 B	Ground	1, 610	43	29. 2	. 075
36 B	Ground	1, 780	52	29. 1	. 075
37 B	Ground	1, 980	61	29. 1	. 075
38 B	Ground	2, 180	75	29. 1	. 075

Temperature, degrees F.

Run No.	Oil inlet.	Oil outlet.	Jacket inlet.	Jacket outlet.	Carburetor, air.
29 B	113	143	116	118	56
30 B	115	139	120	121	57
31 B	117	142	122	124	57
32 B	119	146	126	128	56
33 B	123	153	129	131	52
34 B	124	151	121	123	53
35 B	126	147	115	117	53
36 B	124	147	104	106	54
37 B	127	149	108	108	54
38 B	129	154	113	115	54

TABLE VIII.—*English Units.*

Ground and altitude runs.

R. p. m.	B. h. p.	F. h. p.	I. h. p.	Lb. air per hr.÷(lb. air per hr. at 2,200 r. p. m.).	I. h. p. ÷ (i. h. p. at 2,200 r. p. m.).	Mechanical efficiency, per cent.	Approximate air density, lb. per cu. ft.
1,400	243	34	277	0.66	0.66	88	0.075
1,600	284	43	327	.75	.79	87	.075
1,800	315	53	368	.84	.89	85	.075
2,000	334	62	396	.94	.96	84	.075
2,200	343	72	415	1.00	1.00	83	.075

Air density, lb. per cu. ft.	B. h. p.	F. h. p.	I. h. p.	Lb. air per hr.÷(lb. air per hr. at 0.075 density).	I. h. p. ÷ (i. h. p. at 0.075 density).	Mechanical efficiency, per cent.	R. p. m.	B. h. p. ÷ (b. h. p. at 0.075 density).
0.075	282	43	325	1.00	1.00	87	1,600	1.00
.065	235	41	276	.86	.85	85	1,600	.83
.055	189	38	227	.73	.70	83	1,600	.67
.045	141	36	177	.60	.54	80	1,600	.50
.040	116	34	150	.53	.46	77	1,600	.41
.035	83	33	116	.46	.36	71	1,600	.29
.075	318	53	371	1.00	1.00	86	1,800	1.00
.065	263	50	313	.86	.83	84	1,800	.83
.055	210	47	257	.73	.66	82	1,800	.66
.045	158	43	201	.59	.50	78	1,800	.50
.040	131	42	173	.52	.41	76	1,800	.41
.035	96	40	136	.46	.30	71	1,800	.30

TABLE I.—*Metric Units.*

Ground runs. Full power.

Run No.	Approximate altitude in meters.	R. p. m.	Torque, kg. meters.	B. m. e. p., kg. per sq. cm.	B. h. p.	Kg. of fuel per hr.	Kg. of fuel per b. h. p. hr.
1 A	150	1,420	126	8.6	251	58	0.23
2 A	150	1,640	128	8.8	296	67	.23
3 A	150	1,840	127	8.6	322	75	.23
4 A	150	1,980	121	8.2	335	77	.23
5 A	150	2,190	113	7.7	347	81	.23

Run No.	Temperature, degrees C.					Oil pressure, kg. per sq. cm.	Manifold suction, cm. hg.		Barometric pressure, cm. hg.
	Oil inlet.	Oil outlet.	Jacket water inlet.	Jacket water outlet.	Carburetor, air.		R.	L.	
1 A	35	58	31	43	15	4.6	2.5	2.4	74.7
2 A	50	71	31	41	15	4.4	3.0	2.9	74.4
3 A	52	76	31	43	15	4.4	3.7	3.4	74.3
4 A	....	72	31	43	15	4.6	4.6	3.8	74.0
5 A	....	74	33	42	15	4.4	4.9	4.5	73.7

TABLE II.—*Metric units.*

Ground runs. Full power.

Run No.	Heat distribution based on b. h. p.				Heat distribution based on heat in fuel.			
	Heat in fuel÷(heat in b. h. p.).	Heat in jacket ÷ (heat in b. h. p.).	Heat in exhaust ÷ (heat in b. h. p.).	Residual heat ÷ (heat in b. h. p.).	B. h. p., per cent.	Jacket, per cent.	Exhaust, per cent.	Residual, per cent.
1 A	4. 1	0. 41	1. 9	0. 8	24	10	46	20
2 A	4. 0	. 40	1. 7	. 9	25	10	42	23
3 A	4. 2	. 48	1. 7	1. 0	24	11	40	25
4 A	4. 1	. 55	1. 9	. 6	24	13	47	15
5 A	4. 2	. 39	2. 0	. 8	24	9	47	20

Run No.	Air density, kg. per cu. m.	Kg. air per hr.	Volumetric efficiency, per cent.	Brake thermal efficiency, per cent.	Kg. air per kg. of fuel, ±0.2.
1 A	1. 20	850	90	26	14. 6
2 A	1. 20	970	89	26	14. 5
3 A	1. 20	1, 080	90	26	14. 3
4 A	1. 20	1, 170	89	26	15. 2
5 A	1. 19	1, 260	87	26	15. 2

TABLE III.—*Metric units.*

Altitude runs. Full power.

Run No.	Approximate altitude in meters.	R. p. m.	Torque in kg. meters.	B. m. e. p. kg. per sq. cm.	B. h. p.	Kg. of fuel per hour.	Kg. of fuel per b. h. p. hr.
11 A	Ground	1, 600	128	8. 8	287	68	0. 24
12 A	Ground	1, 800	130	8. 7	320	78	. 24
13 A	1, 520	1, 610	108	7. 4	244	59	. 24
14 A	1, 520	1, 790	107	7. 2	268	64	. 24
15 A	3, 050	1, 600	87	6. 0	195	52	. 27
16 A	3, 050	1, 810	87	5. 9	219	55	. 25
17 A	4, 570	1, 590	71	4. 8	157	43	. 27
18 A	4, 570	1, 790	69	4. 7	173	46	. 26
19 A	6, 040	1, 620	48	3. 2	109	42	. 38
20 A	6, 040	1, 820	53	3. 6	135	42	. 31
21 A	7, 620	1, 780	31	2. 1	77	41	. 53
22 A	7, 620	1, 600	32	2. 2	73	36	. 50

Run No.	Temperature, degrees C.					Oil pressure, kg. per sq. cm.	Manifold suction, cm. hg.		Barometric pressure, cm. hg.
	Oil inlet.	Oil outlet.	Jacket water inlet.	Jacket water outlet.	Carburetor air.		*R.*	*L.*	
11 A	40	61	33	47	15	4. 6	3. 4	3. 0	74. 6
12 A	51	73	33	44	16	4. 5	3. 8	3. 6	74. 4
13 A	45	66	31	42	14	4. 6	2. 7	2. 7	63. 3
14 A	46	69	33	42	5	4. 6	3. 1	3. 1	63. 3
15 A	49	67	33	44	− 3	4. 5	1. 6	2. 0	52. 5
16 A	51	70	34	44	− 3	4. 5	1. 5	2. 5	52. 5
17 A	50	69	35	44	− 6	4. 5		1. 5	43. 7
18 A	49	69	36	45	− 7	4. 5	. 2	1. 6	43. 2
19 A	48	66	34	42	−11	4. 6	. 4	1. 6	35. 6
20 A	49	67	37	44	−12	4. 6	. 1	1. 6	36. 0
21 A	48	67	36	42	−11	4. 4	. 3	1. 6	29. 8
22 A	47	66	34	41	−12	4. 4	. 4	1. 3	29. 6

TABLE IV.—*Metric units.*

Altitude runs. Full power.

Run No.	Heat distribution based on b. h. p.				Heat distribution based on heat in fuel.			
	Heat in fuel ÷ (heat in b. h. p.).	Heat in jacket ÷ (heat in b. h. p.).	Heat in exhaust ÷ (heat in b. h. p.).	Residual heat ÷ (heat in b. h. p.).	B. h. p., per cent.	Jacket, per cent.	Exhaust, per cent.	Residual, per cent.
11 A	4.2	0.59	2.0	0.6	24	14	48	14
12 A	4.3	.46	1.8	1.0	23	11	42	24
13 A	4.3	.54	1.8	.9	23	13	43	21
14 A	4.3	.45	1.8	1.0	23	10	43	24
15 A	4.8	.65	1.8	1.3	21	13	38	28
16 A	4.5	.57	1.8	1.1	22	13	41	24
17 A	4.9	.65	1.9	1.3	20	13	39	28
18 A	4.7	.64	1.9	1.2	21	14	40	25
19 A	6.8	.76	1.9	3.2	15	11	28	46
20 A	5.6	.67	2.1	1.8	18	12	38	32
21 A	9.4	.96	2.3	5.2	11	10	24	55
22 A	8.9	.98	2.0	4.9	11	11	22	56

Run No.	Air density, kg. per cu. m.	Kg. air per hr.	Volumetric efficiency, per cent.	Thermal efficiency, per cent.	Kg. air per kg. of fuel, ±0.2.
11 A	1.16	970	91	25	14.3
12 A	1.20	1,080	91	25	13.9
13 A	1.02	830	91	25	14.2
14 A	1.06	950	90	25	14.7
15 A	.90	710	89	23	13.6
16 A	.90	810	89	24	14.6
17 A	.76	610	91	22	14.2
18 A	.76	650	87	23	14.2
19 A	.63	500	88	16	12.0
20 A	.64	560	88	19	13.5
21 A	.53	470	89	11	11.5
22 A	.53	420	91	12	11.6

TABLE V.—*Metric units.*

Propeller load runs.

Run No.	Approximate altitude in meters.	R. p. m.	Torque, kg. meters.	B. m. e. p., kg. per sq. cm.	B. h. p.	Kg. of fuel per hr.	Kg. of fuel per b. h. p. hr.	Barometric pressure. cm. hg.
1 B	4,570	1,790	39	4.6	170	45	0.27	43.8
2 B	4,570	1,680	35	4.1	144	39	.27	43.8
3 B	4,570	1,580	31	3.7	119	33	.28	43.5
4 B	4,570	1,500	27	3.2	101	33	.32	43.7
5 B	4,570	1,390	23	2.8	80	30	.38	43.7
6 B	3,050	1,790	50	5.9	217	53	.25	52.6
7 B	3,050	1,690	45	5.3	185	42	.23	52.7
8 B	3,050	1,620	40	4.7	157	40	.26	52.4
9 B	3,050	1,490	35	4.2	128	33	.26	52.5
10 B	3,050	1,420	31	3.6	106	32	.30	52.5
11 B	1,520	1,780	60	7.2	263	60	.23	63.3
12 B	1,520	1,700	54	6.5	227	50	.22	63.5
13 B	1,520	1,610	48	5.7	190	42	.22	63.6
14 B	1,520	1,510	44	5.2	162	38	.24	63.7
15 B	1,520	1,410	37	4.4	128	35	.28	63.5

TABLE VI.—*Metric units.*

Propeller load runs.

Run No.	Temperature, degrees C.					Manifold suction, cm. hg.		Air density kg. per cu. m.	Kg. of air per hour.	Kg. of air per kg. of fuel, ±0.2.
	Oil inlet.	Oil outlet.	Jacket water inlet.	Jacket water outlet.	Carburetor, air.	*Right.*	*Left.*			
1 B	45	67	35	45	−7	2.4	2.1	0.76	660	14.5
2 B	51	70	36	45	−7	6.9	6.9	.77	540	13.7
3 B	51	68	35	43	−7	9.4	10.6	.76	450	13.4
4 B	49	65	34	42	−5	12.2	12.0	.76	390	12.1
5 B	47	63	35	43	−10	14.0	13.6	.77	350	11.7
6 B	48	70	37	46	−3	2.8	2.8	.91	770	14.4
7 B	50	70	36	45	−3	6.9	7.4	.91	640	15.3
8 B	50	70	35	45	−3	11.6	11.4	.90	560	13.9
9 B	47	66	34	42	−3	15.1	14.7	.90	470	14.3
10 B	46	64	33	41	−3	18.0	16.4	.90	410	12.7
11 B	49	70	36	48	+6	3.4	3.0	1.05	900	15.2
12 B	50	72	34	44	+5	8.4	8.7	1.06	760	15.3
13 B	48	70	33	44	+5	14.4	14.0	1.06	660	16.7
14 B	44	65	33	42	+5	17.8	17.1	1.07	570	14.8
15 B	42	62	32	41	+5	22.5	21.4	1.06	460	13.1

TABLE VII.—*Metric units.*

Friction horsepower.

Run. No.	Approximate altitude, meters.	R. p. m.	Friction, h. p.	Barometric pressure, cm. hg.	Air density, kg. per cu. m.
29 B	4,570	1,420	29	43.3	0.70
30 B	4,570	1,600	35	43.3	.70
31 B	4,570	1,800	42	43.5	.70
32 B	4,570	1,990	51	43.3	.70
33 B	4,570	2,170	59	43.4	.71
34 B	Ground.	1,390	34	74.3	1.21
35 B	Ground.	1,610	44	74.2	1.21
36 B	Ground.	1,780	53	74.0	1.21
37 B	Ground.	1,980	62	73.8	1.20
38 B	Ground.	2,180	76	73.8	1.20

Temperature degrees C.

Run No.	Oil inlet.	Oil outlet.	Jacket inlet.	Jacket outlet.	Carburetor air.
29 B	45	62	47	48	13
30 B	46	60	49	49	14
31 B	47	61	50	51	14
32 B	48	63	52	53	13
33 B	50	67	54	55	11
34 B	51	66	50	51	12
35 B	52	64	46	47	12
36 B	51	64	40	41	12
37 B	53	65	42	42	12
38 B	54	68	45	46	12

TABLE VIII.—*Metric units.*

Ground and altitude runs.

R. p. m.	B. h. p.	F. h. p.	I. h. p.	Kg. air per hr. ÷ (kg. air per hour at 2,200 r. p. m.)	I h. p. ÷ (I. h. p. at 2,200 r. p. m.)	Mechanical efficiency, per cent.	Approximate air density kg. per cu. m.
1,400	246	34	280	0.66	0.66	88	1.20
1,600	288	44	332	.75	.79	87	1.20
1,800	319	54	373	.84	.89	85	1.20
2,000	339	63	402	.94	.96	84	1.20
2,200	348	73	421	1.00	1.00	83	1.20

Air density kg. per cu. m.	B. h. p.	F. h. p.	I. h. p.	Kg. air per hr. ÷ (kg. air per hour at 1.22 density).	I. h. p. ÷ (I. h. p. at 1.22 density).	Mechanical efficiency, per cent.	R. p. m.	B. h. p. ÷ (B. h. p. at 1.22 density).
1.20	286	44	330	1.00	1.00	87	1,600	1.00
1.04	238	41	279	.86	.85	85	1,600	.83
.88	192	39	231	.73	.70	83	1,600	.67
.72	143	36	179	.60	.54	80	1,600	.50
.64	118	35	153	.53	.46	77	1,600	.41
.56	84	33	117	.46	.36	71	1,600	.29
1.20	322	54	376	1.00	1.00	86	1,800	1.00
1.04	267	50	317	.86	.83	84	1,800	.83
.88	213	47	260	.73	.66	82	1,800	.66
.72	160	44	204	.59	.50	78	1,800	.50
.64	133	42	175	.52	.41	76	1,800	.41
.56	97	41	138	.46	.30	71	1,800	.30

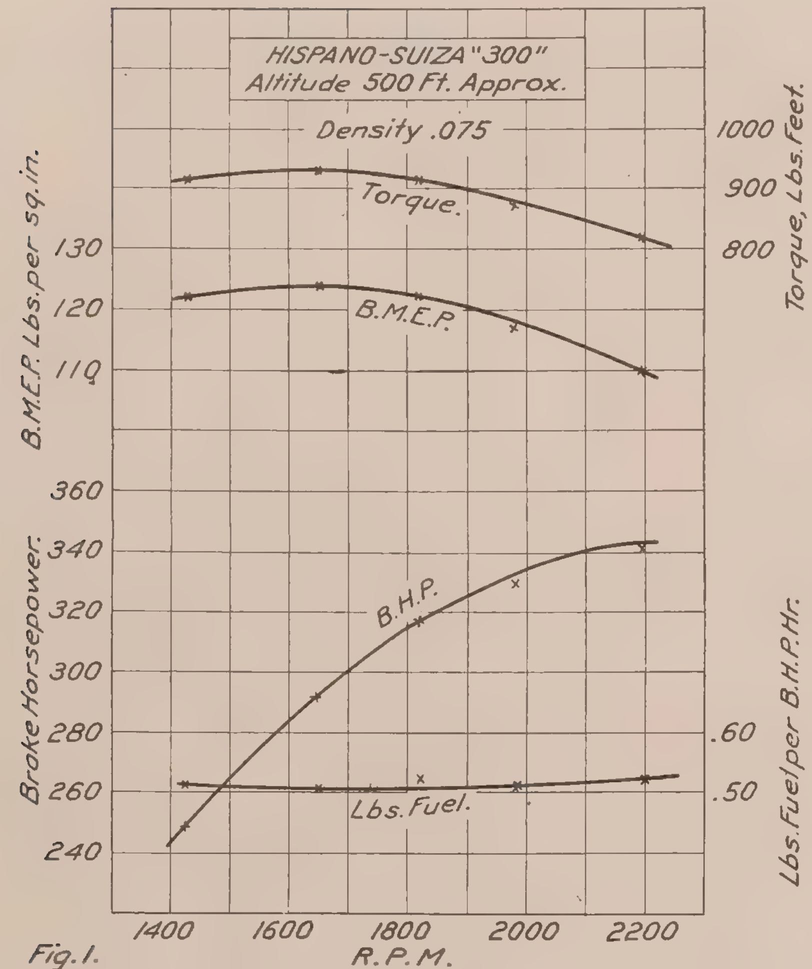
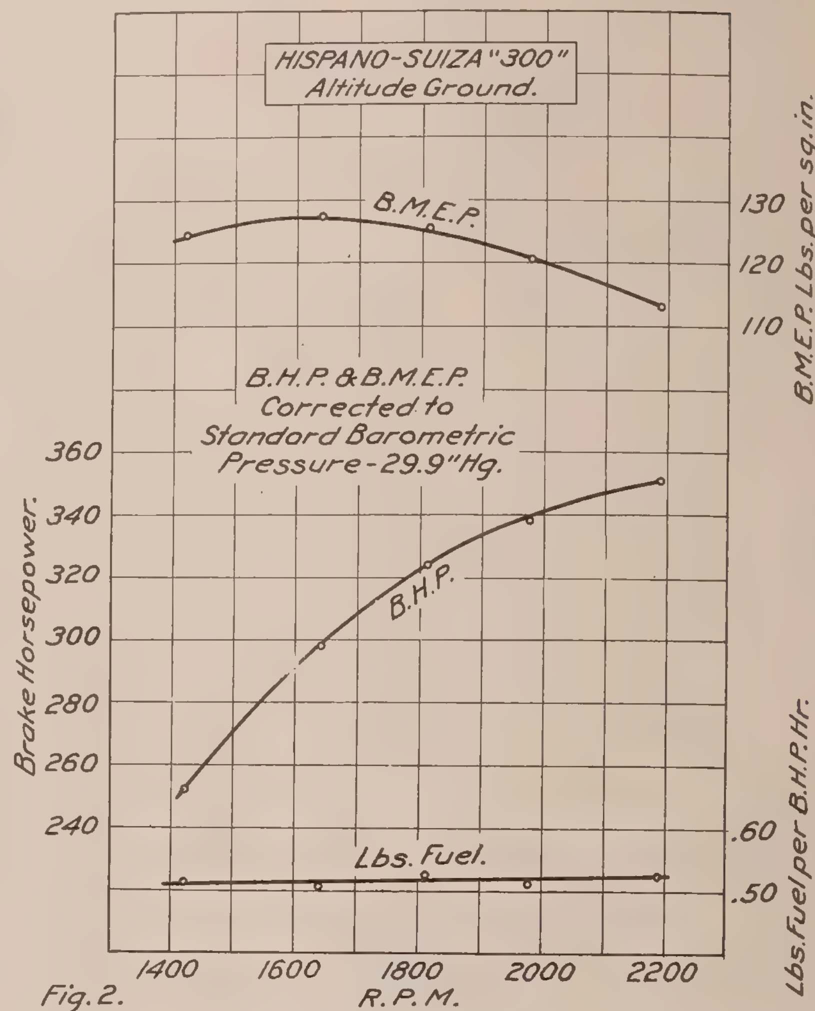


Fig. 1.



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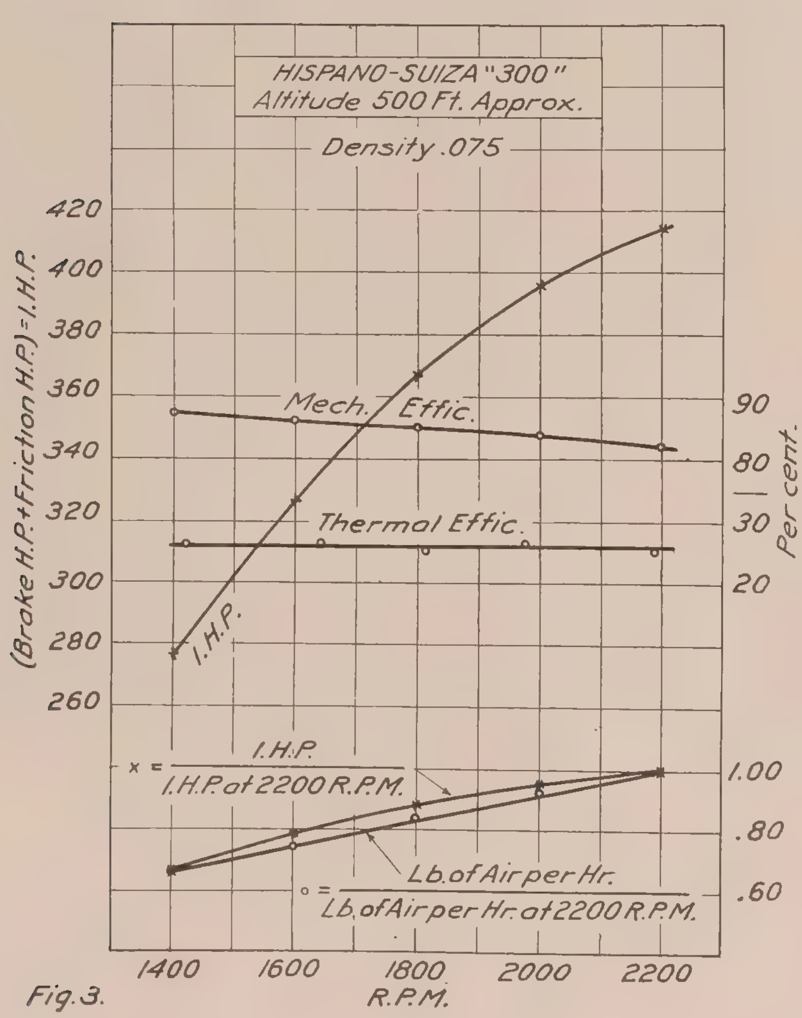

Fig. 3.

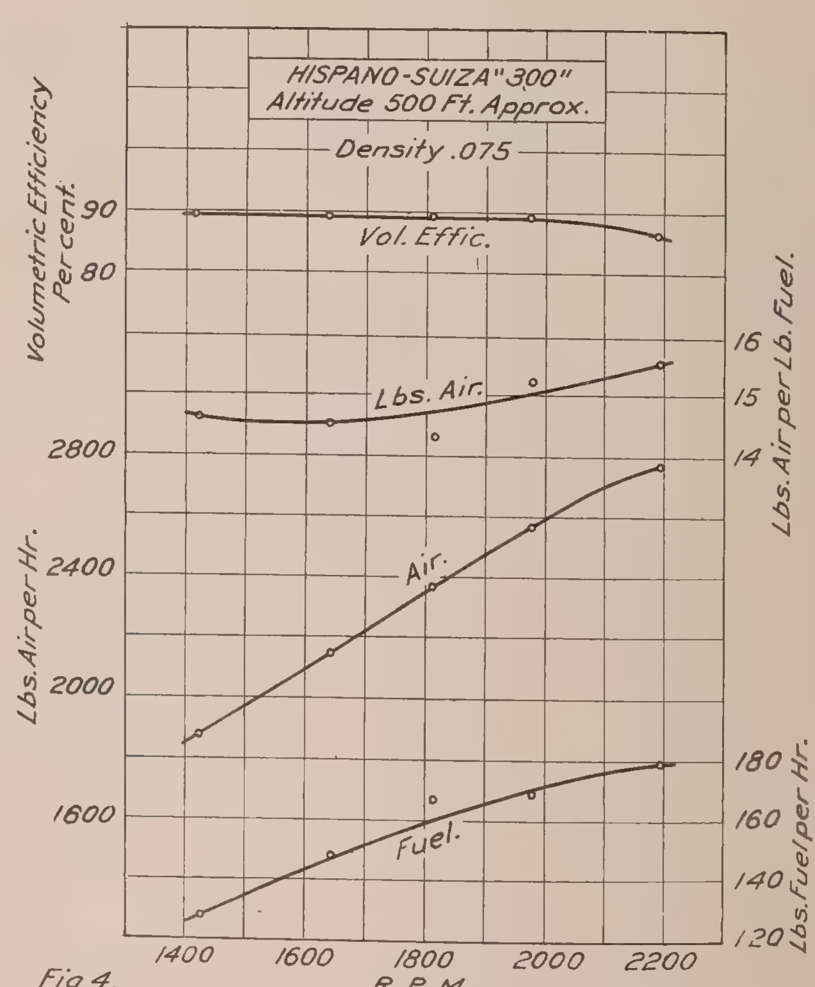

Fig. 4.

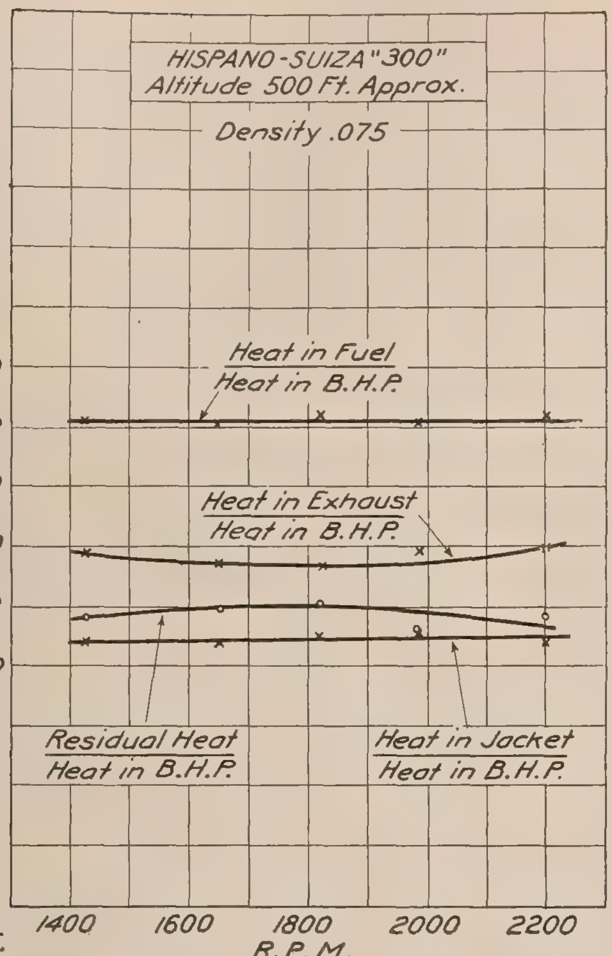

Fig. 5.

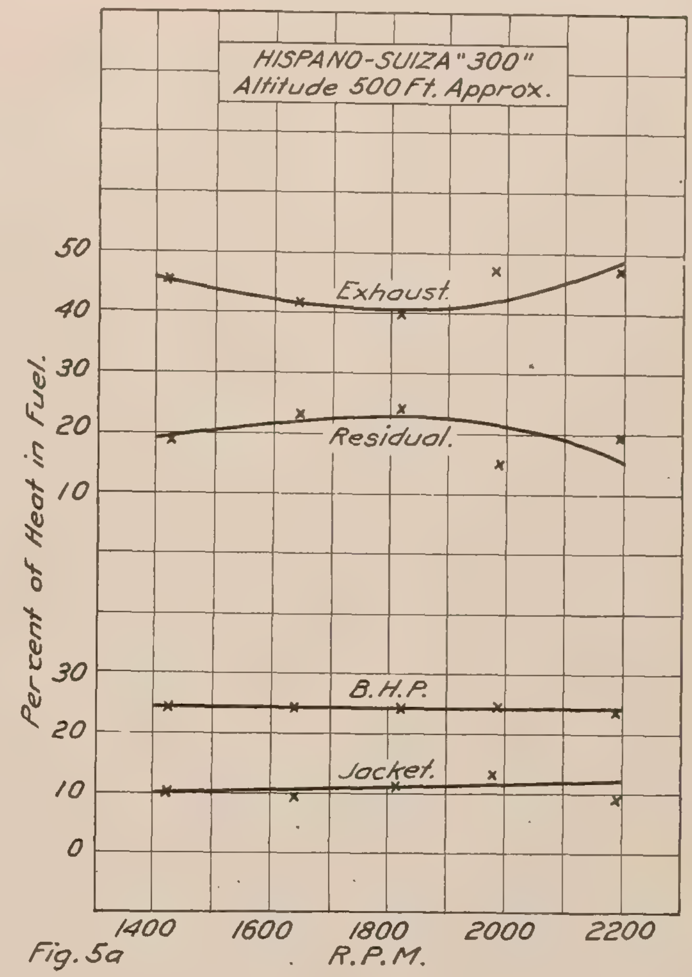

Fig. 5a

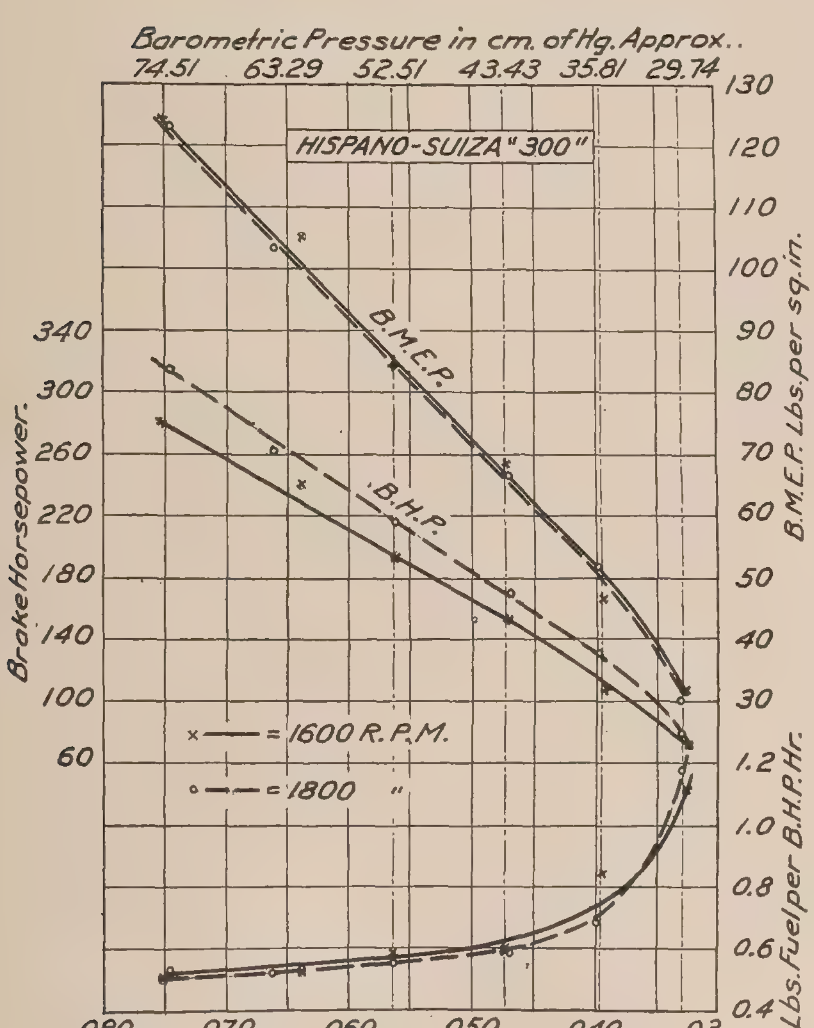

Fig. 6.

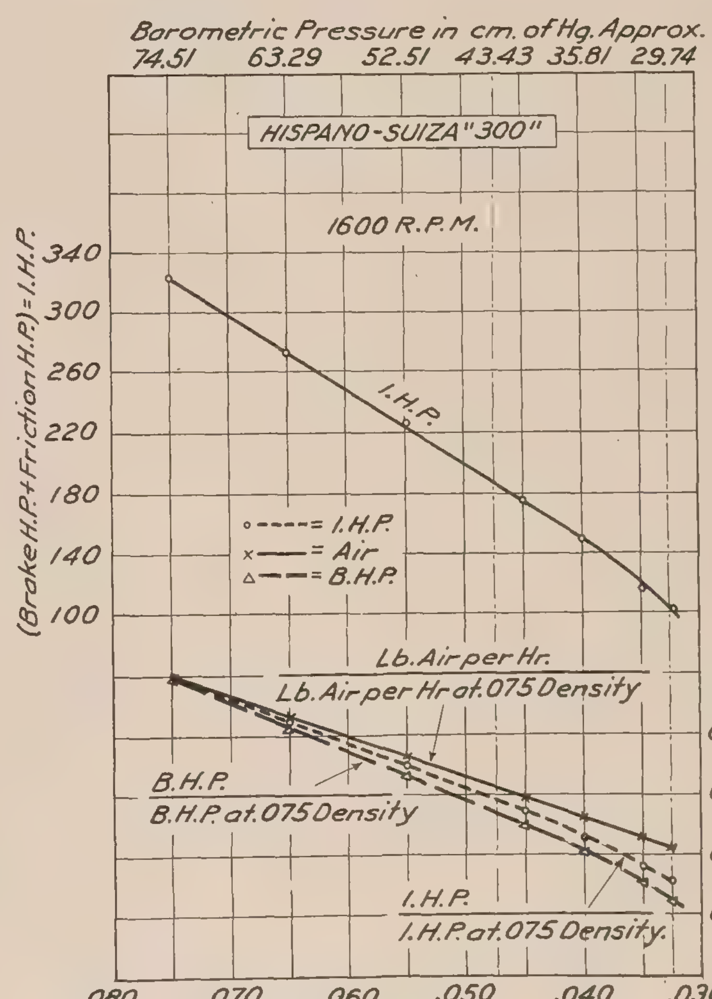

Fig. 7.

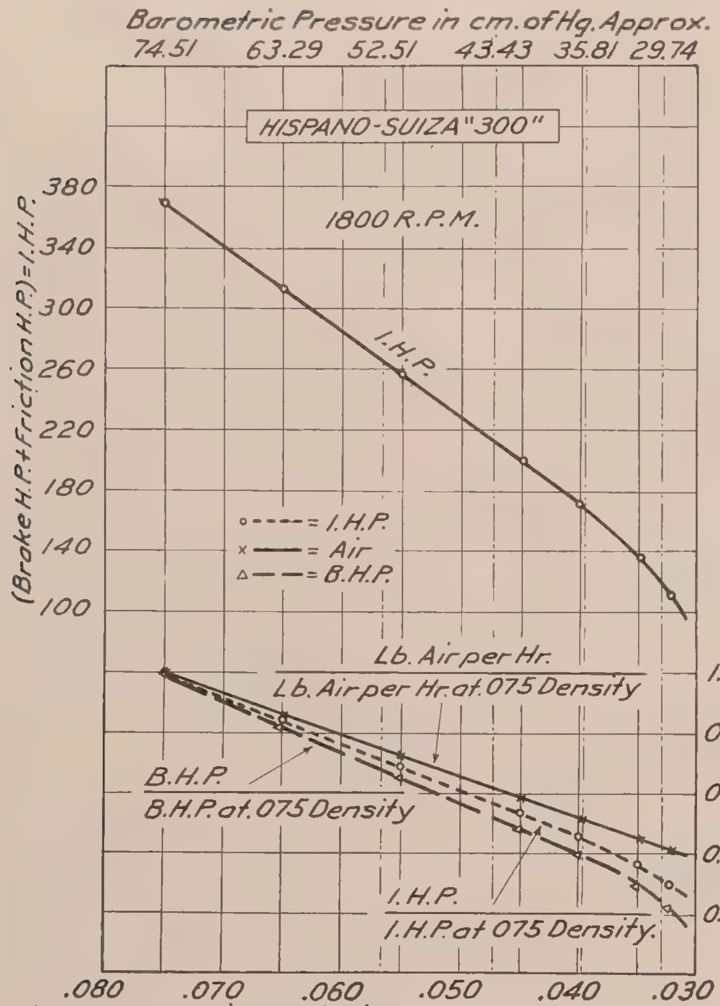

Fig. 7a.

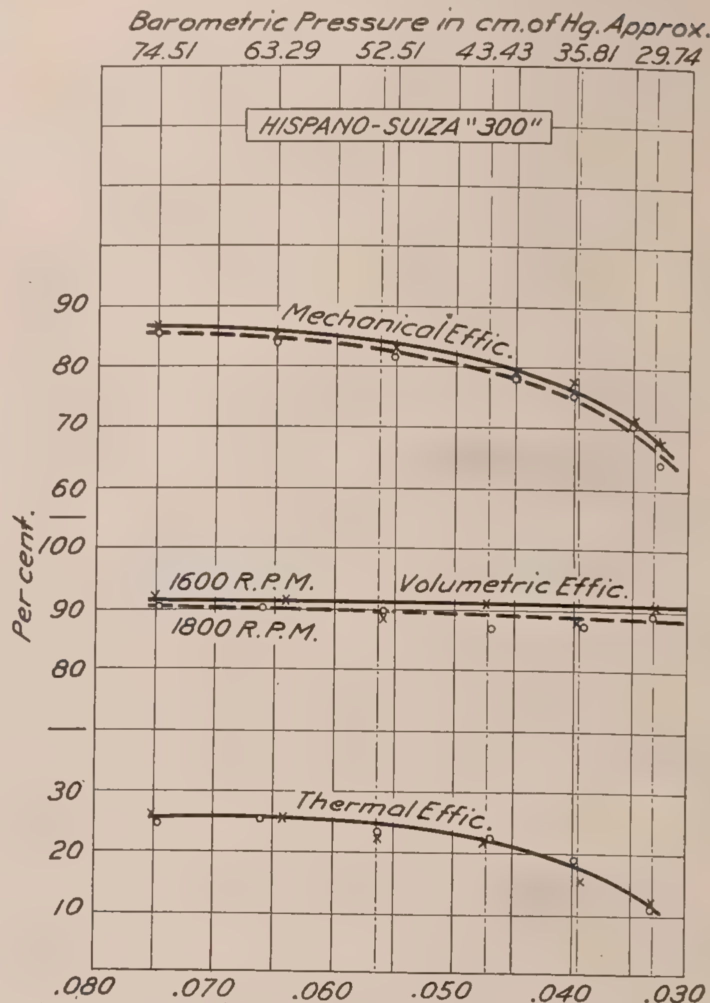

Fig. 8.

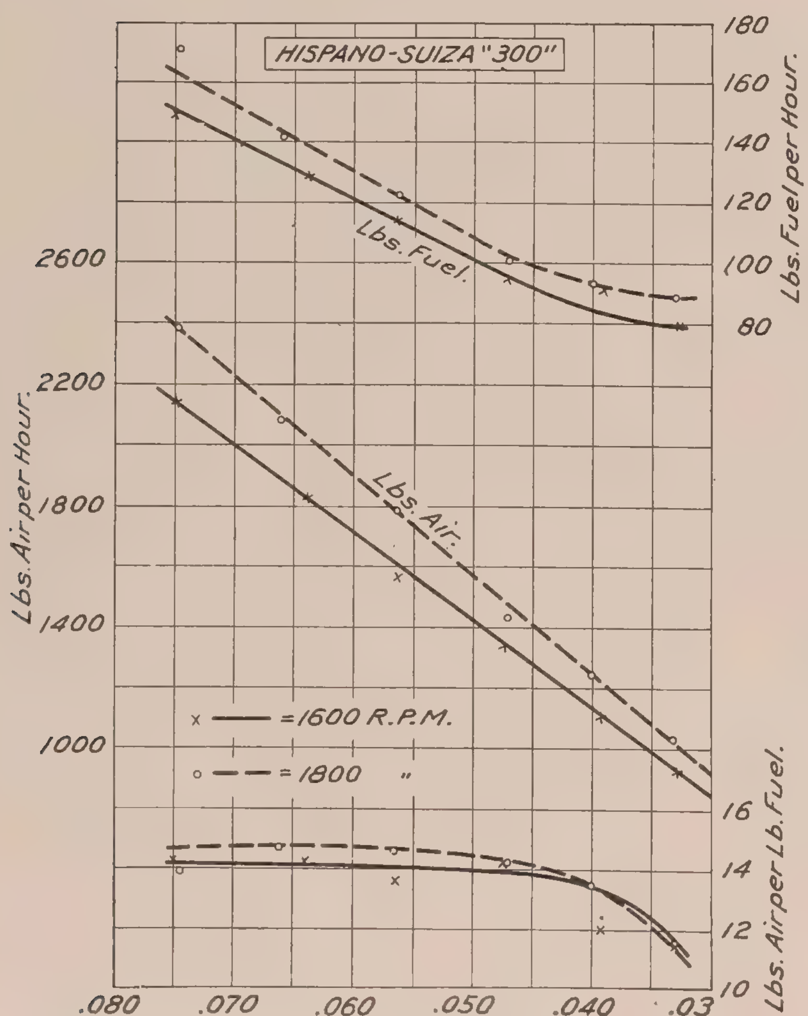

Fig. 9.

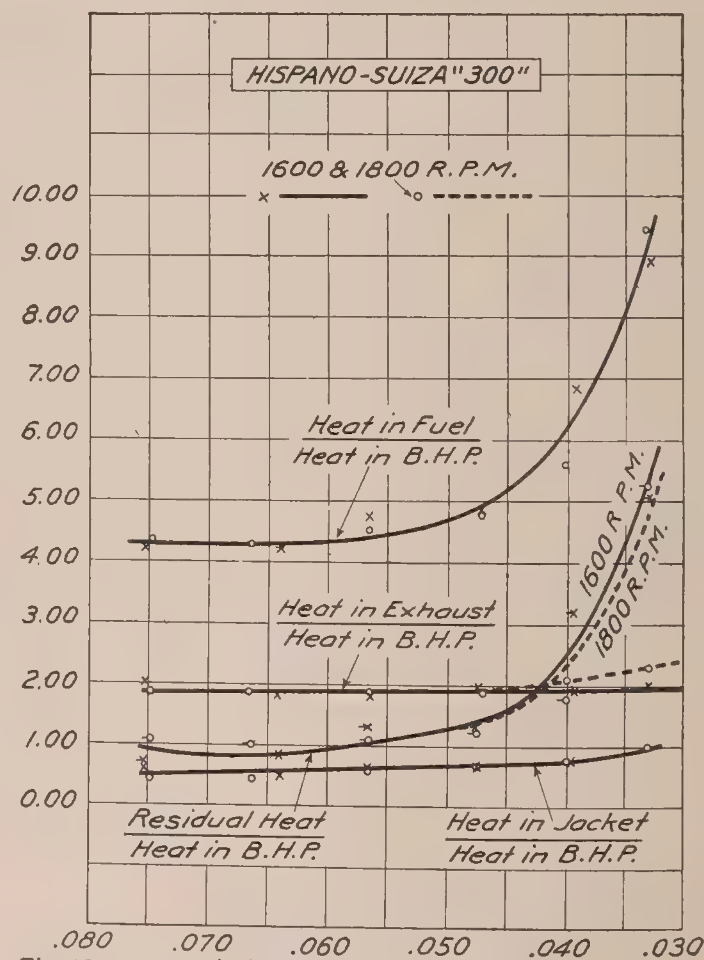

Fig. 10.

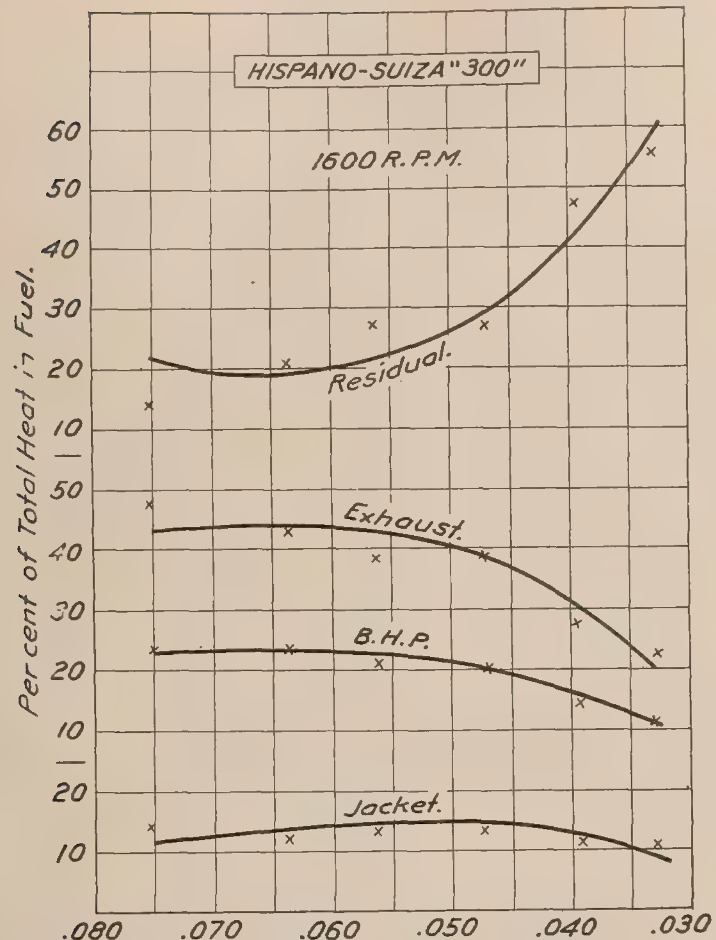

Fig. 10a.

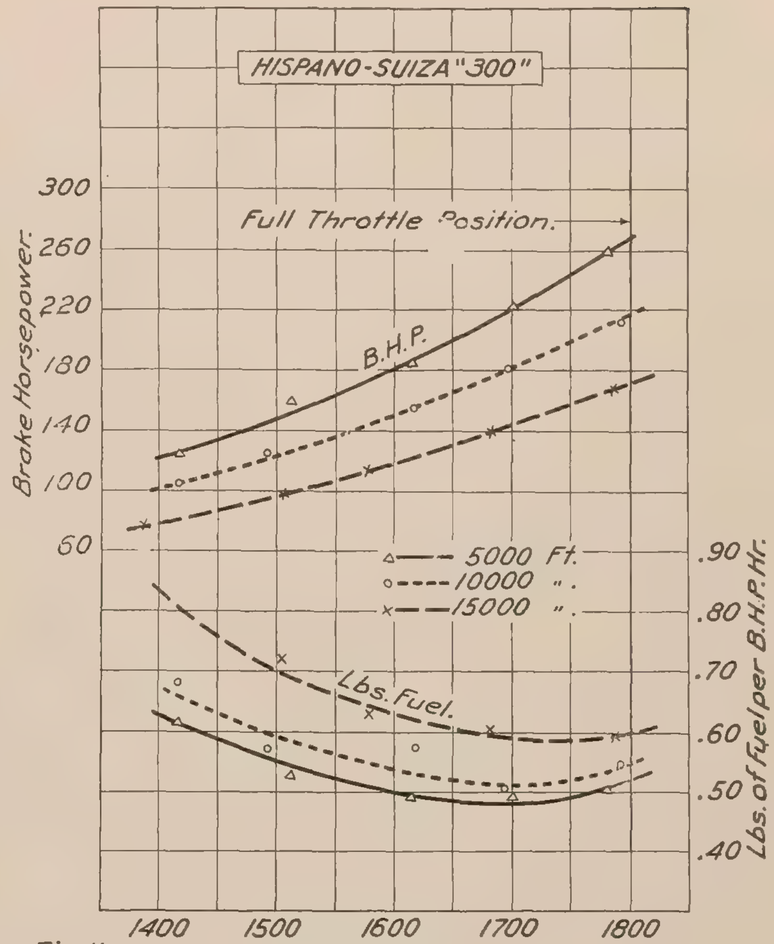

Fig. 11.

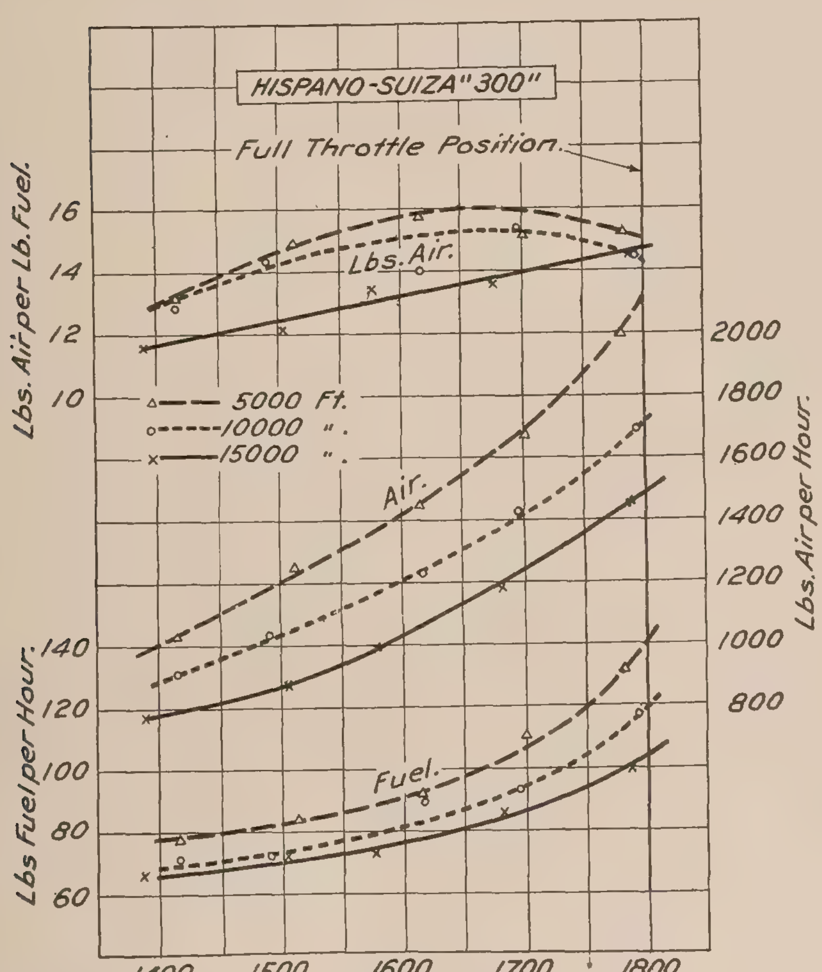

Fig. 12.

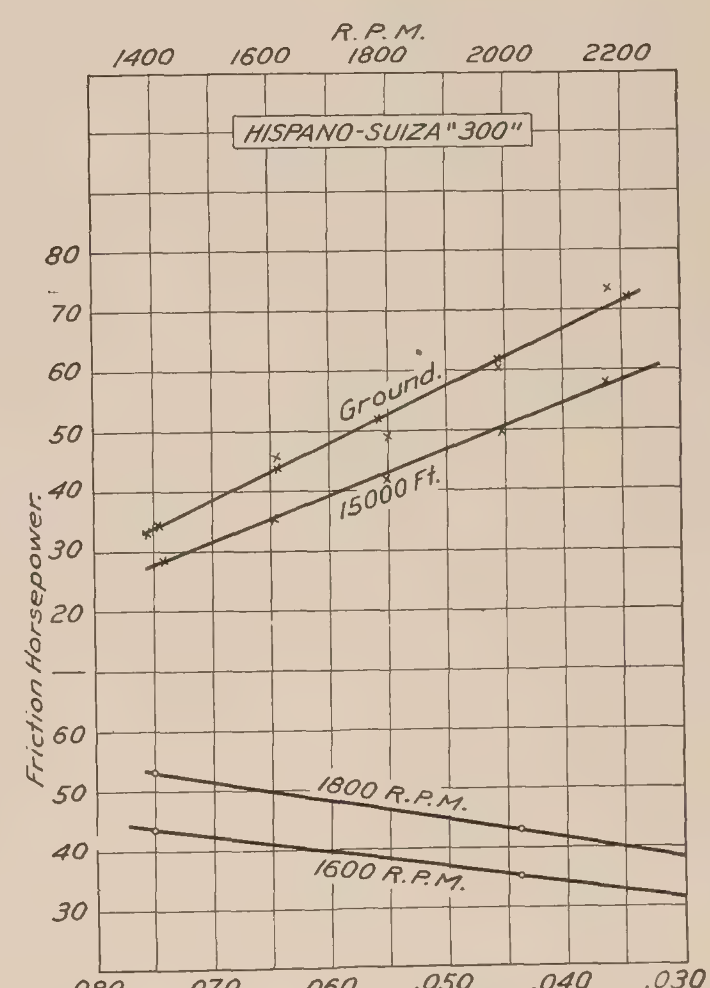

Fig. 13.

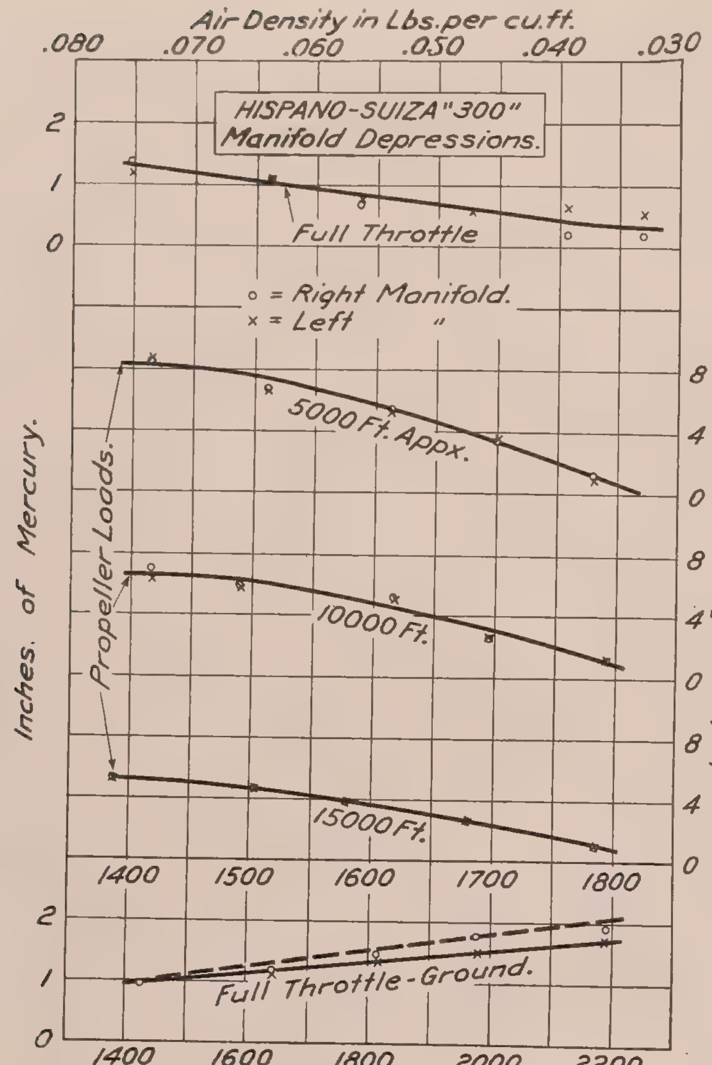

Fig. 14.

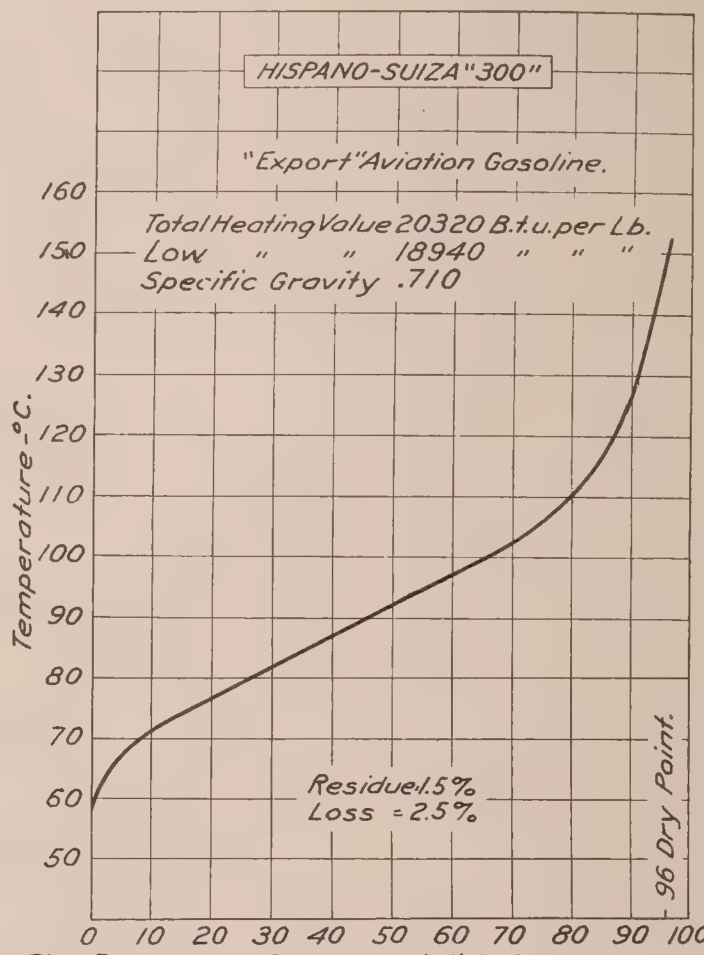

Fig. 15.

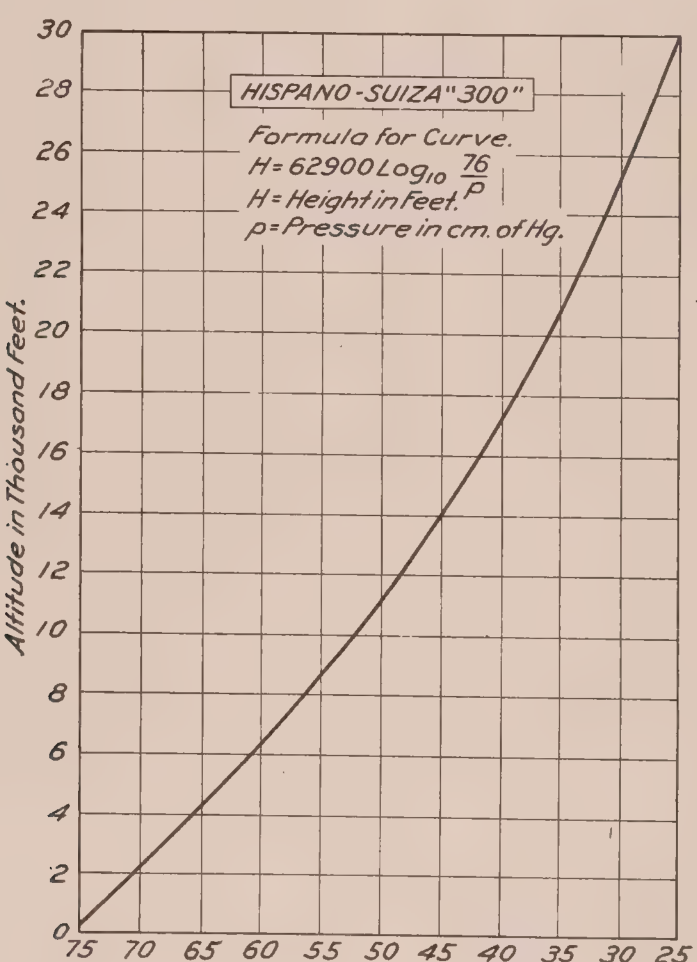

Fig. 16.

REPORT No. 104

TORSION OF WING TRUSSES AT DIVING SPEEDS

By ROY G. MILLER
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REPORT No. 104.

TORSION OF WING TRUSSES AT DIVING SPEEDS.

By Roy G. Miller.

Langley Memorial Aeronautical Laboratory.

INTRODUCTION.

As there seems to be no apparent agreement as to the methods to be pursued in making the analysis of the stresses in a wing truss in a vertical dive at limiting velocity, the following report was prepared at the Langley Memorial Aeronatutical Laboratory of the National Advisory Committee for Aeronautics:

The most easily applied, but least accurate, assumption considers the drag load of the wing structure uniformly distributed along the span with no regard to the unstable moment imposed on the aerofoil by the air load. This is the method called for by the United States Navy specifications.

A second method as applied to the conventionally constructed biplane takes into consideration the unstable moment by combining with the drag load an upload on the rear lift truss and a down load on the front antilift truss, but disregards the stagger wires which tend to equalize the stresses. This is the loading called for by the British, but it is hardly more accurate than the first type, being far more severe than the actual conditions. This method seems particularly inconsistent where there are two bays, as it places dependence upon single antilift wires, but not upon either of the two sets of stagger wires. It is very desirable to so design a structure that it will still be safe with any one redundant member removed, but in such a case two load factors—one for the complete structure and a lower one for the crippled structure—should be specified and a stress analysis should be made with each redundancy in turn removed.

A third method considers the same loading as the second mentioned above, but resorts to the method of least work [1] for determining the stresses in redundant members. If there are several redundant members the method of least work may also be used for analysis with each single member in turn removed.

A final refinement is the correction of the load distribution for distortion of the truss and the consequent warping of the wings. In the case of a nose dive the warping of the wings changes the load distribution in such a way as to increase the stresses. In view of this fact it is evident that the structure should be as rigid as possible.

It does not seem that the designer would be justified in using either of the first two methods, except for an approximate analysis in the course of design to be checked by more exact method later. The first method grossly underestimates the stresses, while the second method is far too severe.

The third method seems to be sufficiently accurate for most designs, even for an exact final analysis.

It is the purpose of this investigation to analyze a typical wing structure by the fourth method and to draw conclusions as to what types of design require allowance for torsion of the wing truss and a change of the angle of attack along the span. At the angle of maximum lift the structural deflections are such that no serious warping of the wings would occur. There

[1] The Analysis of Wing Truss Stresses, including the Effect of Redundancies, by E. P. Warner and R. G. Miller. Report No. 92, National Advisory Committee for Aeronautics, Washington, 1920.

is an upload on both trusses, and the angular distortion at any point is proportional to the difference between the deflections of the front and rear trusses instead of the sum of the deflections as is the case in a dive. Any change in the angle of attack at high angles would make a very small change in the loading because the slope of the lift curve is small in the region of maximum lift. The percentage variation in loading is further reduced by the fact that the net change is divided by a high lift. This condition of loading has been approximated in a great many static load tests which resulted in no great angular distortions. In view of these facts it was not considered desirable to investigate angular distortion for high angles of attack.

PRELIMINARY ASSUMPTIONS.

For the purpose of an illustrative example, the RAF–15 aerofoil was chosen as being the basis of most wing sections used at the present time. A biplane wing structure with overhang was considered as the best example from which to draw conclusions. The warp of the overhanging portion of the upper wing would show approximately what could be expected of a monoplane structure. The JN–4 wing truss layout was selected as an example of this construction.

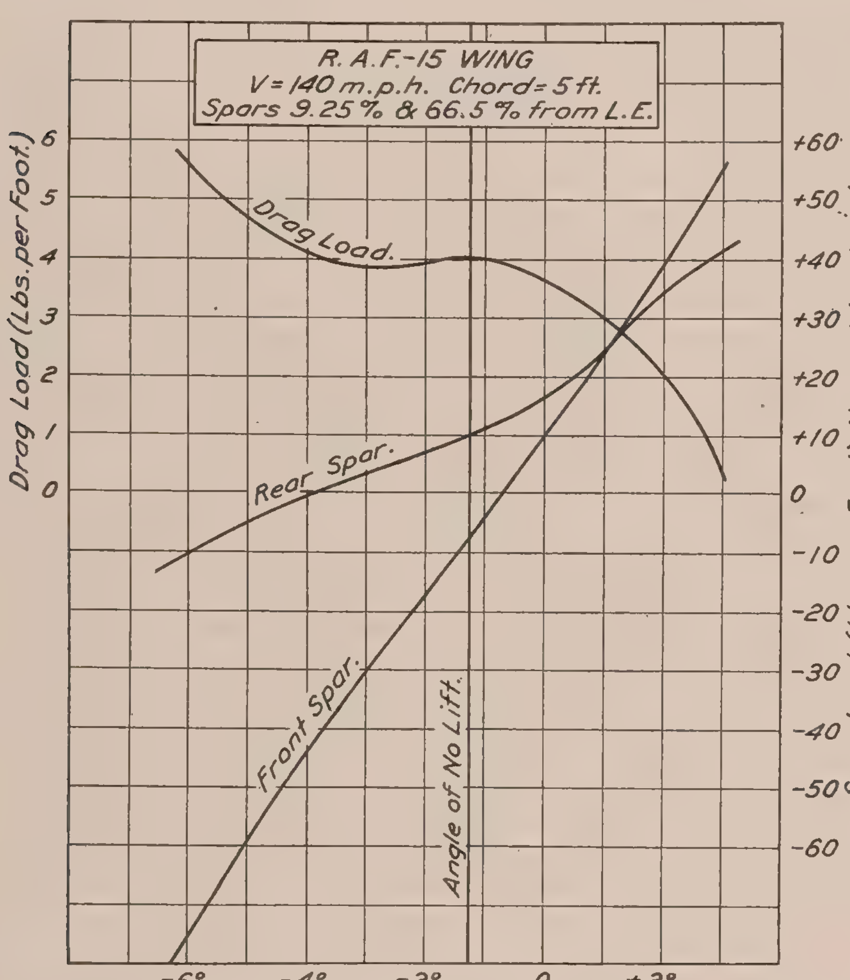

Fig. 1.

LOADING.

The resultant air load on the RAF–15 wing was resolved into components parallel and normal to a 5-foot chord for angles of attack from −6° to +3° at a speed of 140 miles per hour, which is roughly the limiting speed of vertical dive for the JN–4. The component parallel to the chord was plotted (Fig. 1) as pounds per foot of wing span against angle of attack. The component normal to the chord was divided into pounds per foot for the front and rear spars, respectively, and plotted against angle of attack. This is very similar to curves plotted by the British[2] for the RAF–6 wing, the principal difference being that the British assumed the condition of steady flight instead of the simpler and more severe assumption of constant speed. The spars were taken at 9.25 and 66.5 per cent of the chord, respectively, from the leading edge. The characteristics for the RAF–15 wing were taken in accordance with tests[3] made at the National Physical Laboratory, England.

It will be noticed that the curves representing the loads on the front and rear spars normal to the chord (Fig. 1) rapidly diverge for negative angles of attack. This would indicate that the condition for producing a maximum warping of the wings would be in "diving under" beyond the vertical at terminal velocity, but this maneuver would put a top loading on the wings far in excess of what practice has shown to be probable, especially as there is added to the negative load on the wings a negative load on the tail, which may be equal to one-quarter of the weight of the airplane. In nosing over suddenly from high speed or at the top of a slow loop, reverse loading often occurs, but at speeds much lower than terminal velocity. A nose dive at

[2] Handbook of Strength Calculations, by Pippard and Pritchard, p. 6. Ministry of Munitions, Technical Department, Aircraft Production, 1918. Also, C. I. M. No. 34, by Miss Cave-Brown-Cave, Technical Department, Air Board, June, 1917.
[3] Advisory Committee for Aeronautics. Report No. T.709, May, 1916.

terminal velocity and an angle of attack of $-0.7°$ (there being no normal load on the front spar at this angle the calculations are simplified by this assumption) were chosen as a basis for this investigation.

The air load for the first approximation was assumed to be uniform over the entire wing span and corresponding to an angle of attack of $-0.7°$. The reactions at panel points were determined by use of the three-moment equation on the assumption that each spar is a con-

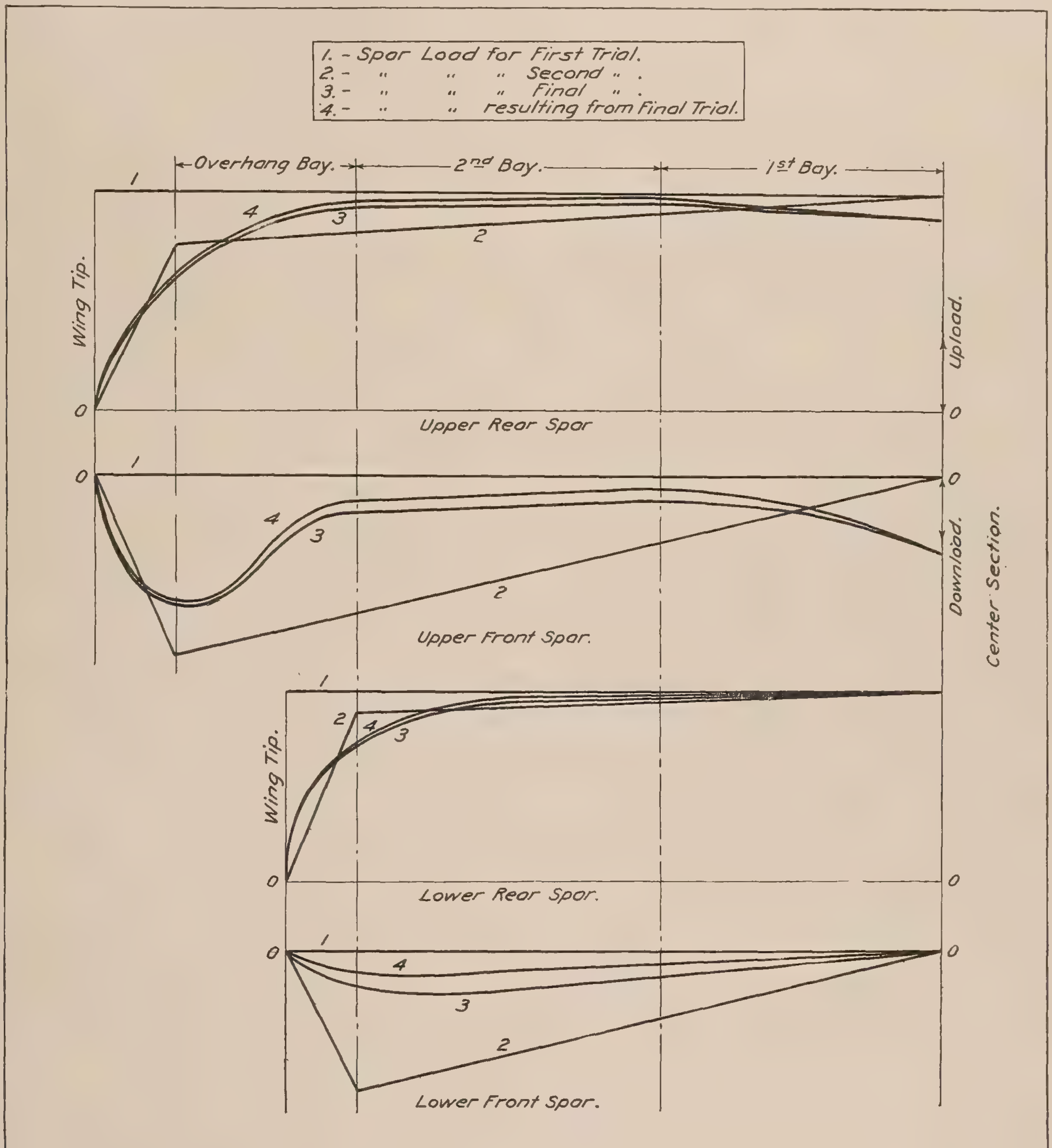

Fig. 2.

tinuous beam, uniformly loaded, and with points of support in a straight line. The loads parallel and normal to the chord were treated separately. The reactions normal to the chord were each resolved into two components—one parallel to the chord and one in the plane of the lift truss. The parasite resistance was estimated and added at the panel points of the drag truss.

SUCCESSIVE APPROXIMATIONS.

The truss was first solved for the case of uniform loading by the method of least work. The stress and strain were computed for each member. A Williot diagram[4] was drawn for each of

[4] The Theory of Structures: Spofford, New York, 1915, p. 368.

the lift trusses to determine the deflection of each panel point under load. If each stagger bay were a perfect parallelogram the angle of attack would not be affected by the deflection of the drag trusses, but the front interplane struts are slightly shorter than the rear ones due to the greater depth of the front spar. The error introduced by neglecting the deflections of the drag trusses is even smaller than the error involved by the use of the Williot diagram. For this reason the diagrams were not drawn for the drag trusses.

The algebraic difference between the deflections of corresponding panel points in the front and rear lift trusses divided by the normal distance between interplane struts is approximately equal to the tangent of the angle of distortion. Substracting this angle from the angle of attack originally assumed determined the corrected angle of attack for that point. The corrected loading for the new angle thus obtained was then read from the curve shown in Fig. 1.

The second approximation was carried through with the angle of attack at the last panel point equated to that determined by the first approximation, and the load distribution curve was assumed to be a broken line dropping to zero at the wing tip and varying uniformly over the span up to the last panel point. (See Fig. 2.)

The loading for the third approximation was plotted at the panel points along the span in accordance with the variation in angle of attack as determined by the second approximation. A load line for each spar was faired in connecting these points except near the tip, where the

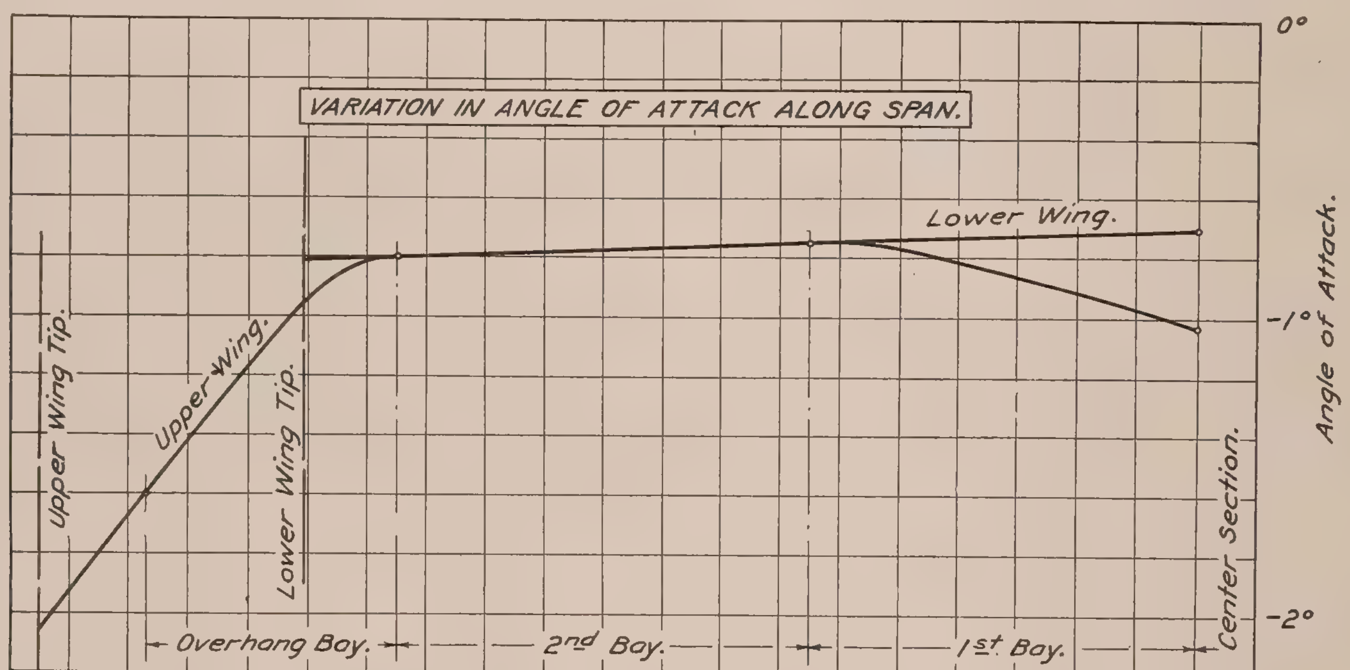

Fig. 3.

load was assumed to drop to zero along a parabolic curve, which broke away from the faired-in curve at a point one chord length from the wing tip. The panel point reactions were solved by Wilson's method [5] of treating continuous beams, the deviation of panel points from collinearity being determined from the deflection diagrams for the second approximation and being allowed for in computing the third set of reactions.

RESULTS.

The forms of the load curves used for the successive approximations, together with the load curves as determined by the final approximation, are shown by Fig. 2. The scale of the curves for the first two approximations is purposely exaggerated to indicate the forms of the curves rather than the actual magnitude of the loading. An inspection of these curves indicates that the second approximation might well be omitted. The uniform variation in load along the span was used to simplify the treatment of stresses for this case, but it is probable that the same accuracy could be obtained in the final approximation by omitting this step entirely and taking only two successive loadings instead of three.

The variation in angle of attack along the span is shown graphically for the upper and lower wing by Fig. 3.

[5] Strength of materials: Morley, London, 1916, p. 218.

Fig. 4 shows the load per foot for each spar as determined by the final approximation. The broken lines indicate what the loading at the tips would be if there were no end losses. It is evident by an inspection of Fig. 4 that there is considerable net lift on the wings. This lift would be largely balanced by the down load on the tail plane required to balance the unstable wing moment.

The maximum variation in angle of attack due to bending of the spars between panel points was found by a preliminary analysis to be less than eight minutes. This variation being negligibly small it was considered best to simplify the treatment of the structural deflection by neglecting the bending of spars.

Attention is particularly directed to the fact that the warping of the wings under diving conditions is too small to be considered for practical design wherever there is adequate stagger bracing, being larger at the center section of this particular machine, where the alignment is maintained only by the external drag and antidrag wires, and being quite great in the overhang where there is no incidence bracing.

If there were high initial tensions in the wires of the wing trusses, the effect would be similar to a great increase in the number of redundancies, and thus the deflections and con-

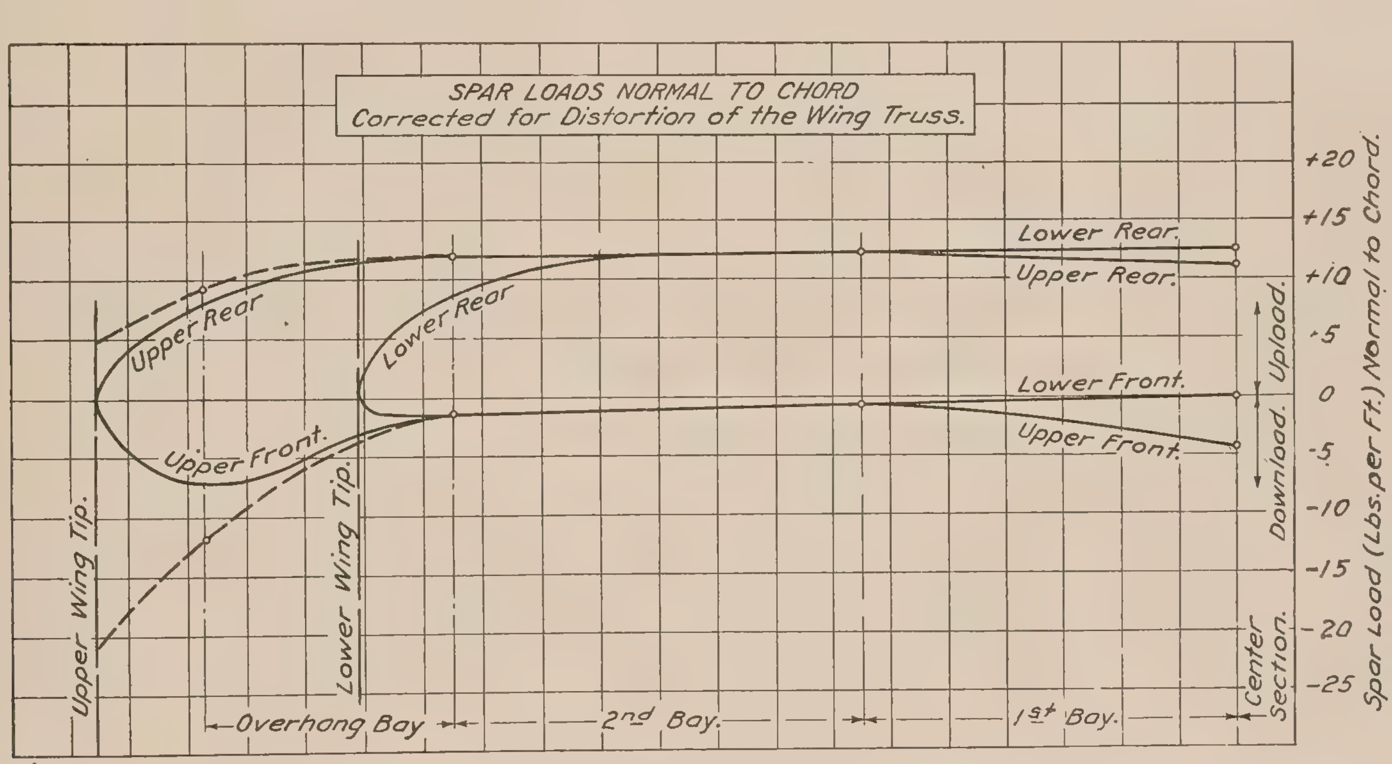

Fig. 4.

sequently the warp of the wings would be reduced. In particular, the initial tensions in the stagger wires are nearly always great enough to keep both wires tight under all conditions, and the warp of the wings inside the outermost panel point are thereby reduced approximately 30 per cent. It must not be considered, however, that this is an argument in favor of high initial tensions, the more favorable load distribution being more than counterbalanced by the higher stresses in individual members.[6]

CONCLUSIONS.

1. In the case of the conventional biplane with adequate stagger bracing and no overhang, it is impractical to refine the stress analysis to the extent of correcting the load distribution for warping of the wings under load. The wing drag should be considered uniform and carried to the wing tip. The loads normal to the chord should be considered uniform and carried to a point one-sixth of a chord length from the wing tip.

2. For the biplane with an overhang supported by steel struts which are capable of withstanding either tension or compression, it should be sufficiently accurate to neglect the effect of the twisting of the wing truss; but where the down load on the overhang is supported by wires

[6] See Analysis of Wing Truss Stresses, Including the Effect of Redundancies, by E. P. Warner and R. G. Miller. Report No. 92, National Advisory Committee for Aeronautics, Washington, 1920.

at a very acute angle to the spars the loads normal to the chord should be carried to the extreme wing tip to allow for the effect of distortion upon the load distribution.

3. The wing warp is a very important consideration in treating monoplane stresses. The monoplane wing truss lacks the efficient stagger bracing of the biplane, and the members supporting the wing are generally long and at a very acute angle to the spars. It seems reasonable to believe that some of the accidents which have occurred as a result of diving monoplanes were due to the unstable nature of structural distortions. The results of static tests may make it appear that a monoplane is safe for both upload and down load, and still the wing structure may be unable to withstand the loads due to the torsion produced by the air load in a nose dive. The members most effected by the torsion of a monoplane wing are the antilift wires attached to the front spars near the wing tips. If the torsion at the tip of the overhanging wing of a biplane can be as high as 2°, as indicated by Fig. 3, then the magnitude of the torsional deflection at the tip of a monoplane would probably exceed 2°. Fig. 1 indicates that a change of 2° in the angle of attack would multiply the down load on the front spar by 4. The only safe course to follow in the design of a monoplane is to make an exhaustive stress analysis, taking into consideration the effect of structural distortions upon the load distribution.

4. Owing to the relatively high deflection of the internally braced wing the load distribution should be corrected for the variation in angle of attack along the span when loaded. In the case of a biplane of this type stagger bracing may well be used near the wing tips, as in the case of the Fokker. The Germans have made performance tests with the stagger bracing omitted and found that both the speed and climb of the Fokker were reduced. There is nothing to indicate how much the factor of safety was reduced by this omission, but it is obvious that deflections of the magnitude required to injure the performance more than it is helped by cutting out the parasite resistance of the struts would certainly greatly change the load distribution. In the case of the internally braced monoplane it is obviously impossible to use anything corresponding to stagger bracing, but if a relatively strong and stiff front spar is used it will do much to stabilize the structure because the load on the front spar changes more rapidly and has higher maxima than the load on the rear spar.

5. It must not be inferred from this discussion that an exact stress analysis for the case of a dive constitutes a complete stress analysis. The nose dive at terminal velocity is included as a part of the complete analysis because it generally imposes the most severe stresses in particular members, namely, drag bracing, stagger bracing, and sometimes the front antilift bracing; but other members are most stressed at other conditions of flight, which must be just as carefully considered.

REPORT No. 105

ANGLES OF ATTACK AND AIR SPEEDS DURING MANEUVERS

By E. P. WARNER and F. H. NORTON
Langley Memorial Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Field, Va.

REPORT No. 105.

ANGLES OF ATTACK AND AIR SPEEDS DURING MANEUVERS.

By Edward P. Warner and F. H. Norton, Langley Memorial Aeronautical Laboratory.

INTRODUCTION.

The following report was prepared at the Langley Memorial Aeronautical Laboratory of the National Advisory Committee for Aeronautics, as it seemed desirable that there should be some study of the attitude assumed by an airplane, and more particularly of its motion with respect to surrounding air when maneuvering, either in ordinary turns, spirals, climbs, and dives, or in those more spectacular feats commonly known as stunts. It is important to secure this information, among other reasons, in order to have definite knowledge as to the distribution of load on the wings, and so to furnish the basis for improved accuracy in stress analysis. An accelerometer can be counted on to give the total load on the wings with great accuracy, but it tells nothing about the distribution of that load along the chord or between the upper and lower wings or about its partition between the front and rear trusses. Knowledge as to these factors can only be gained from measurements of the angle of attack.

The second reason for wishing data on behavior in maneuvers is aerodynamic. If airplanes are to be designed intelligently it is essential that the designer know what they will have to do. If a machine is required to loop easily and rapidly, or to resist falling into spins, or to show any other particular maneuverability characteristic, a necessary preliminary is the securing of information as to the reactions of the air on the machine and the manner in which they should be modified to gain the desired end.

Experiment in this direction has been exceedingly sparse compared with that in other lines. Some work has been done at the Royal Aircraft Establishment with a recording air-speed meter, as well as by observing with a camera obscura and by taking moving pictures from another airplane, the second method being used particularly for spins and spirals, the third for rolls. All of these experiments, with the exception of those on air speeds which were performed in conjunction with some accelerometer tests, were directed primarily toward the determination of attitudes and motions in space rather than with relation to the surrounding air, and this, while interesting, is of little immediate application if taken alone. To make a complete dynamic analysis it is of course necessary that both the angles of attack and the angles of inclination be known; but the angles of attack and accelerations are more important to the designer than are the positions in space. The angles of attack and side slip could obviously be computed if the attitude and component velocities were completely known and if the air were still, but the measurements would have to be made with greater refinement, to give a satisfactory degree of relative accuracy in the determinations of the angles to the relative wind, than is readily attainable by the methods hitherto used.

Some experiments on the direct measurement of angles of attack during maneuvers have accordingly been carried out at Langley Field.

METHODS OF TESTING.

The angle of attack was measured with a simple vane, counterweighted by a rod projecting forward from the pivot. The vane was exceedingly steady, and the angle could easily be read to within a degree under any normal conditions when the vanes were well away from the center

[1] Applied Aerodynamics, by Leonard Bairstow, New York, 1920. Ch. V.

of the machine. An attempt to measure the direction of flow in the slip stream, however, was less successful, the vane fluttering badly because of the more turbulent nature of the relative flow.

The direct measurement of angle of attack during maneuvers is subject to two errors, and the means which reduce one error unfortunately aggravate the other. The most obvious of these two sources of difficulty is the interference of the wings and the disturbance of the air to considerable distances in front of the machine. If there were no complicating factors this interference could be reduced to a point where it would become negligible by carrying the vane on a long pole projecting one and a half or two chord lengths forward of the leading edge, as is the practice when the angle of attack is to be measured during steady straight flight. This is impracticable when the flight path is curvilinear, because the rotation causes different parts of the airplane to move in different directions at the same instant, and a vane carried well forward of the wings would not travel through the air in the same direction as do the wings themselves. In a tight loop the rotation is rapid enough so that moving the vane forward 3 feet would cause an error of approximately 1.5° in the vane reading, entirely aside from any interference effects. Because of these inevitable errors, the measurements can not be relied on for great accuracy; but the two sources of trouble fortunately tend to cancel each other in most instances, the interference making the reading too high, while the rotation in the sense usual in maneuvers (stalling) makes it too low, and the total resultant error probably does not exceed 2° in any instance except at very large angles of attack.

The air speed was measured with a meter of the pressure-plate type in order to bring the dial close to the vane and to facilitate the photographic recording of angle of attack and air speed on a single film. The vane and air-speed meter are shown together in figure 1. The whole instrument is pivoted and has a counterweighted vane behind the pivot to keep the meter always in the same position with respect to the relative wind. The meter is pivoted in the vertical plane only, no attempt having been made to allow swiveling about a vertical axis when side slipping. The member on which the air pressure acts is an aluminum disk 2 inches in diameter (seen edge-on in the lower right-hand corner of the cut). This disk is rigidly attached to one member of a jointed parallelogram linkage. The side opposite to that which bears the disk is pivotally fixed to the airplane, and an extension of one of the other sides carries a pointer moving over a scale. The rearward motion of the disk is opposed by the pull of a rubber band which connects extensions of two adjacent sides of the parallelogram. The air-speed meters were calibrated in flight by flying the airplane steadily at various speeds and observing the position of the pointer. At speeds below the minimum for steady flight the meters were calibrated in a wind tunnel.

Maneuvers may be divided into four classes—steady symmetrical, steady asymmetrical, unsteady symmetrical, and unsteady asymmetrical. The first class includes only rectilinear flight, both horizontal and inclined. The second group includes turns on the levels, spirals, and some spins. The extent to which spinning is a steady motion is a matter still open for argument, but it is certain that there are some airplanes in which it is a periodic motion of substantially unvarying amplitude. The class of symmetrical unsteady motions is a very large one, taking in loops, zooms, ordinary longitudinal oscillations, and pulling out of dives. Finally, the fourth classification includes rolls and reverse, or Immelman, turns.

These four classes are arranged approximately in order of the difficulty which they present to the experimenter. The steady motions are easy to study, because no recording instruments are necessary, the readings being taken and recorded by the observer. This method was used in the first work done on free-flight testing by the National Advisory Committee for Aeronautics at Langley Field.[2] The symmetrical motions are easier than the unsymmetrical to deal with, because in a symmetrical motion all points along a line perpendicular to the plane of symmetry have the same speed and direction of motion, and it is therefore only necessary, in general, to take readings at one point, whereas in unsymmetrical maneuvers at least two points and sometimes more must be used.

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[2] Preliminary Report on Free-Flight Testing, by Edward P. Warner and F. H. Norton; N. A. C. A. Report No. 70, Washington, 1920.

Fig. 1.—The Vane and Air Speed Meter.

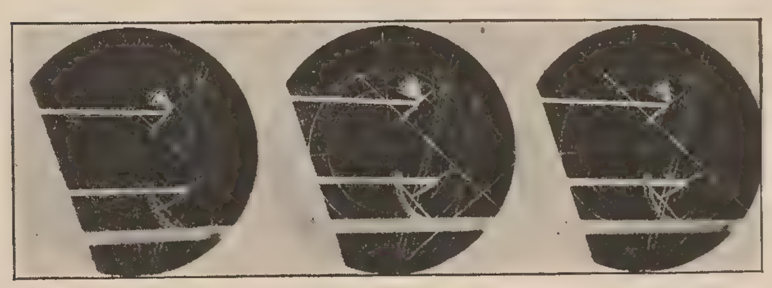

Fig. 2.—Photographs of Vane and Air Speed Meter taken with Gem Camera.

Fig. 3.—The JN 4H Test Airplane.

The method used in the present series of tests was to take moving pictures of the vane and air-speed meter, which were fastened about 18 inches apart, with a gun camera. A few exposures from one of the records are shown in figure 2. Only one wing tip at a time could be worked on in this way, and unsymmetrical maneuvers had to be repeated in order to secure complete records. The gun camera is very unsatisfactory for this work, as it does not run as long on a single winding as would be desirable, and as its speed is not sufficiently constant to furnish a good time scale. The governor is of rather crude construction, although amply good enough for gunnery practice, and the rate of taking pictures is considerably affected by accelerations of the air plane. Furthermore, many records were spoiled by jamming of the mechanism.

RESULTS OF TESTS.

The first experiments were made in steady straight flight and with direct reading in order to see how serious was the error due to interference between the wings and the vane. A JN4H airplane was used in these and in all subsequent tests, and is shown, as fitted up for the tests, in figure 3. The vane readings were compared with the values of the angle of attack determined for the same airplane in Report No. 70 by the use of a liquid inclinometer when flying level, and it was found that interference increased the vane reading by from 1° to $1\frac{1}{2}$° at all air speeds from 70 miles per hour to 90 miles per hour. At speeds lower than these the interference is more marked, increasing to 4° at 50 miles per hour. It is not probable that the error due to interference goes on increasing rapidly as the angle of attack increases beyond the burble point, since the degree of upward diversion of the air forward of the wings is dependent on the lift coefficient.

The next type of evolution tested was an ordinary spiral. During a tight spiral at 80 miles per hour the angle of attack rose to 7°, although the angle for equilibrium in straight flight at this speed is only 0.8°. Allowing for an interference error of 1.5° in the angle when spiraling, and taking lift coefficients determined in free-flight tests of the JN4H, it appears that the lift coefficient at 5.5° is 1.84 times as large as that at 0.8°. Neglecting the effect of descent in spiraling, the theoretical angle of bank corresponding to this load factor would be 57°. No apparatus for making accurate measurements of banking angle was available, but it was evident from direct inspection that the angle of bank was approximately 60° (within 10° plus or minus). The experimental determination of the angle of attack, therefore, checks well with the computed value in this simple case.

The first real stunt to be considered was the spin. It can easily be shown that the angles of attack at the wing tips during any unsymmetrical maneuver are:

$$\tan \alpha_R = \frac{-w+\frac{s}{2}\cdot p}{-u-\frac{s}{2}\cdot r}$$

for the right wing, and

$$\tan \alpha_L = \frac{-w-\frac{s}{2}\cdot p}{-u+\frac{s}{2}\cdot r}$$

for the left, where u and w are the components of velocity parallel to the X and Z axes, respectively, and p and r are the angular velocities about those axes, as is customary in stability work, and $\frac{s}{2}$ is the distance from the plane of symmetry, the X axis being taken parallel

to the wing chord. Similarly, the components of velocity parallel to the plane of symmetry are:

$$V_R = \sqrt{\left(u+\frac{s}{2}\cdot r\right)^2 + \left(w-\frac{s}{2}\cdot p\right)^2}$$

and

$$V_L = \sqrt{\left(u-\frac{s}{2}\cdot r\right)^2 + \left(w+\frac{s}{2}\cdot p\right)^2}$$

if v, the velocity of side slip, be neglected.

The determination of α_R, α_L, V_R, and V_L, therefore, makes possible the calculation of μ, w, p, and r. The solutions for these four velocities are:

$$u = -\left(\frac{V_R \cos\alpha_R + V_L \cos\alpha_L}{2}\right)$$

$$w = -\left(\frac{V_R \sin\alpha_R + V_L \sin\alpha_L}{2}\right)$$

$$p = \frac{V_R \sin\alpha_R - V_L \sin\alpha_L}{s}$$

$$r = \frac{V_L \cos\alpha_L - V_R \cos\alpha_R}{s}$$

Direct observation of the angle of attack after the attainment of steady conditions showed a great difference among spins. The angle of attack at the inner wing varied from 35° in the slowest true spins to 75° in the most rapid maneuvers. The angle on the outer wing varies much less, ranging only from 7° to 10° in the tests made.

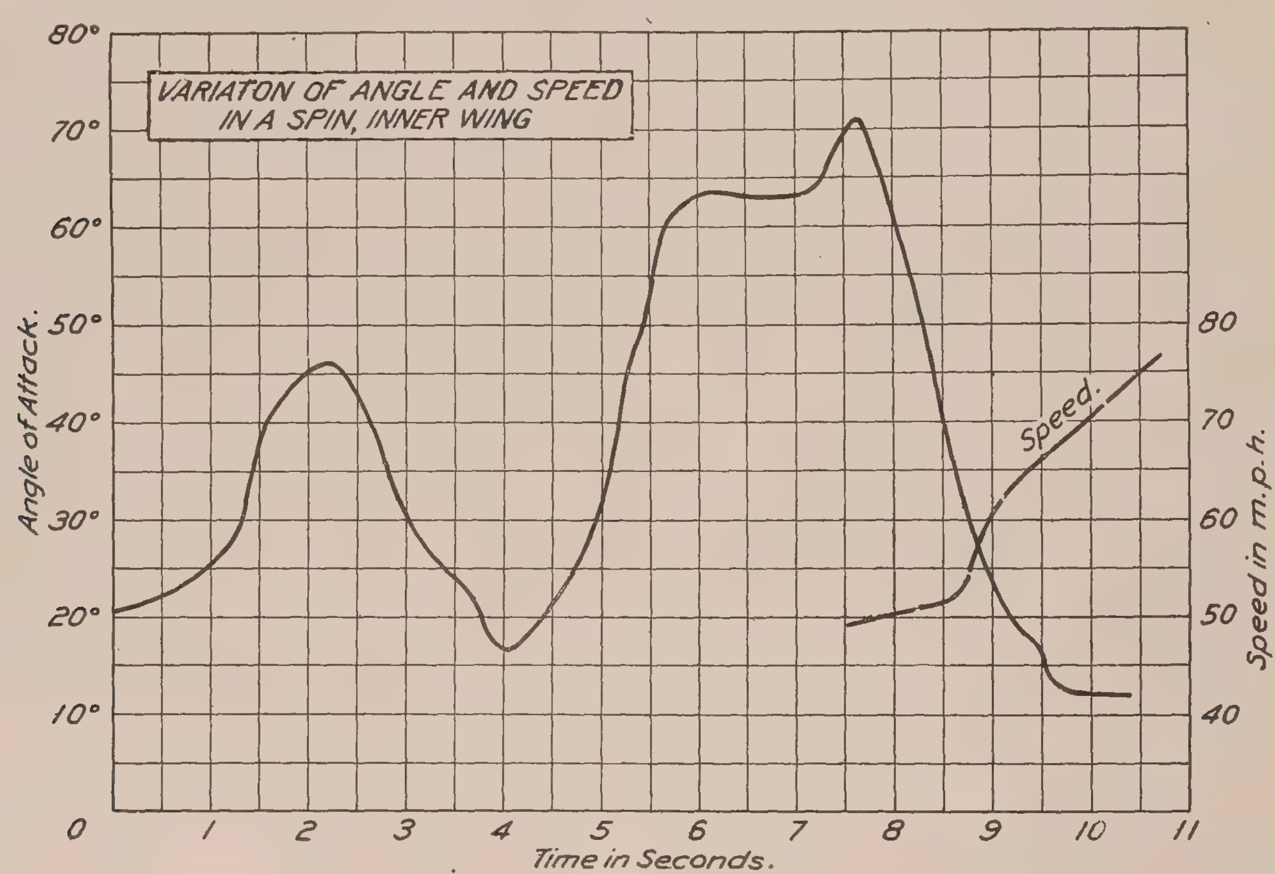

FIG. 4.

In using the gun camera the records were started early enough to catch the beginning of the maneuver from the instant when the pilot pulled his stick back to stall the machine. The variation of angle of attack on the inner wing tip during a spin and of speed during the recoveries are plotted in figure 4. Complete speed records could not be plotted because the speed during the spin was below the lower end of the scale (48 miles per hour). Both of these spins were executed with the stick clear back and with the controls crossed (left rudder, right aileron). It will be noted that the spin starts with a large oscillation as the rudder is put over, and that

this oscillation is very quickly damped out, disappearing after a single swing. As the controls are centralized the angle increases for an instant and then falls off rapidly. Accelerometer records on this same airplane show a continuous oscillation of small amplitude throughout the duration of the spin, but this oscillation could not be observed on the vanes. In spins of long duration (too long to be followed throughout with the gun camera) the vanes reached a perfectly steady reading, varying less than one degree therefrom.

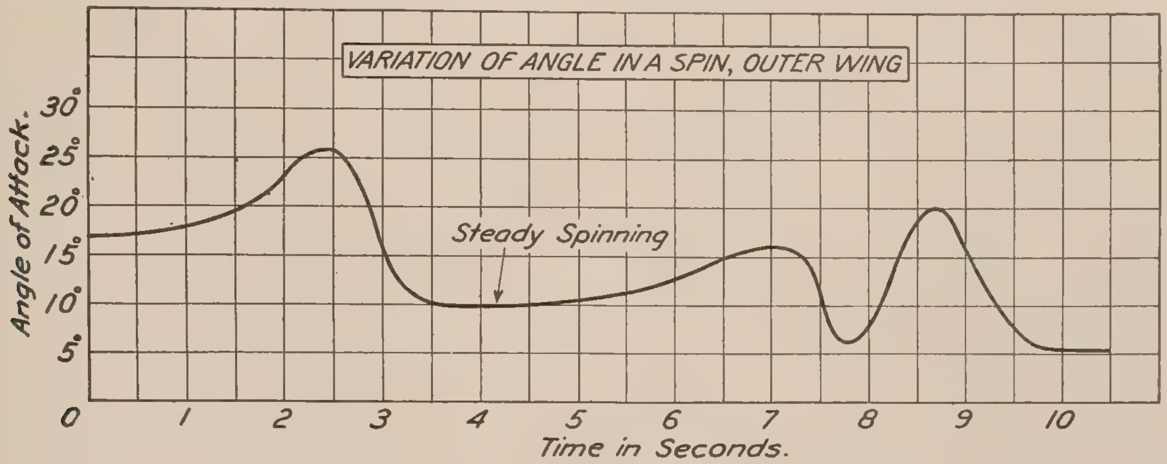

FIG. 5.

The variation of angle on the outside wing is plotted in figure 5. Here, too, there is an oscillation, the angle of attack increasing as the machine starts to fall into the spin and then falling off as the rate of rotation approaches its steady value. The speed during steady spinning was about 60 miles per hour, the descent during these spins being very slow while the rotation was rapid.

A very approximate solution for the velocities, taking V_R as 88 feet per second, V_L as 66, α_R as 8°, and α_L as 61°, due allowance having been made for interference, gives:

$$u = -59.6 \text{ feet per second.}$$
$$w = -35.0 \text{ feet per second.}$$
$$p = -1.56 \text{ radians per second.}$$
$$r = -1.90 \text{ radians per second.}$$

The angle of attack at the plane of symmetry is 30.4°, which is of the same order of magnitude as the angles of attack found during spins by British experimenters [3] using an entirely different method.

It is evident that, if spinning is a steady motion, there must be no unbalanced forces or moments, the air reactions on the airplane being completely expressed by a single vector passing through the center of gravity. The resultant of all the components of force parallel to any given line must then pass through the C. G. The particular line which is of most interest in this connection is the Z axis of the airplane, which has been taken as perpendicular to the plane of the wings. The coefficient of force normal to the chord of an aerofoil reaches a maximum at the burble point, drops off slightly thereafter to a minimum at about 25°, and then increases again until an angle of attack of 90° is reached. The curve in figure 6 represents the variation of normal force with angle of attack for the wing used on the JN4H, although a part of the curve was lifted from a wind tunnel test of a slightly different aerofoil, the tests of the JN wing itself having extended only to 20°. It is probable, however, that all thin aerofoil sections of the same general form have closely similar characteristics at angles beyond that of maximum lift.

To find the condition necessary in order that the resultant of the normal components may lie in the plane of symmetry it may be assumed that each aerofoil element formed by two planes parallel to the plane of symmetry acts independently of every other such element, and that the forces and moments acting on the wing may be found by summing the elementary forces and moments arising on each element. This method has been found to give good results

[3] A Mathematical Study of Spinning, by Lindemann, Glauert, and Harris: R. and M. No. 411, British Advisory Committee for Aeronautics.

in propeller design, and its application to wings has been justified in numerous experiments on the auto-rotation of aerofoils in the wind tunnel and on warped aerofoils, carried out at the National Physical Laboratory. It furnishes a very powerful means of analyzing the unsymmetrical motions of airplanes.

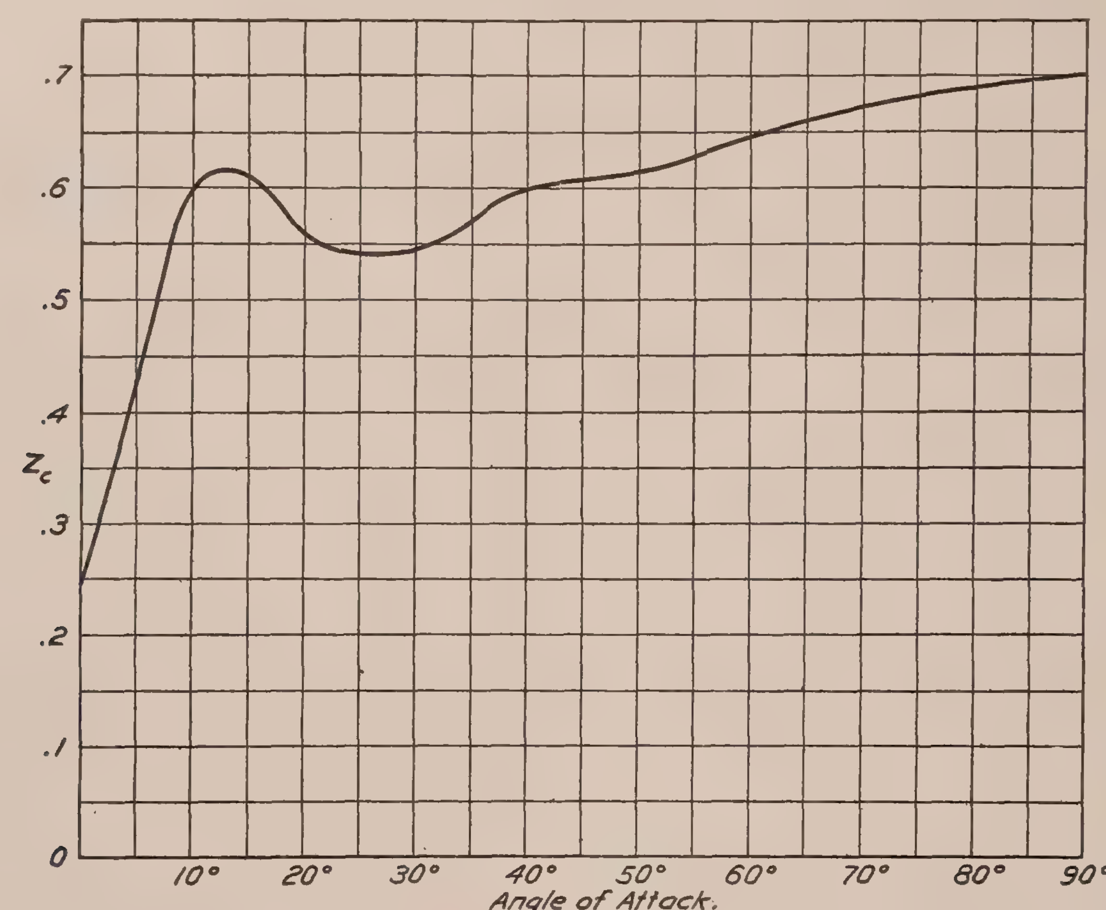

Fig. 6.

The condition of equilibrium, if the aerofoil element theory be used, is that

$$\int_{-\frac{s}{2}}^{+\frac{s}{2}} Z_c \cdot c \cdot [(u+rx)^2+(w-px)^2] \cdot x \cdot dx$$

must be equal to zero, where s is the total span, c the chord, x the distance of an element from the plane of symmetry, and Z_c the coefficient of normal force at the angle

$$\alpha = \tan^{-1}\frac{-w+px}{-u-rx}$$

Z_c over the inner wing, where the angle of attack varies roughly from 35° to 65°, is nearly constant, while the coefficient on the outer wing increases from the plane of symmetry, reaching a maximum about halfway out, the wing there meeting the air at the angle of maximum lift, and then falls off again as the tip is approached. If the angles of attack at the outer strut on a JN are 8° and 61° the angles of attack at the extreme tips of the upper wings, which overhang the outer strut points by about 7 feet, are 0.7° and 75°. There, therefore, is no change in the direction of the load at the outer tip, as the angle of attack does not pass the angle of zero lift. It is very probable, however, that it would do so in some instances, as the spins from which these data have been taken were not extreme ones. In some spins the vane on the inner wing continuously recorded an angle of from 77° to 79°, corresponding to an angle of approximately 90° at the wing tip.

In figure 7 there are plotted curves of V, α, Z_c, and $Z_c V^2$, or mean loading per square foot, against x, the values taken for u, w, p, and r being those already given as computed from the observations. The total moment about the line corresponding to the plane of symmetry ($x=0$) of the area under the third of these curves should be zero if the experimental results and the assumptions made in the computation were correct. The agreement proves to be

rather poor, the center of action of the normal force, as found from the curve of loading in figure 7, lying on the outer wing and 1.25 feet from the plane of symmetry, but it is good enough to make sure that the load curve in figure 7 represents with reasonable accuracy the form of the curve of load distribution along the span during a spin. The maximum loading occurs near the middle of the outer bay, and is 38 per cent larger than the mean loading. The load factor given by the curve is 1.25, which is somewhat smaller than the load factor found with an accelerometer in similar spins. It is probable that the principal cause of the discrepancies lies in the fact that the ailerons are not centered, and that the actual coefficient of normal force on the outer portion of the outer wing is therefore less than it was assumed to be in making the computations.

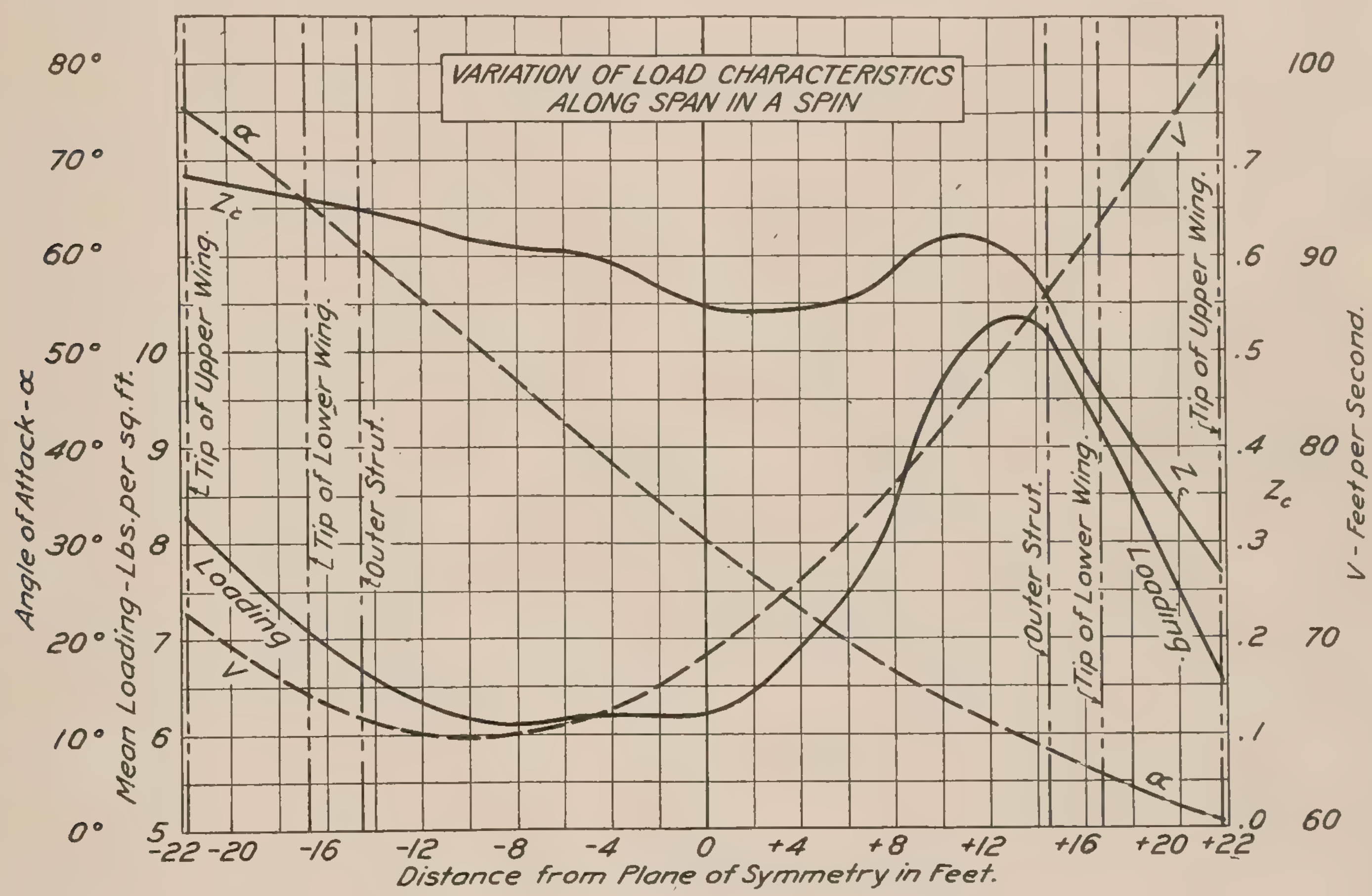

Fig. 7.

It will be noted in figure 7 that the resultant speed does not by any means vary uniformly along the wing, but falls off to a minimum just beyond the inner strut on the inner wing and then increases again toward the tips.

From the structural standpoint the spin presents no very special terrors, and need not be taken into account at all as a controlling factor in an airplane built for general stunting. It is, however, of special interest because it is a maneuver which almost any airplane is liable to have to execute on some occasion. No pilot loops or does vertical banks without intending to, but there is always a possibility of falling into a spin when turning too flatly or attempting to fly at a very large angle of attack, so large that the ailerons become ineffective.

The most striking features of the spin, structurally speaking, are the uneven distribution of load along the span of the outer wing and the distribution between the upper and lower wings of a biplane combination. The first of these features has already been commented on. The second applies particularly to the inner wing. Near the tip of that wing, where the angle of attack is from 60° to 75°, the center of pressure lies about 45 per cent of the way back on the chord, and it is probable, although no direct experiments are available at such angles, that the lower wing carries about 70 per cent of the load. This concentration of load on the lower wing is, of course, most marked when the upper and lower wings are of equal span.

The load on the drag truss when spinning is negligible. The drag is very large, but is carried through the interplane bracing, the resultant reaction on all elements of the wings except those near the outer wing tip being virtually perpendicular to the chord.

Accelerometer tests have shown a load factor which does not exceed 2.2 during the worst spins, and is materially less than that in most instances. Making all due allowance for the unusual distribution of load, it appears certain that an airplane designed to sustain a load factor of 3.5 with the C. P. forward and a factor of 4 in the lower rear spar with the C. P. at 45 per cent of the chord from the leading edge will be strong enough to stand any kind of a spin.

All along the inner wing the resultant force on the wing is virtually perpendicular to the wing chord, and the same condition prevails over a part of the outer wing. As the angle of attack falls below 20°, however, the vector inclines forward of the perpendicular to the chord and remains so inclined until the angle reaches about 4°. At angles smaller than that the perpendicular is inclined to the rear. This forward inclination of the vector over a part of the outer wing furnishes the force which keeps up the yawing velocity in spite of the damping due to the lateral motion of the vertical tail surfaces. In order to check the spin it is necessary either to centralize the rudder, thus increasing the damping effect of the vertical surfaces over that given by the fin alone when the rudder is trailing ineffectively toward the inside of the path of spin, or, in extreme cases and in unstable machines, to put the rudder over toward the outside, thus furnishing an additional yawing moment independent of the yawing velocity. The rolling and the yawing in a spin act as a check on each other to some extent. If the rolling velocity be supposed to be increased from any cause the angle at the outer wing tip becomes smaller, and the backward angle of inclination becomes larger at the tip and extends over a larger part of the wing. The positive yawing moment due to the wings (assuming the yaw to be in the positive direction) is lessened, and the yawing velocity must therefore be lessened in order that the negative moment due to the tail surfaces may decrease and that the motion may be a steady one. The result is a decreased yawing velocity, a decreased angle of attack, and a steepened path of descent.

A washout of angle toward the wing tips should be beneficial in spinning for two reasons: In the first place, equilibrium in respect of normal forces is secured with a smaller rolling velocity than would exist if all the chords lay in one plane, as the normal force drops off with increased abruptness toward the tip of the outer wing because of the washout, while the force on the other wing, presented to the air at an abnormally large angle, is not affected by the washout at all. The washout, therefore, slows up the spin. Second, the yaw is resisted because the backward force on the outer wing tip is increased by the washout, while the normal force at the inner wing tip, inclined forward of the perpendicular to the plane of the chords near the plane of symmetry, has a forward component. Both of these changes in the forces act together to give a negative yawing moment and to resist the spin.

The yaw in a spin is primarily dynamic, not static, and it is damping of yawing velocity, not "directional stability" in the common sense, that is of concern. To speak in more correct terms N_r, not N_v, is the dominant factor, although it is impossible to frame any estimate of the importance of the latter derivative until measurements of side-slipping velocity in spins have been made. Since damping moment varies as the square of the distance from the C. G., while statical moment varies only as the first power, the fin surface should all be as far away from the C. G. as possible. The flat sides of a deep fuselage may be useful in giving "weathercock stability," but they are of little assistance in damping motions; and it is of interest to note in this connection that the old pushers with open tail structures and all the fin surface at a maximum distance from the C. G. were practically immune from spins even when heavily loaded, and that the first true spin of which there is any record was performed by the late Lieut. Parke, R. N. A. S., on the all-inclosed Avro—a machine with an extremely deep flat-sided fuselage.

What is really wanted to make airplanes which will not spin, as has been pointed out by Dr. Leonard Bairstow and others as a result of experiments on the auto-rotation of stalled aerofoils, is a wind cell for which the normal force has no maximum for any angle less than 90°. Failing such a wing, it appears that spinning dangers can be ameliorated by using washout of angle near the tips and by securing the largest possible negative value of N_r.

LOOPS.

The experiments on loops were carried out in the same way as those on spins, except that only one wing needed to be considered, since a properly executed loop is a symmetrical maneuver The best record obtained is plotted in figure 8, and is typical of the form of all the others.

The most interesting feature of these curves of speed and angle is the light which they shed on the question of "hang loops," and on the reasons for the great difficulty experienced by some pilots in getting around a loop without hesitating on top, in the upside-down position. It will be observed that, as the stick was pulled back, the angle of attack increased rapidly. At the end of two seconds the angle of maximum lift was reached, but the angle continued to increase until, after another one and a half seconds, it attained a value of 24°, at which point the lift is only 80 per cent of that at 12°, and is equal to that at 6°. The effectiveness of the wing in the necessary centripetal force to carry the airplane around the loop was then no greater, at that instant, than it would be if the angle of attack had been reduced by 6°, but the drag, acting to decrease the speed and so to decrease the lift, is 780 per cent greater at 24° than at 6°.

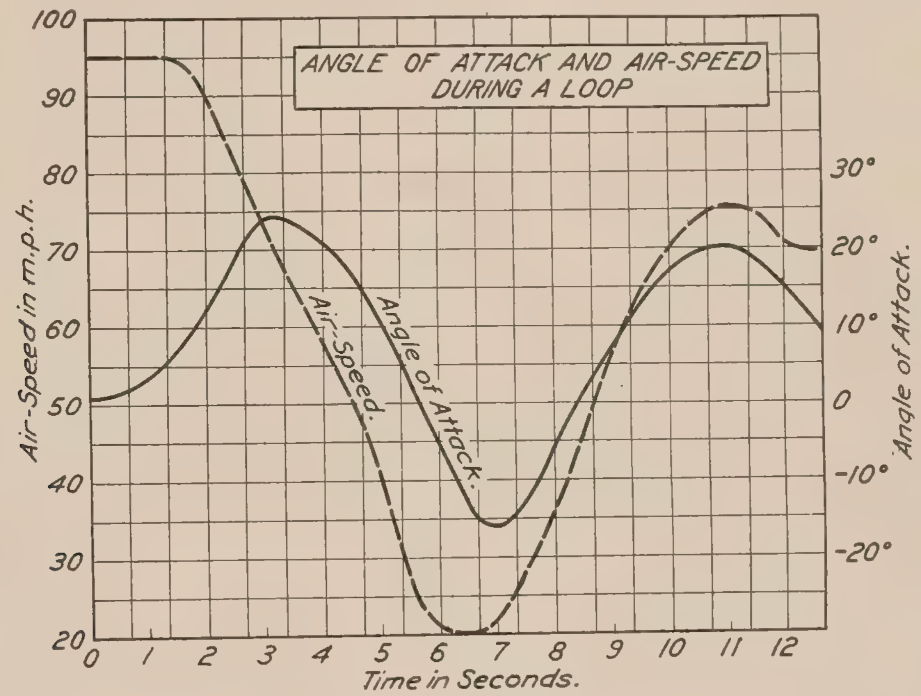

FIG. 8.

After reaching the maximum just mentioned the angle of attack decreased rapidly to a value of $-15°$, the speed at the same time falling to approximately 20 miles per hour (the air-speed meter was modified to extend its range to lower speeds in these tests, but the readings of minimum speed are not very accurate). In other words, the airplane was stalled on its back and was dropping, gaining speed at the same time. The angle then increased to a second maximum of 20° as the machine began to flatten out. It was very evident to the pilot and observer that there was a negative loading on the wings at the top of the loop, and the machine seemed to hang in the inverted position for a perceptible interval before the nose whipped downward. The natural tendency is for a pilot who finds himself in this position to attribute it to having failed to pull the nose up quickly enough to get around the critical portion of the loop before the momentum is exhausted, and his next attempt he will try to pull the stick back more sharply, resulting in the attainment of a still larger angle of attack and a still quicker loss of speed and more rapid dimunition of angle. It appears that the trouble is due entirely to too abrupt a pull up, and that far better results would have been attained if the stick had been eased forward as soon as the nose of the machine was well started upward. There has been no opportunity up to the present time to repeat these experiments on a loop executed more gently.

A comparison of the results of these tests with a computation from model experiment o. the probable path of a similar machine in looping disposes one to pessimism with regard to the possibility of basing an analysis of such maneuvers on tests of models under steady conditions. Although the computation was carried out [4] on the assumption of an instantaneous pulling up

[4] Stresses in Diving and Looping; Bulletin of the Airplane Engineering Department, U. S. A., June, 1918.

of the elevator to the limit of its travel, an abruptness which would never be approached in practice, neither the first maximum value of the angle nor the minimum were as extreme as the angles found on actual test, the first being 17° and the second 6°. The difference between the two results is partly accounted for by the difference in speed at the start of the loop, the computation having been carried through on the assumption that the airplane was diving at 123 m. p. h. when the elevator was pulled up, partly by the fact that the C. G. was farther back in the actual airplane than it was assumed to be in the model, and the machine was therefore somewhat more responsive to longitudinal control, and partly by the effect of the slipstream of the control; but these factors can hardly be allowed for quantitatively, and the extent of the difference between the free-flight test and the computation is rather disappointing.

If the angle of attack and the speed be known it should be possible to compute the normal force acting on the wings, provided that a wind tunnel test on a model of the aerofoil used is available. This has been done for the case illustrated in figure 8, and the computed normal force is plotted against time in figure 9. No test of the actual aerofoil used was available for very large positive or negative angles (beyond +20° and −5°), and it was therefore necessary, as in the computations for spin, to take the curves for a slightly different but similar aerofoil. It is probable that the differences in lift coefficient between thin aerofoils of normal type and with virtually flat lower surfaces are neglible at angles beyond those of maximum lift coefficient.

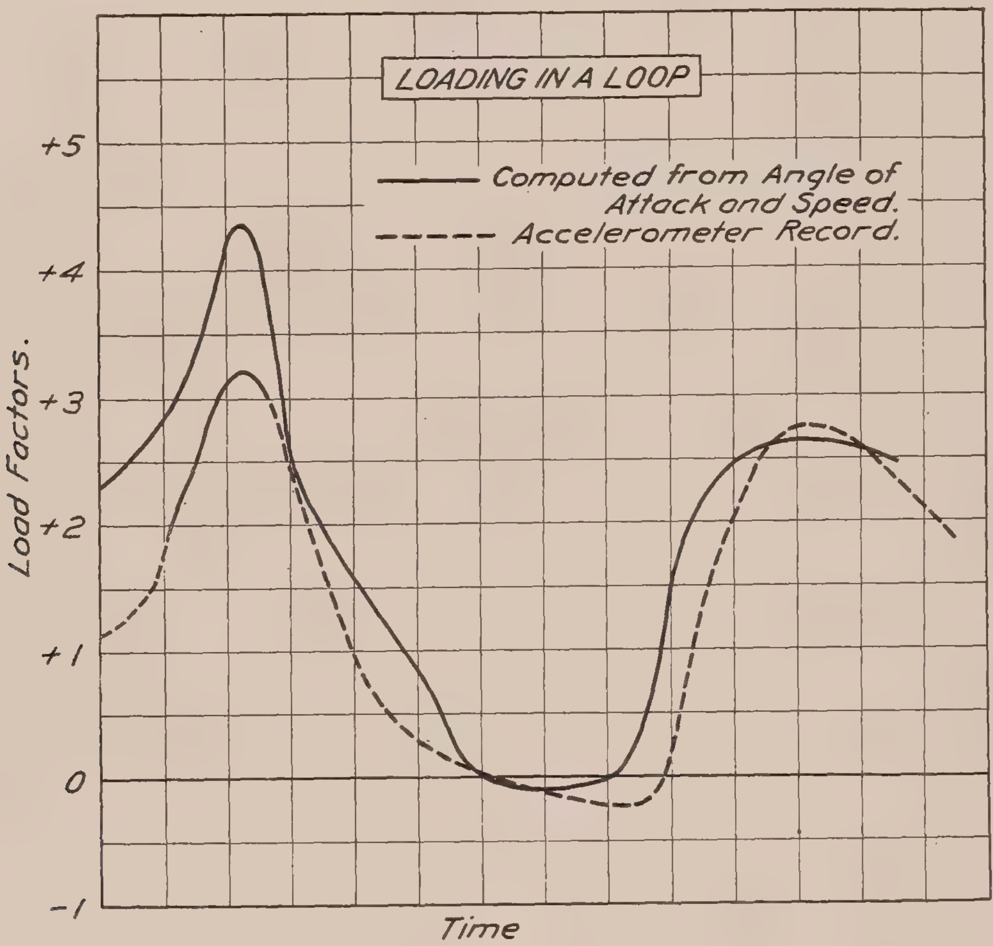

FIG. 9.

The computed curve of normal force shows a general correspondence of form with the curves determined directly by the use of the accelerometer for similar loops, but the maximum loading found by computation is far larger than that actually existing, The difference between the two maxima appears too large to be accounted for by errors or lag in the instruments, crude though they are, and the only other explanation that occurs is that the actual coefficient of normal force may be different when the airplane is accelerating and when the angle of attack is changing rapidly from that existing under steady conditions. Further tests on this point both in full flight and on models, are to be undertaken as soon as possible, as the point is one which has an important bearing on the maximum loading attainable and consequently on the necessary load factors to be used in design, as well as on longitudinal controllability.

REPORT No. 106

TURBULENCE IN THE AIR TUBES OF RADIATORS FOR AIRCRAFT ENGINES

By S. R. PARSONS
Bureau of Standards

REPORT No. 106.

TURBULENCE IN THE AIR TUBES OF RADIATORS FOR AIRCRAFT ENGINES.

By S. R. Parsons, Bureau of Standards.

This report describes an investigation of the characteristics of flow in the air passages of aircraft radiators, the work being done at the Bureau of Standards for the National Advisory Committee for Aeronautics.

RÉSUMÉ.

The existence of turbulent flow in the air passages of aircraft radiators, and of variations in character or degree of turbulence with different types of construction, is shown by the following experimental evidence:

(1) Pressure gradients along the air tubes are roughly proportional to the 1.7 power of the speed, which is characteristic of turbulent flow in long circular tubes of the same diameters.

(2) The surface cooling coefficients of radiators vary widely (0.002 to 0.007) when expressed as heat dissipated per unit time, per unit cooling surface, per unit temperature difference between air and water, and at a given average linear speed through the tubes.

(3) A fine wire electrically heated shows different cooling coefficients in the air tubes of different radiators.

(4) Temperature gradients in the air tubes are of the form characteristic of turbulent flow, and fail to show sudden breaks, such as might indicate a dividing line between regions of viscous and of turbulent flow.

The use of special devices for increasing turbulence may increase the heat transfer per unit surface for a given flow of air through the radiator, but decreases that flow for a given speed of flight, and increases head resistance. At very low flying speeds, or in cases where the radiator is mounted in the nose of the fuselage, turbulence devices may sometimes be used to advantage, but every type known to this bureau is detrimental to the general performance of the radiator at high speeds.

INTRODUCTION.

The primary requirement of a cooling radiator is evidently that it shall dissipate heat; and for cooling the engines of aircraft it is very important that the head resistance shall be low. But both heat transfer and head resistance are greatly affected not only by the speed of air past the cooling surfaces, but by the character of the flow—whether the air passes through the radiator in smooth streams, or with eddies and vortices. Furthermore, if the flow is turbulent, the questions arise whether the turbulence can be increased by changes in construction, and if so whether the result is beneficial or harmful to the general performance of the radiator.

This paper will not attempt to take up the questions of turbulence from a theoretical standpoint, but rather to present experimental evidence bearing on the problems and to state certain conclusions that seem to be warranted by the evidence. In particular, the paper seeks to answer the questions (1) whether the flow of air through ordinary radiators is turbulent at usual speeds; (2) whether there may be different degrees or types of turbulence; and (3) whether forms of construction intended to increase turbulence are beneficial or detrimental to the radiator.

The heat dissipated by a radiator is taken up by streams of air flowing through its tubes, and both the quantity of air delivered by the streams, and their condition of turbulence, are

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important factors in the dissipation of heat. Since air transmits heat only slowly by conduction, but principally by convection, the amount of heat taken from the metal surface depends very greatly on the number of *different* molecules of air that come in contact with the metal, and the most rapid transfer of heat requires a considerable scouring of the surface, while a layer of stagnant air acts as an effective insulator. But the collision of molecules of air with those of metal, imparting heat to the air because of the molecular motion in the hot metal, also ends because of the *mass* motion, to drag the air along with the radiator, instead of allowing it to pass through the tubes; and while turbulence in the air streams facilitates heat transfer, it also increases head resistance.

It is well known that in long tubes with smooth walls, at points not too near the ends, air flow is of two kinds, depending upon the relations between diameter of the tube and speed, viscosity and density of air. At low speeds, the flow is practically along stream lines parallel to the walls of the tubes, and is called viscous or "streamline" flow; while at higher speeds in the same tube, the flow is broken up into vortices, and is called turbulent flow. In viscous flow the skin friction, or resistance to flow, is due principally to the viscosity of the air, and is roughly proportional to the first power of the speed; while in turbulent flow viscosity is of less importance than density, and the resistance is roughly proportional to the square of the speed. The shortness of the tubes of radiators, and the irregular and broken forms often employed, make it unsafe to apply the theory of long tubes.

Most investigators measuring pressure drop in long tubes take no measurements of pressure nearer than 50 to 100 diameters to either end, but the total length of radiator tubes is usually not more than 10 to 30 diameters, and the fact that the rate of flow is ordinarily far above the critical velocity for a long tube of the same diameter is not sufficient basis for a statement that the flow in the radiator is turbulent.

It may be expected, however, that in a cluster of tubes such as a radiator, conditions corresponding to *long* tubes may be found much nearer the ends than in a single tube. Unless some kind of a mouthpiece is provided, air coming over the edges of a single tube enters from many directions; and the same is true to some extent of radiator tubes that are near the edges of the section; but the air that enters tubes near the center is confined by that entering the other tubes, and is fairly well directed even before it reaches the tubes.

Three general methods may be used for detecting turbulence; a visual method using some kind of smoke; measurements of pressure gradients; and measurement of heat transfer or of temperature. The visual method has great advantages, but is inconvenient for work inside of the radiator because of difficulty of arranging the apparatus so that it shall not disturb the flow of air, and at the same time so that air currents in the radiator tubes may be distinguished from currents before and behind it. The fact that the pressure gradient along an air stream is roughly proportional to the first power of the speed for streamline or viscous flow, and to the square of the speed for turbulent flow, may be used to determine the nature of the motion. The transfer of heat from a surface swept by a stream of air depends upon the turbulence of the stream as well as upon its velocity, and while with the present limited knowledge of coefficients of heat transfer a single measurement might be of little value for detecting the presence or absence of turbulence, considerable information may be gained from comparative measurements. Temperature measurements at different points in a stream of air that is being heated or cooled may, if reliable, furnish some indication of the condition of the air, by showing how heat is transmitted through different portions of the stream.

The experimental work on which the conclusions are based was carried out in an 8-inch (20 cm) square wind tunnel, and the evidence here presented is not sufficient for a confident answer to the question whether the same kind of flow is found in a stationary radiator with air blowing through it, as in a radiator moving through still air. It seems to be shown conclusively, however, that at least when the radiator is in the wind tunnel there are characteristic conditions of turbulence in different types of core, and it seems reasonable to suppose that such characteristic conditions would also be found in radiators moving through still air.

The experimental work undertaken for the present investigation includes the following parts, which will be treated in detail:

I. Flow in the channel in front of the radiator.
II. Pressure gradients in radiator tubes.
III. Cooling coefficients of radiators.
IV. Cooling of wires in an air stream.
V. Temperature gradients.

I. FLOW IN THE CHANNEL IN FRONT OF THE RADIATOR.

The characteristics of the air stream in the tunnel before entering the radiator were studied by two methods—by measuring the velocity at different points across a section of the stream and by observing the behavior of ammonium chloride smoke.

The velocity was measured by a movable pitot-static tube, and found to be uniform within 2 per cent to within 1 cm ($\frac{1}{2}$ inch) of the walls.

Ammonium chloride smoke was introduced into the tunnel through a glass tube about 8 mm ($\frac{5}{16}$ inch) in diameter, projecting through the bell mouth of the tunnel to a few centimeters beyond the straightening honeycomb at its entrance. On looking into the tunnel, either downstream through the mouth or across the stream through a window in the top, the smoke was seen to follow a straight course down the stream, with very little spreading. When a radiator was placed in the tunnel the smoke was found deposited over an area of the face which was fairly sharply defined, rather than shading off gradually. At a distance of 1 meter from the mouth of the smoke tube, the areas ranged from 4 to 9 per cent of the cross section of the tunnel, indicating a slow mixing of the air stream.

II. PRESSURE GRADIENTS IN RADIATOR TUBES.

The measurement of static pressure and pressure gradients within the air tubes of radiators has been described in Technical Report No. 88, "Pressure Drop in Radiator Air Tubes," and subsequent to the preparation of that report other work has involved incidental measurements on a number of additional types of core. Figure 1, reproduced from Report No. 88, is typical of the results obtained by plotting pressures against distance along the tube at different speeds of air. In most cases measurements were taken at either three or four speeds, and the following table shows the powers of the speed to which the pressure gradients are proportional. In many cases the pressure curve has no straight portion, and the difference in pressure between two points inside of the tube was used in place of a gradient. The formula of C. H. Lees[1] for surface friction in long circular tubes gives an exponent of about 1.73 for the sizes found in most radiators, and the grouping of these powers around that value furnishes good evidence of turbulent flow.

TABLE I.—*Exponent of air flow to which pressure drop in the air tubes is proportional.*

Radiator.	Type.	Exponent.
A-7	Square cell	1.6
A-9	do	1.7
A-14	Square cell, irregular	1.7
A-19	Square cell, walls swaged	1.8
A-29	Square cell	1.8
A-31	do	1.9
B-3	Pseudo-hexagonal	1.8
B-13	do	1.6
B-17	True hexagonal	1.7
C-2	Pseudo-circular	1.7
C-12	True circular cell	1.8
C-13	do	1.7
D-1	Irregular	1.6
D-3	do	1.9
D-4	do	1.5

[1] Proceedings Royal Society of London, A 91, 1914, p. 46.

III.—COOLING COEFFICIENTS OF RADIATORS.

Cooling coefficients of surfaces in radiators have been obtained from tests of heat transfer on about 60 types of core[2] and for the purpose of comparison have been reduced to heat dissipated per unit time per unit cooling surface, per unit temperature difference between the air and the water in the radiator, when the flow of air is such as to give a mean speed of 26.8 meters per second (60 mi./hr.) through the radiator tubes.[3] The effect of a large amount of indirect cooling surface (surface not backed by flowing water) is to decrease the value of the coefficient because of the fact that for a given temperature of *water*, indirect cooling surfaces have lower mean temperature than the water-tube walls. This effect seems wholly insufficient, however, to account for the wide variation of the coefficients shown in the following table:

TABLE II.—*Surface cooling coefficients of radiators.*

Coefficient in $\frac{\text{cal.}}{\text{sec.} \times \text{sq. cm.} \times {}^\circ\text{C.}}$

Number of radiators.	Type.	Dimensions of cell.		Coefficient × 1000.
		Centimeters.	Inches.	
	SQUARE CELLS.			
6	Smooth walls	0.6	$\frac{1}{4}$	2.7–3.1
2	Walls swaged	.6	$\frac{1}{4}$	2.8–2.9
3	.....do	.8	$\frac{5}{16}$	2.6–2.8
9	Irregular, approximately square	.6	$\frac{1}{4}$	2.6–3.5
1	.....do	1.2	$\frac{1}{2}$	2.4
	HEXAGONAL CELLS.			
12	Pseudo-cellular	.9	$\frac{3}{8}$	2.3–3.2
1	.....do	.6	$\frac{1}{1}$	3.4
3	True hexagonal cells	.8	$\frac{5}{16}$	2.8–2.9
	CIRCULAR CELLS.			
1	Pseudo-cellular	1.2	$\frac{1}{2}$	2.2
3	True circular cells	.8	$\frac{5}{16}$	2.8–2.9
4	.....do	.9	$\frac{3}{8}$	2.9–3.1
	OTHER FORMS.			
5	Irregular cells, smooth walls	....	....	2.1–3.4
3	Flat plate water tubes	....	....	2.9–3.1
4	Perforated plate water tubes (whistling type)	....	....	3.0–7.7
2	Spiral vanes, in good thermal contact	....	....	4.3–4.4

The table shows that for radiators whose air passages have straight smooth walls, the coefficient ranges from 0.0021 to 0.0034; for cells with broken walls ("pseudo-cellular" types), from 0.0024 to 0.0034; for the perforated plate types that whistle in an air stream, from 0.0030 to 0.0077; and for a type with spiral vanes, is about 0.0044. In general, the coefficient decreases as the size of the cell increases.

The wide range of coefficients even for straight tubes, the high coefficient for the section with spiral vanes, and the very high values found with some of the perforated plate types, appear to show very strong evidence of varying conditions of turbulence in the different classes. The presence of a peculiar turbulence in the perforated plate types is also indicated by their whistle.

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[2] Technical Report No. 63.

[3] This coefficient is the factor q of an empirical equation that has been found applicable to radiators with smooth straight tubes:

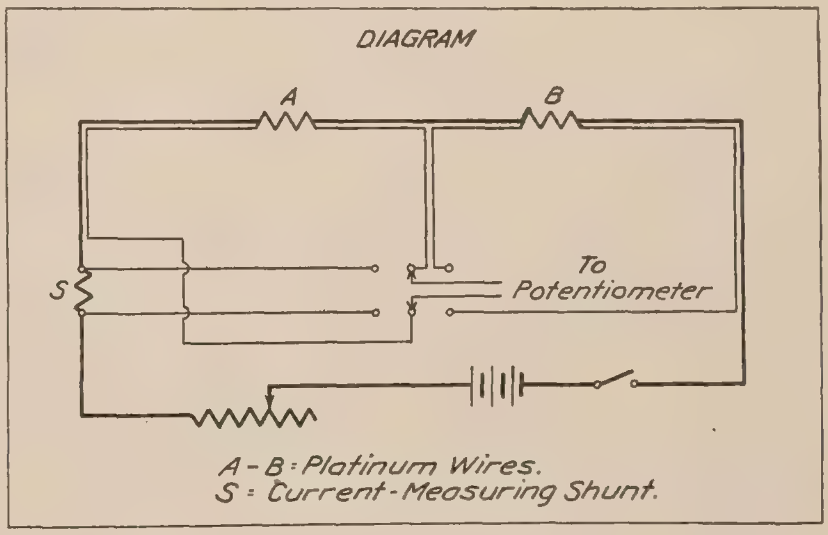
$$H = CMT\left(1 - e^{-\frac{qpx}{CM}}\right)$$

where H=heat transfer, units of power per unit frontal area.
C=specific heat of air at constant pressure.
M=air flow through the radiator, units of mass per unit time per unit frontal area.
T=difference between mean water temperature and temperature of air at entrance to radiator.
e=base of Naperian logarithms.
q=cooling coefficient, units of heat per unit time per unit surface per unit temperature difference.
p=total perimeter of air tubes (in frontal area) around a section perpendicular to the direction of air flow.
x=depth of radiator.

Coefficients for the radiators with true circular air tubes, if computed by the equation of Nusselt[4] would be 0.0026 which is somewhat lower than those observed. Nusselt's equation, however, can not be expected to apply to the radiator tubes, for it represents conditions with turbulent flow, in the central portions of *long* tubes.

IV. COOLING OF WIRES IN AN AIR STREAM.

The temperatures maintained by electrically heated wires, connected in series so that each should carry the same current, were used for a comparison of the cooling effect of the small streams of air flowing through a cold radiator, with that of the stream in the tunnel in front of it. A detailed description of the method follows:



Two pieces of 0.1 mm (0.004 inch) platinum wire, each about 6 cm (2.4 inches) long, and separated by 48 cm (19 inches) of No. 36 copper wire, were strung by copper leads straight along the channel so that one wire was in the stream some distance in front of the radiator, while the other was in one of the tubes, well back toward the rear face. The platinum wires, connected in series with a current-measuring shunt, as shown in the diagram, were heated by a small electric current, and potential drop was measured with a potentiometer across each platinum wire and the shunt. In order that the wires might disturb the air as little as possible, the potential and current leads were laid side by side (not twisted) and shellacked together for some distance from the platinum wires. The current leads supporting the wires were passed over rods set across the channel about 55 cm (22 inches) in front of and behind the radiator, and out through small holes in the floor of the tunnel, beyond which one wire was made fast, while a small weight hung on the other served to keep the wires taut.

The magnitude of the electric current was determined from the known resistance of the shunt and the potential drop across it, while from the current and the potential drops across the platinum wires, their resistances were obtained. The resistances when unheated were measured with small currents (about 0.006 amperes) and in an air stream of about 13 meters per second (30 miles per hour). It seemed hardly worth while to attempt to use the wires as resistance thermometers to the extent of measuring the temperatures to which they rose, but since for small temperature changes the resistance of the platinum wire is roughly proportional to the absolute temperature, the fractional increase in resistance of each wire was used as a rough measure of its rise in temperature on being heated.

Heat transfer from fine wires has been found to be proportional to the temperature difference and the square root of the velocity, for a silver wire in a tube of flowing water,[5] and for a platinum wire moved through air on a whirling arm,[6] and if the same relation is assumed

[4] Zeitschrift des Vereines Deutscher Ingenieure, 54, II, 1910, p. 1154.
[5] Rogovsky, Comptes Rendus, 136, June 8, 1903, p. 1391.
[6] King, Philosophical Transactions of the Royal Society of London, A 214, 1914, p. 373.

for the present case, a factor depending upon turbulence may be found as follows. The relation may be expressed by the equation

$$H = c\Theta\sqrt{v} \tag{1}$$

where

H = heat transfer from the wire, in units of power per unit length of wire.
Θ = temperature difference between wire and air.
v = linear velocity of air past wire.
c = a coefficient which includes a factor representing degree or nature of turbulence.

The air velocity is greater past the rear wire than past the front wire, because the radiator restricts the cross section of the stream, and

$$v_2 = \frac{v_1}{a} \tag{2}$$

where a = the fractional part of the frontal area of the radiator that is open for the passage of air, called its "free area," and the subscripts 1 and 2 represent the front and rear wires, respectively.

Turbulence at the front wire, i. e., in the open channel of the tunnel, is represented by the coefficient

$$c_1 = \frac{H}{\Theta_1\sqrt{v_1}} \tag{3}$$

and at the rear wire, i. e., in the radiator tube, by

$$c_2 = \frac{H}{\Theta_2\sqrt{v_2}} \tag{4}$$

The heat transfer H is the same in both cases, because the wires carry the same current and are of practically the same resistance per unit length. A comparison of turbulence in the radiator with that in front of it is obtained by substituting equation (2) in (4), and dividing by (3), which gives

$$\frac{c_2}{c_1} = \frac{\Theta_1}{\Theta_2}\sqrt{a}$$

As indicated above, the per cent of increase in resistance on being heated is used as a rough measure of the temperature difference. The values, including the ratio of the coefficients, are shown in Table III, and indicate differences in turbulence in different types of radiator, with the greatest turbulence in the perforated plate type, which whistles in an air stream.

TABLE III.—*Cooling of wires in the air stream.*

Type of radiator.	Air speed, m/sec.	Current, amperes.	Per cent increase in resistance. Front Θ_1.	Per cent increase in resistance. Rear Θ_2.	Free area a.	Ratio of coefficients $\frac{\Theta_1}{\Theta_2}\sqrt{a}$.	Mean ratio.
Flat plate	13	0.285	2.3	2.1	0.88	1.05	
	13	.457	8.1	7.8		.98	
	18	.456	7.4	7.0		1.00	1.01
Circular cell	13	.485	9.6	9.5	.65	.81	
	17	.472	9.3	8.5		.88	
	13	.304	3.7	3.3		.90	
	17	.303	3.6	3.3		.88	
	17	.497	9.9	9.0		.88	.87
Perforated plate	18	.208	0.8	0.5	.88	1.7	
	18	.407	5.9	4.4		1.26	
	13	.404	5.5	4.0		1.30	
	13	.448	7.6	5.6		1.29	1.30

V. TEMPERATURE GRADIENTS.

In order to obtain some indication of the distribution of temperature within the air passages of radiators, the following procedure was followed:

A section of radiator was mounted in the 8-inch (20 cm) wind tunnel, and hot water was pumped through it as in calorimetric tests. A copper-constantan thermocouple was strung through one of the air tubes of the radiator, and supported by its own copper leads, in the manner described above for the platinum wires. The constantan wire was about 30 cm (12 inches) long, and the cold (upstream) junction was outside of the radiator in the stream of incoming air, when the hot (downstream) junction was in any position in the radiator tube, or even somewhat behind the rear face. Screw threads with a pitch of 0.16 cm ($\frac{1}{16}$ inch) on the rods supporting the wires, and on the supports for the rods, furnished rough micrometers for setting the position of the thermocouple and for moving it horizontally and vertically.

The mean temperature difference between the water in the radiator and the air passing through it, and the speed of the air stream, were maintained approximately constant, and corrections for variations in the temperature difference were made on the assumption that the temperature rise indicated by the thermocouple was proportional to this difference. No correction was made for slight fluctuations in the speed of the air stream, for trial showed that the effect on the thermocouple readings was small, even when the speed was varied over a wide range.

In order that the air might be disturbed as little as possible, fine wires were used, and were bared for some distance each side of the constantan section. At first No. 38 wire was tried, but so much trouble was experienced with breakage that most of the work was done with size No. 36.

Figures 2 to 5 show temperature gradients across the center of the tube in four radiators, at different distances from the front face, the sides of the plots representing the walls of the tubes. Figures 6 to 10 show isothermal lines plotted from the data shown on the other curves, and the upper and lower sides of the plots indicate the walls of the tubes. It must be emphasized that the quantities shown on the curves are very rough values, and can be used quantitatively only with very great caution, if at all, because the steepness of the temperature gradients across the tube, and the uncertainty in the position of a thermoelectric junction suspended by a meter of fine wire in an air stream make individual readings quite unreliable. Indeed, the uncertainty of position made it impossible to duplicate readings with any accuracy after the wires had been moved forward or backward, and readings were accordingly taken across the tube from one side to the other before moving the thermocouple to a new position along the stream. But although individual readings are not very reliable, the qualitative indications of the plots are probably correct.[7]

It has been suggested that at sections of the tubes near the forward end both viscous and turbulent flow might be found, each occupying a certain part of the cross section. If such were the case, a sudden break would be expected in the temperature gradient across the tube, at the boundary between the two kinds of flow, but no indication of such a break is found in the data; and although the inability to get reliable readings very close to the walls (because of contact between the swinging thermocouple wire and the wall) might have concealed this condition in some parts of the tube, it seems reasonable to suppose that it would have been detected at least in the longest tube.

The curves are of the form characteristic of turbulent flow, and are similar to curves of temperature gradients,[8] and isothermal lines [9] found by other observers when working with long tubes at velocities well above the critical values.

[7] The possible effect of errors due to lead conduction in the thermocouple was investigated, and found to be entirely negligible in comparison with the known errors due to uncertainty of position.

[8] T. E. Stanton and Dorothy Marshall, British Adv. Com. Aero., Reports and Memoranda, No. 243, June, 1916.

[9] Groeber, Zeit. des Ver. Deut. Ing., 56, March, 1912, p. 421.

VI. EFFECTS OF TURBULENCE UPON RADIATOR PERFORMANCE.

In considering the effects of turbulence and turbulence devices on the performance of the radiator, it is necessary to bear in mind the importance of the *quantity* of air flowing through the core, and the fact that at a given speed *of flight* rather widely different amounts of air flow through different radiators. The comparisons made above have been based on a given air flow through the core, but speed of flight is the proper basis for comparing the general performance characteristics of a radiator.

Any form of construction that imparts additional turbulence to the air may be expected to increase the resistance to flow of air, and consequently to decrease the flow through the radiator for a given flying speed, while at the same time increasing the head resistance. If, then, there is to be a gain in general performance, any device for producing turbulence must, by increasing the amount of cooling surface, or by causing the air to scour the surface more thoroughly, or both, increase the heat transfer enough to overbalance both the decrease in amount of air flow (which tends to decrease the heat transfer), and the increase in head resistance.

The general performance of four types of radiator, each representing one of the best of its class, is shown in figure 11, by the "figure of merit," which is the ratio of the rate of dissipation of heat (expressed in units of power) to the power absorbed in overcoming the head resistance and sustaining the weight of the radiator.[10]

It is noticeable that at the higher speeds the flat plate and square cell types show much higher figure of merit than the other two types, although at a speed of 9 meters per second (20 miles per hour) the type with spiral vanes would perhaps be better than any of the others. The figures of merit as drawn apply only to radiators mounted in "unobstructed" positions, such that the flow of air through and around them is practically unaffected by other parts of the aircraft. For use in such positions at high speeds, every form of turbulence device known to this bureau is detrimental to the general performance of the radiator. On the other hand, if the radiator is to be used in such a position as the nose of the fuselage, the air flow through it at best is low, and an increase in air flow is accompanied by an increase in head resistance of the combination of fuselage and radiator. In this case, heat must be transmitted as rapidly as possible to the small amount of air that does flow through, and it may be profitable to use turbulence devices. It is possible that the rate of heat transfer for the whole radiator may be increased, while added air resistance of the core may actually reduce the head resistance of the fuselage and radiator.

[10] The figure of merit is computed on the assumption of temperature difference of 100° F. (55.6° C.) between air and water, and a "lift-drift" ratio of 5.4 for the airplane.

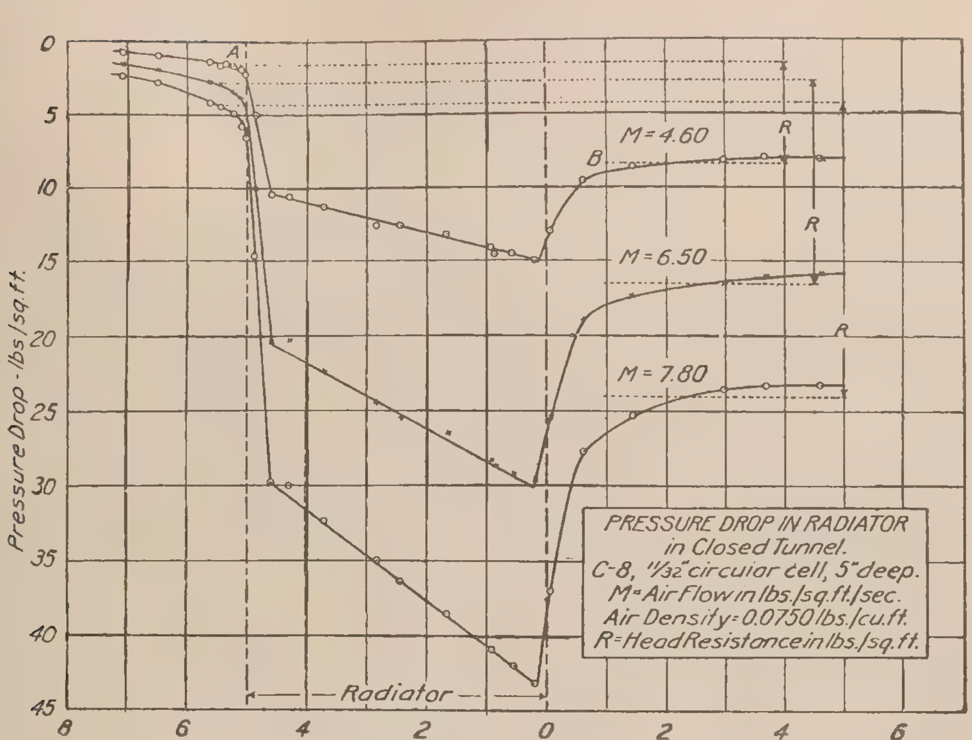


Fig. 1.

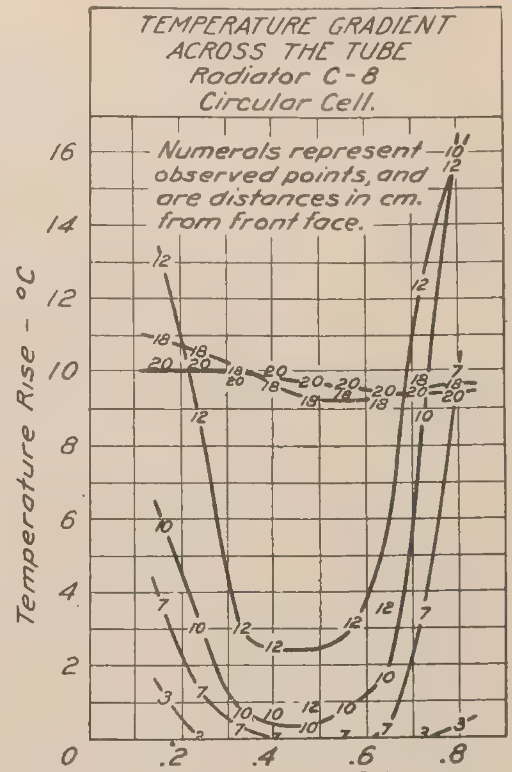


Fig. 2.

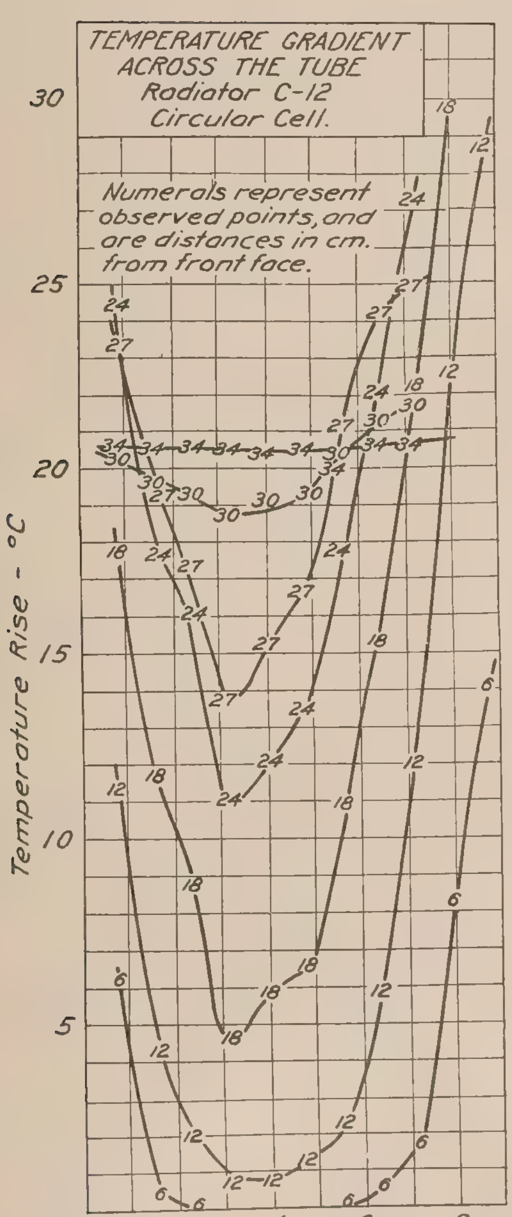


Fig. 3.

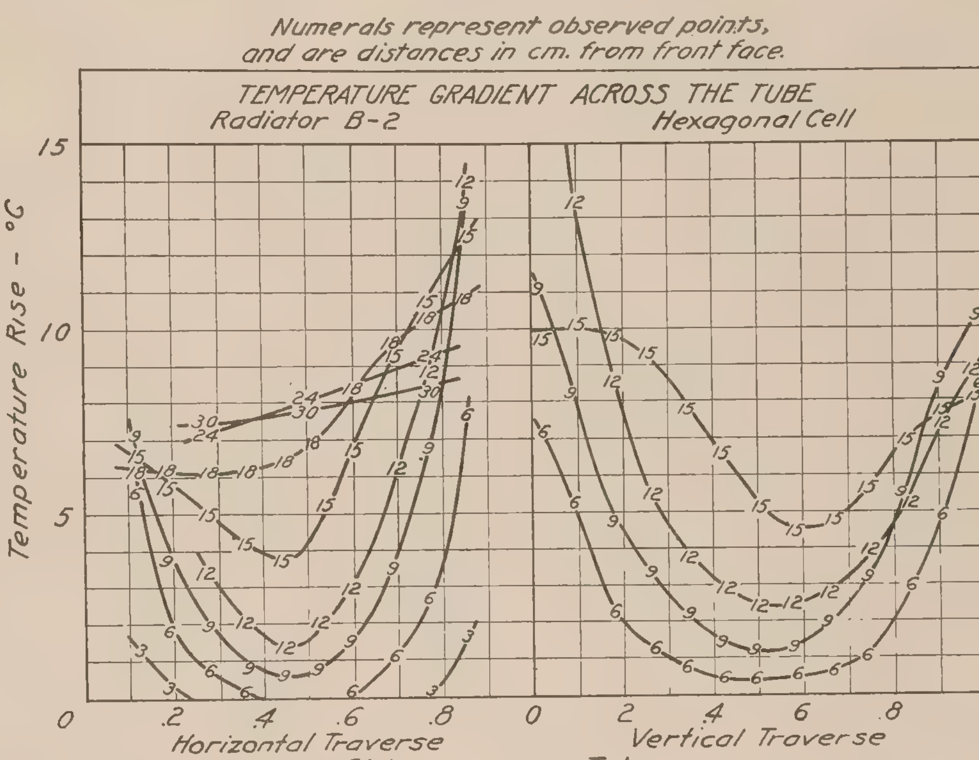


Fig. 4.

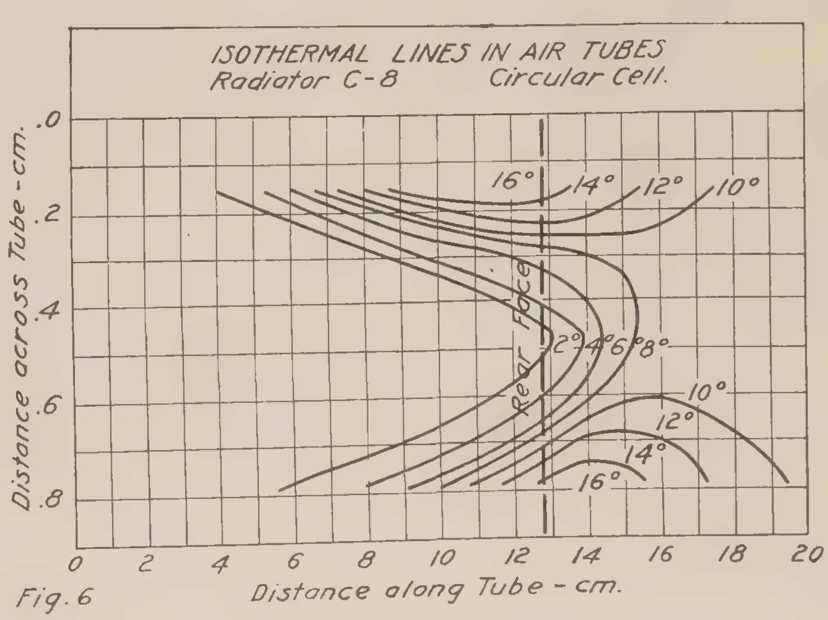


Fig. 6

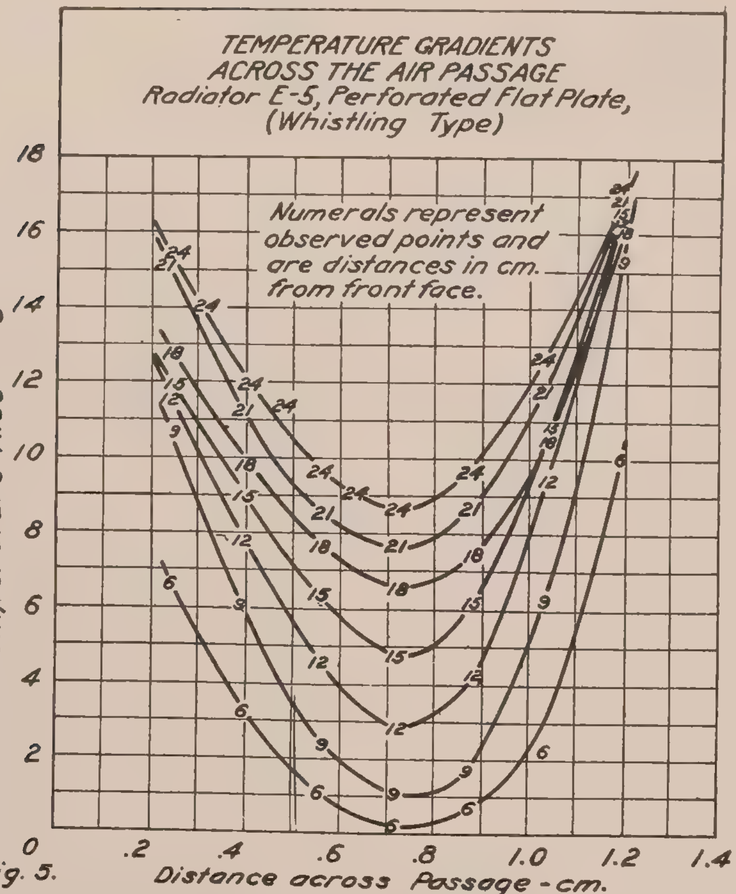


Fig. 5.

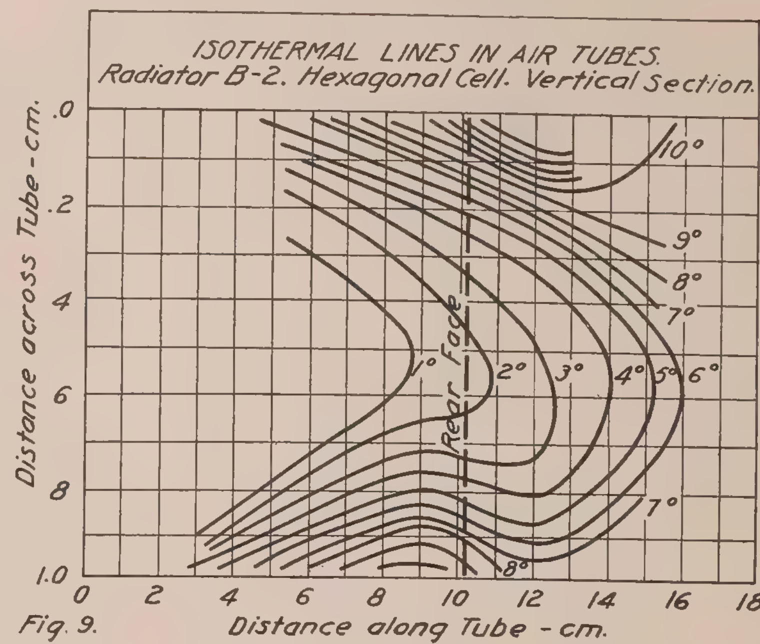


Fig. 9.

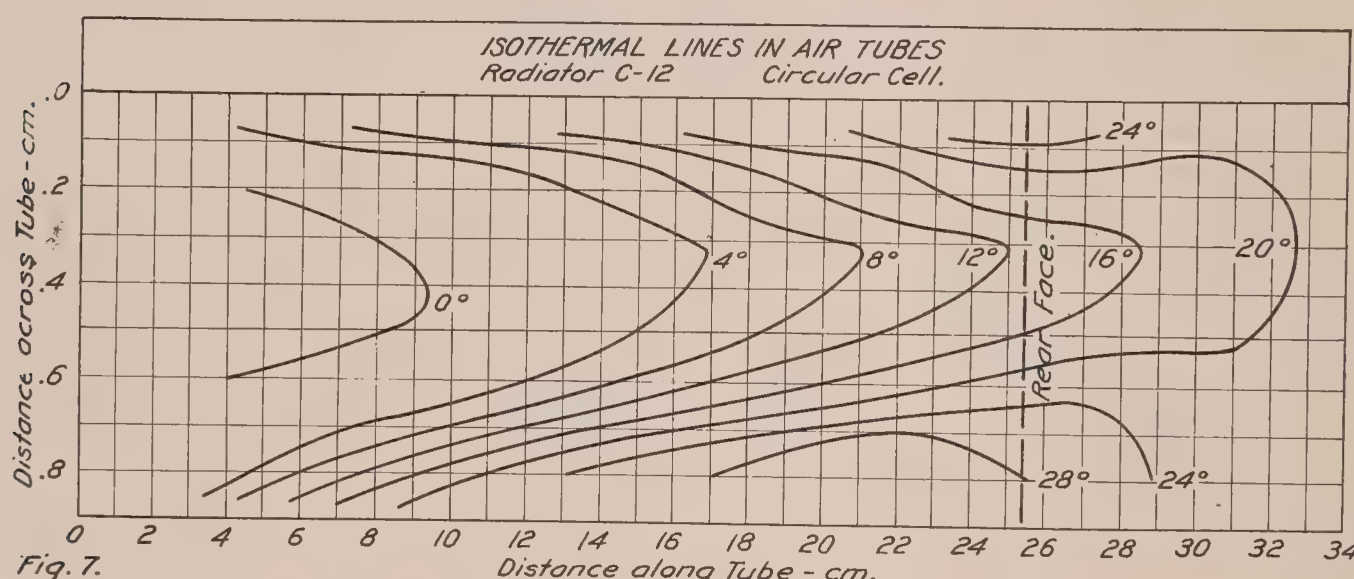


Fig. 7.

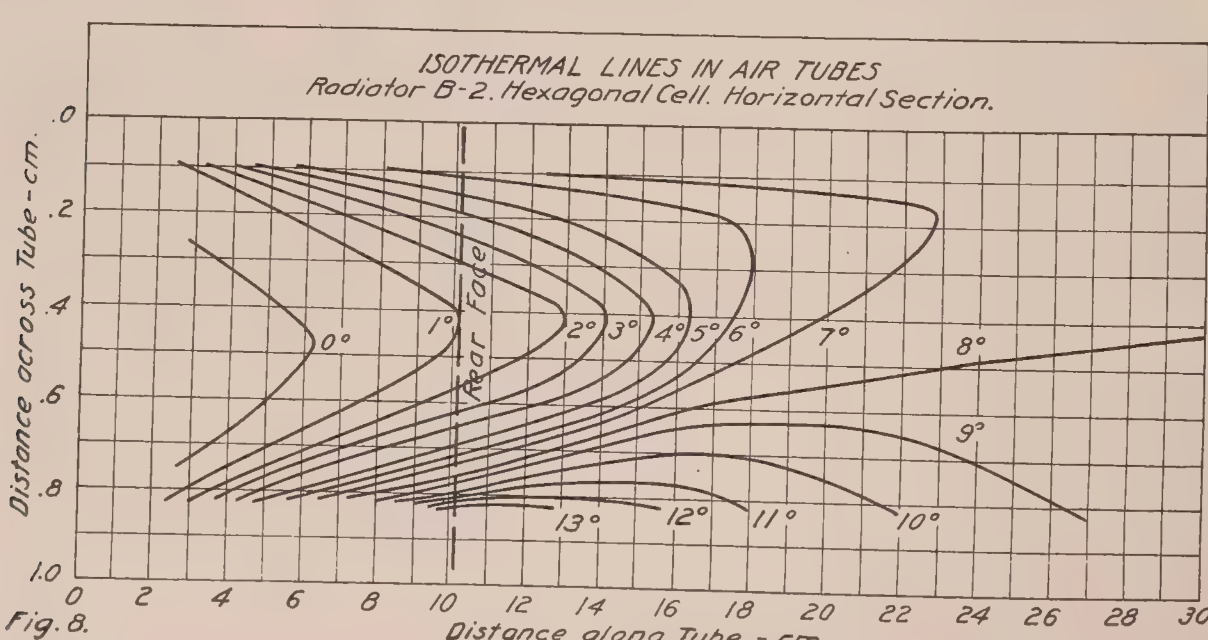
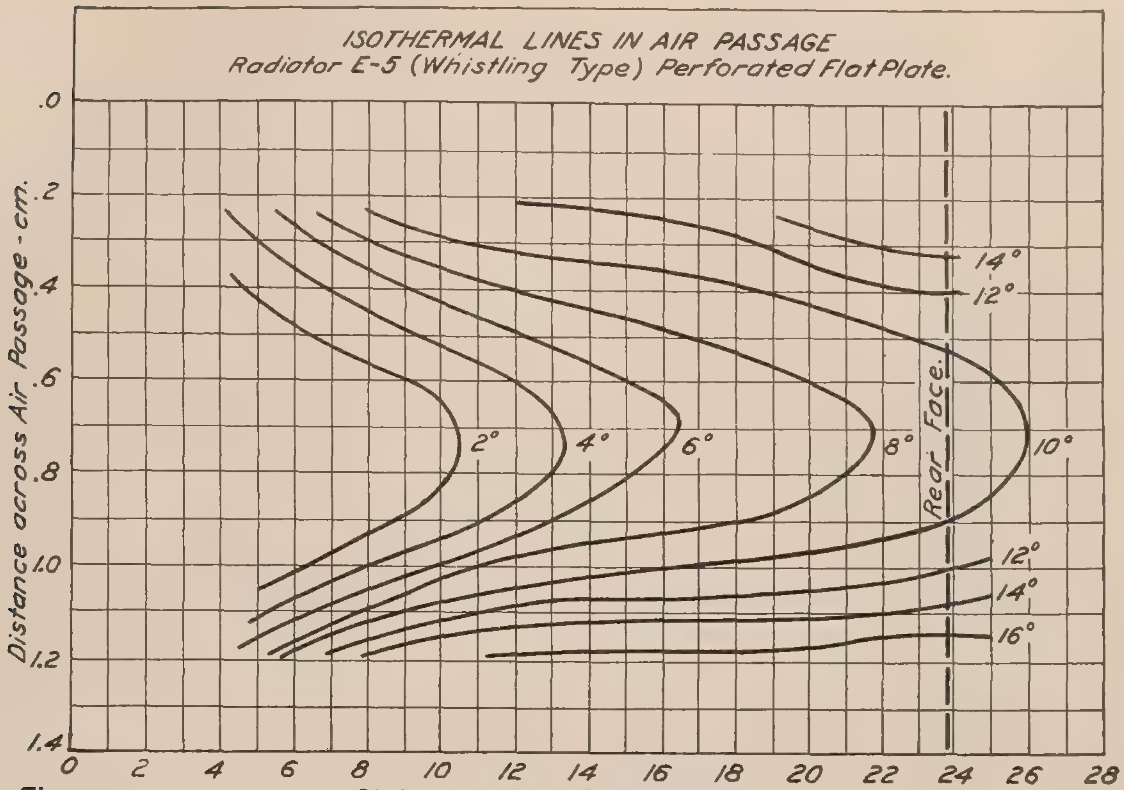


Fig. 8.


ISOTHERMAL LINES IN AIR PASSAGE
Radiator E-5 (Whistling Type) Perforated Flat Plate.
Distance across Air Passage - cm.
.0
.2
.4
.6
.8
1.0
1.2
1.4
0
2
4
6
8
10
12
14
16
18
20
22
24
26
28
2°
4°
6°
8°
10°
12°
14°
16°
Rear Face.
Distance along Air Passage - cm.
Fig. 10.

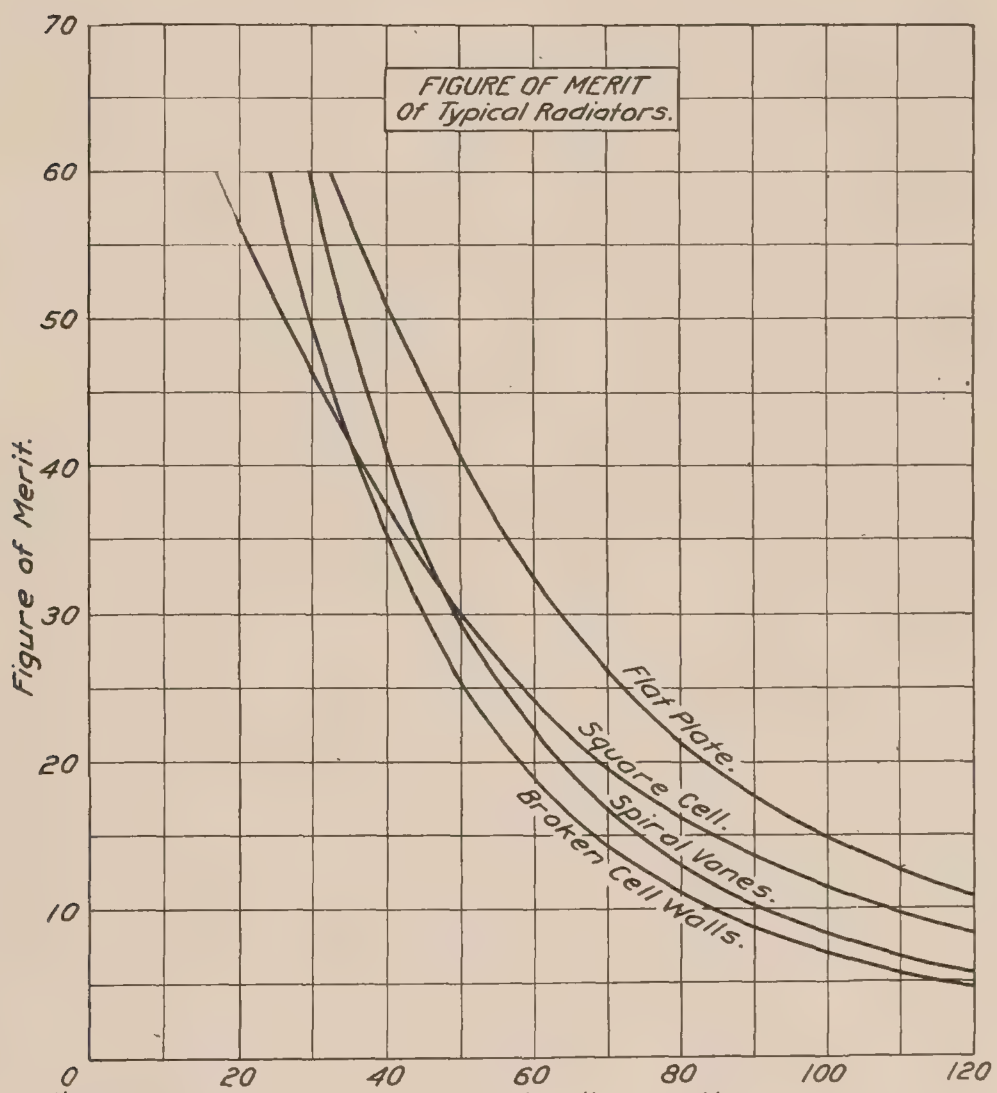


FIGURE OF MERIT
Of Typical Radiators.
Figure of Merit.
0
10
20
30
40
50
60
70
20
40
60
80
100
120
Flat Plate.
Square Cell.
Spiral Vanes.
Broken Cell Walls.
Flying Speed - Miles per Hr.
Fig. 11.

REPORT No. 107

A HIGH-SPEED ENGINE PRESSURE INDICATOR OF THE BALANCED DIAPHRAGM TYPE

By H. C. DICKINSON and F. B. NEWELL
Bureau of Standards

REPORT No. 107.

A HIGH-SPEED ENGINE PRESSURE INDICATOR OF THE BALANCED DIAPHRAGM TYPE.

By H. C. Dickinson and F. B. Newell.

Bureau of Standards.

RÉSUMÉ.

This report was prepared for the National Advisory Committee for Aeronautics and describes a pressure-measuring device especially adapted for use in mapping indicator diagrams of high-speed internal combustion engines. The cards are obtained by a point-to-point method giving the average of a large number of engine cycles. The principle involved is the balancing of the engine cylinder pressure against a measured pressure on opposite sides of a metal diaphragm of negligible stiffness. In its application as an engine indicator the phase of the engine cycle to which a pressure measurement corresponds is selected by a timing device. The report discusses briefly the errors which must be avoided in the development of an indicator for light high-speed engines, where vibration is serious, and outlines the principles underlying the design of this instrument in order to be free of such errors. A detailed description of the instrument and accessories follows, together with operating directions. Specimen indicator diagrams are appended. The indicator has been used successfully at speeds up to 2,600 revolutions per minute, the highest speed engine available for trial. Its sensitivity is approximately that of a standard 6-inch dial gauge of the Bourdon tube type.

INTRODUCTION—REASONS FOR THE DESIGN.

Prior to 1917 there were available several types of instruments for the measuring and recording of pressures in internal combustion engine cylinders. Some of these were refinements of the conventional pressure indicator designed for low-speed steam engines while others were designed primarily for high-speed work with a view of minimizing the effects of inertia. These instruments were found to be useful for various classes of work, depending upon their design and characteristics, although none of them had found more than very limited application in a comparatively few laboratories.

Some of the inherent difficulties which have prevented the development of a wholly convenient and successful high-speed indicator are inertia, friction, and back lash in moving parts where mechanical means of recording are adopted; inertia and vibration of the system when a photographic method of magnification is applied to an instrument mounted on the engine; time lag of the gases in the connecting tube where a photographic apparatus is mounted independently of the engine and connects to it by a flexible tube; as well as the usual mechanical difficulties in the construction and operation of instruments of this class.

It should be noted, too, that these difficulties are very greatly increased in the case of the aircraft engine, which usually must be mounted on a more or less flexible stand and in which at best the mechanical vibrations of the parts are excessive, due to their light weight and lack of rigidity.

An important mechanical consequence of this excessive flexibility of the engine structure seems to have been overlooked, since, so far as the authors know, it has not been discussed in the literature or taken account of other than accidentally in design of any indicator. To illustrate this effect, assume an indicator whose moving parts are mounted on the head of an engine cylinder. In order to reduce the effect of inertia, the range of motion of the piston or diaphragm is reduced as much as practicable and the motion magnified either mechanically or optically so

as to give a readable scale. It should be noted that the motion actually recorded is always the relative motion between the movable part of the indicator and the (supposedly) fixed part mounted rigidly on the cylinder head. But if the cylinder head itself flexes under pressure or mechanical vibration, this motion of the fixed support relative to the movable piston is recorded and magnified as well as the motion of the piston relative to its support. Hence it may happen that thus limiting the range of motion of the piston or heavy diaphragm and greatly magnifying the record, increases the bad effects of inertia on a light flexible engine cylinder since the motions of the cylinder head itself, relative to the moving member, are subject to the same degree of magnification.

The rapid increase in volume and scope of experimental work on gasoline engines, particularly of aircraft engines, due to the impetus given by war conditions, intensified the need for suitable indicators and several laboratories undertook their development.[1] The Bureau of Standards was particularly interested in securing an indicator suited to use in the altitude chambers[2] where aircraft engines are operated for purpose of test and analysis of their performance under reduced pressure and temperature simulating conditions of flight. The altitude chambers inclose only the engine, all controls and measuring apparatus being outside, whence in addition to all other requirements, it was essential that any indicator adopted should possess the feature of remote control and reading. For the purpose in hand, for general analysis of engine performance, accurate indicator cards are of more importance than are individual records of single cylinder cycles; therefore a point-to-point method can be employed.

A successful instrument[3] embodying the foregoing requirements has been developed and a half dozen of them have proved satisfactory for use under conditions of actual practice from 200 to 2,600 revolutions per minute (the highest engine speed available for test), and from 10 pounds per square inch below atmospheric pressure to 1,000 pounds per square inch above. The instrument has proved convenient in use and of high accuracy, being capable of measuring pressures to an accuracy comparable with that of the standard 6-inch pressure gauges used for recording these pressures. It is not only suited to the measurement of the pressures in internal combustion engines, but in any engine, compressor, or other machine in which gas pressures occur in successive cycles of the same form. For instance, the pressures occurring in the intake or exhaust manifold of a gasoline engine may be measured with the same instrument.

PRINCIPLE OF THE APPARATUS.

Fundamentally, the principles involved are the balancing of the cylinder pressure against a measured gas pressure on opposite sides of a metallic diaphragm of negligible stiffness and the indication or recording by means of a timing device of the instant at which equality of pressure occurs.

The indicator outfit thus consists of three parts: The pressure-balancing element, which is screwed into an opening in the engine cylinder as is a spark plug; the timing element, which is fastened securely to a revolving part of the engine; and the coordinating, measuring, and recording apparatus, which is located at any convenient place and is connected to the indicator and timer with wires and flexible pressure tubes. A very small portion of the engine cylinder gas whose pressure is to be measured surges back and forth through small, short water-cooled passages in the pressure element and transmits the cylinder pressure to the lower side of a thin metallic diaphragm which is clamped at its periphery. The deflection of this diaphragm is limited by two corrugated and perforated supports to a few thousandths of an inch. At least two instants in each cycle the pressure on the lower side of the diaphragm (cylinder pressure) is equal to the measured pressure which the operator has applied by means of gas supplied through a small copper tube to the space above the diaphragm. The timer selects that portion of the cycle, approximately 1 degree of arc, for which the measured pressures is to be made equal

[1] "High Speed Internal Combustion Engines," by Arthur W. Judge, Ch. V, Engineering, vol. 84, p. 570; vol. 102, p. 422. Auto Car, Feb. 2, 1907, p. 157. American Machinist, Nov. 29, 1906, p. 693; May 13, 1920, p. 1061. Horseless Age, Nov. 1, 1915, p. 418. Machinery, Dec., 1910. Journal of the Society of Automotive Engineers, Apr., 1920, p. 254.

[2] Report No. 44, National Advisory Committee for Aeronautics, "The Altitude Laboratory for Testing of Aircraft Engines."

[3] This indicator is manufactured by the American Instrument Co., Washington, D. C.

to the cylinder pressure. The coordinating apparatus indicates when the pressures are in balance, as will be explained. The pressures are measured on calibrated Bourdon pressure gauges, closed or open end mercury manometers, or any other accurate pressure-measuring instrument.

The indicator permits of plotting cylinder pressures from point to point in the cycle, giving at each point the average value from a number of cycles. It is suited only to application to engines operating under conditions sufficiently constant that successive cycles repeat their values of pressures within reasonable limits.

DETAILS OF CONSTRUCTION.

The mechanical details of design are best described by the drawings and photographs, figures 1 to 7.

The pressure element.—The pressure element is shown in figures 1, 2, and 3. A thin metal diaphragm (1) divides the chamber into two parts, the lower one communicating directly with the engine cylinder by screwing the threaded portion (2) into a spark-plug hole. The cylinder pressure is thus impressed on the diaphragm with a minimum of inertia or lag due to a long connecting passage. This close connection to the engine necessitates water-cooling, an annular circulation being provided (space 3).

The balancing pressure impressed on the top of the diaphragm is supplied by compressed air or other gas conducted through small copper tubing to the capacity space forming the upper chamber of the instrument and transmitted from this space through the perforations in the support to the disk.

The motion of the diaphragm, when the pressures are out of balance, is limited to about 0.13 millimeter (0.005 inch) by upper and lower supports. These are circular plates of brass about 5 millimeters (three-sixteenths inch) thick, perforated with No. 60 drill holes and surfaced with concentric corrugations where they have contact with the diaphragm. The upper support is plane and the lower is concave, 0.005 inch less in thickness at the center than at the periphery. These supports prevent distortion of the diaphragm beyond the elastic limit under pressures for which the diaphragm is intended to be used. The instrument can be taken apart with little trouble to insert a new diaphragm when necessary.

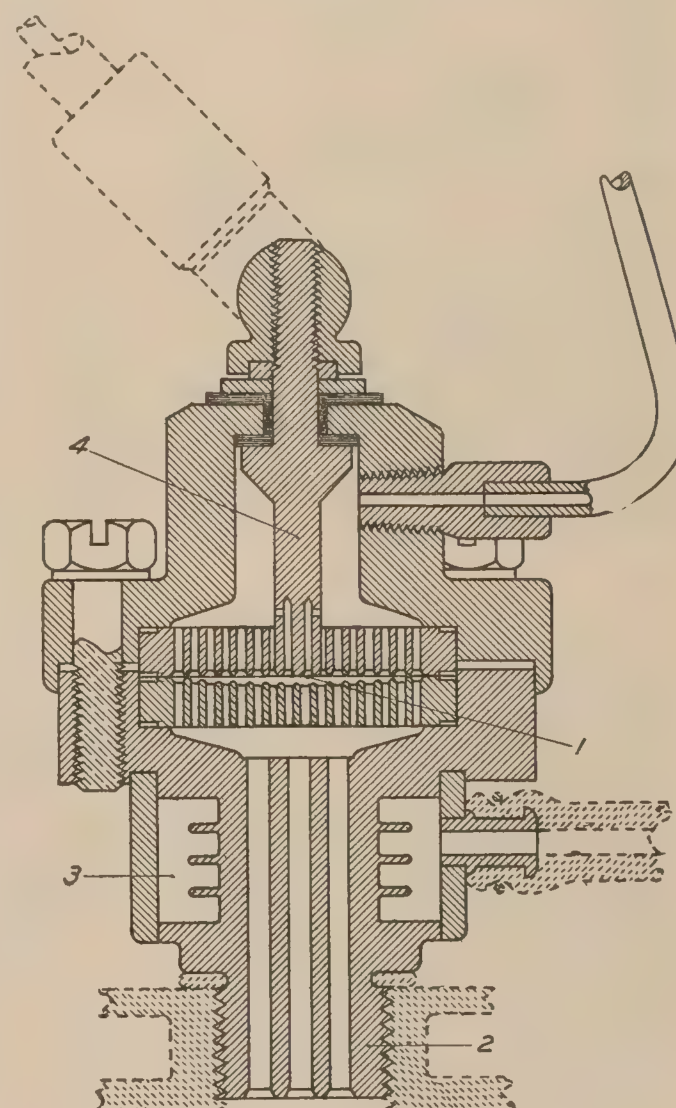


Fig. 1.—Pressure element: Principal section.

It is important to bear in mind the actual operation of the diaphragm and the function fulfilled by the supports. Above the diaphragm is the controlled pressure, sensibly constant over many seconds or minutes. Below is the engine pressure, varying from that of explosion to that of intake suction, the frequency depending on the engine speed (at 3,000 revolutions per minute for a 4-cycle engine the frequency is 25 per second). Accordingly, the diaphragm vibrates many times a second between its supports, moving each time the cylinder pressure becomes greater or becomes less than the balancing pressure. At only two points in each cycle (in normal operation) is the pressure on both sides of the diaphragm balanced. Except for these instants, the diaphragm is pressed against one support or the other, according to which pressure is the higher.

The diaphragm is a metal disk about 30 millimeters ($1\frac{1}{4}$ inches) diameter and about 0.08 to 0.15 millimeter (0.003 to 0.006 inch) thick. When clamped in the annular supports the free diameter is about 25 millimeters (1 inch). Steel diaphragms have been used most often,

although phosphor bronze ones have given satisfactory service. A metal having the mechanical characteristics of steel without its susceptibility to corrosion would be desirable. Nickel-plated and silver-plated steel have been used recently with success.

The time of response of these diaphragms is extremely short, but difficult to calculate or measure with accuracy. Its order of magnitude may be determined as follows: The pressure necessary to displace the diaphragm from one support to the other (i. e., the total pressure which the elasticity of the diaphragm can support) is normally from 0.1 to 0.5 pound according to thickness. In the normal use of the instrument, before the difference in pressure on the two sides of the diaphragm has reached an amount readable on the gauges (i. e., 0.1 to 0.5 pound) the diaphragm will be subjected to a pressure difference greater than can be supported by the elasticity, and will be thrust against one plate or the other. The actual time lag will be the time required for the diaphragm to move into or out of contact when the above pressure difference is applied to it. This time can be calculated roughly and is so small as to have no appreciable effect in any condition met with in practice.

The position of the diaphragm against one support or the other would give merely qualitative indication of which of two pressures is the greater, and the operation which makes a measuring instrument of the device is the passage of the diaphragm from the one side to the other, thereby indicating equality (or near equality) of the unknown and the measured pressures. The movement of the diaphragm is recorded by making or breaking an electric circuit, and an examination of figure 1 will reveal the insulation of the center part (4) of the upper support from electrical connection with the rest of the instrument. This electrode is connected in series with a telephone as a detector and one side of a battery of which the other pole is grounded on the engine frame and therefore the diaphragm. When the diaphragm moves upward against the electrode it closes the circuit, clicking the telephone; when it moves down it opens the circuit, also clicking the telephone. Telephone clicks are thus the means by which the observer is informed every time that the pressure in the engine cylinder is just balanced by the measured pressure which is transmitted to the upper chamber of the pressure element.

The timer.—The pressure element described above gives a signal of equality of a measured pressure with a particular value of the pressure in an engine cylinder, normally twice for each cycle, but gives no indication of the phase at which these equalities occur. To fulfill this function the timer is an essential element of the mechanism. It operates either to locate the portion of the cycle corresponding to a particular pressure, or to select a specific point in the cycle and permit the adjustment of the measurable pressure so as to equal the cylinder pressure for this point. The object is attained by introducing in the electric circuit a rotating member which closes the circuit only for about 1 degree of arc during each cycle, this member being in synchronism with the engine cycle and adjustable with respect to the phase of the latter.

The timer is shown in figures 4 to 7. The rotating part, an insulating disk, must be fastened rigidly to some part of the engine revolving at crank-shaft or cam-shaft speed. In the periphery of this disk is inlaid a narrow strip of brass extending over one-half degree of arc and serving as an electric contact, when it rotates past a fixed brush. The brush is a piece of hardened steel, fastened to the end of a flat spring so that it will rub on the periphery of the disk. It is carried by a graduated ring mounted on a ball-bearing concentric with the rotating disk. This ring can be rotated by hand and set at any desired angle with an index line on a portion of the frame that is immovable with respect to the engine frame.

An auxiliary device on the timer measures the angle in the engine cycle at which the ignition spark passes. This has no direct bearing on the use of the apparatus as a pressure indicator, but is extremely useful in mapping the ordinary indicator diagram of an internal combustion engine, because of the great importance of locating accurately in such a diagram the moment of firing the charge. The provision for this measurement is shown in figure 7. It adds very little complication to the timer.

Coordinating apparatus.—The auxiliaries to the pressure element and the timer may be divided into two general classes—those having to do with the control and measurement of pressure and those pertaining to the electric circuit. As used for measurements in the labora-

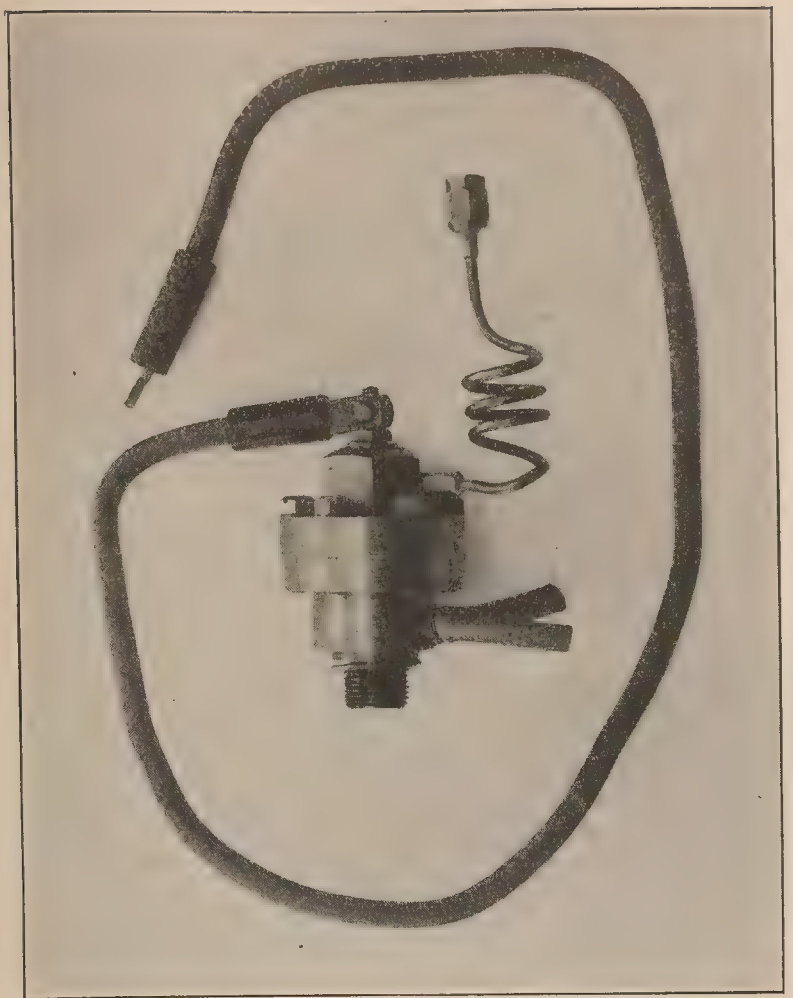


FIG. 2.—PRESSURE ELEMENT: ASSEMBLED.

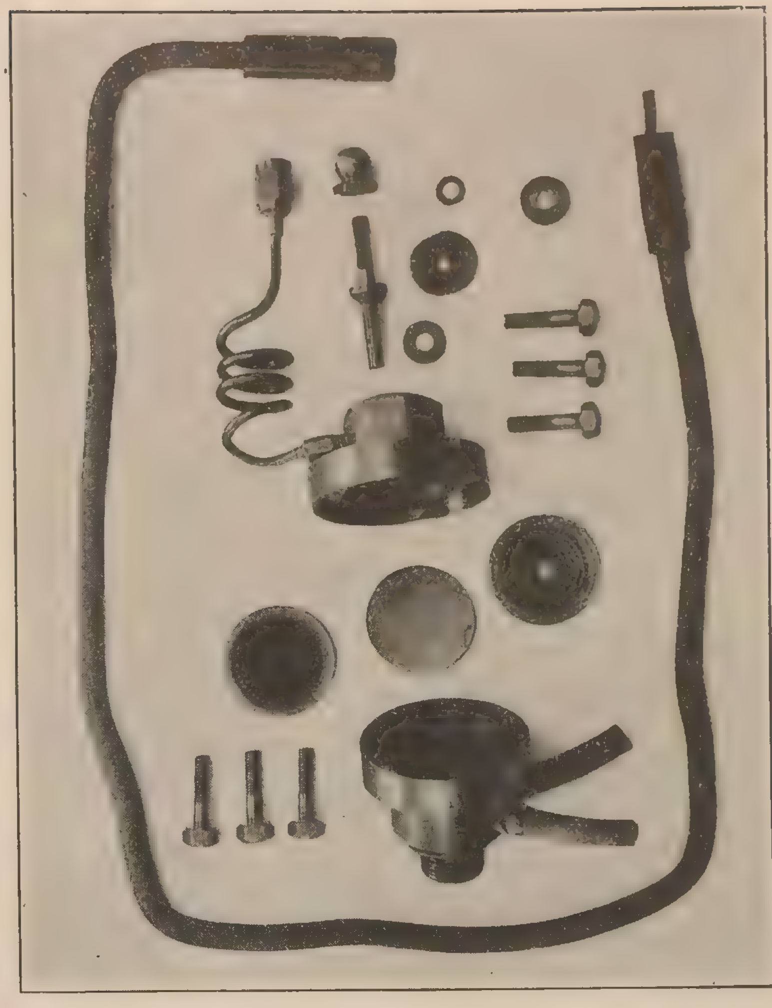


FIG. 3.—PRESSURE ELEMENT: PARTS.

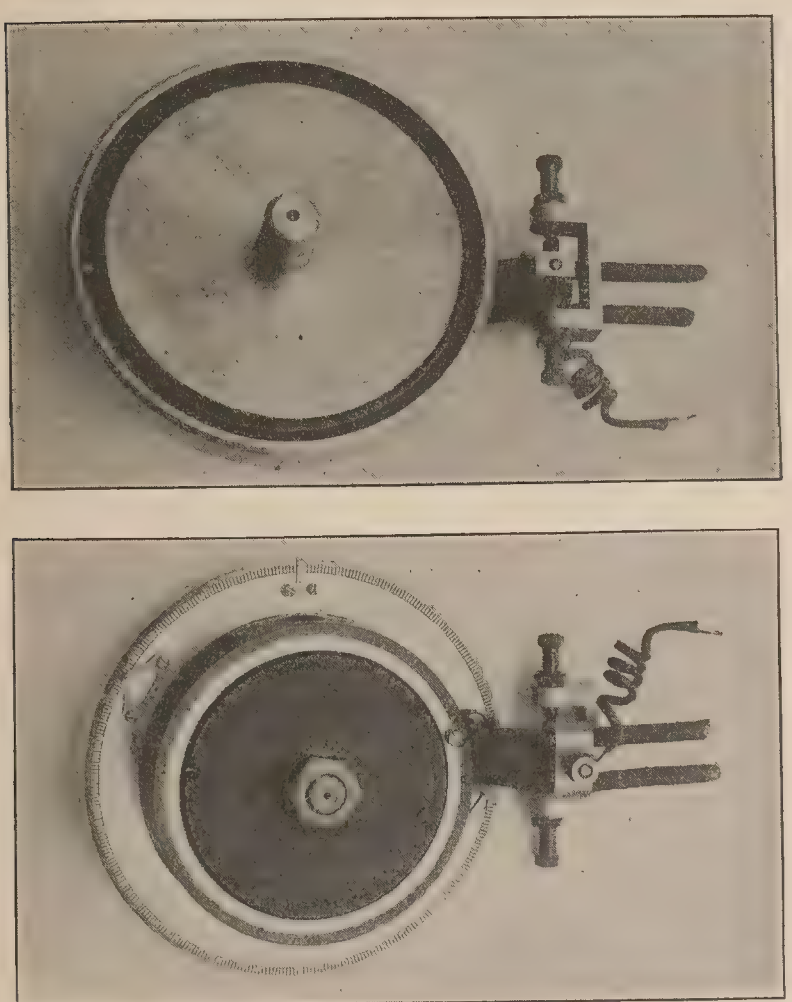


FIG. 4.—TIMER: FRONT AND REAR VIEWS.

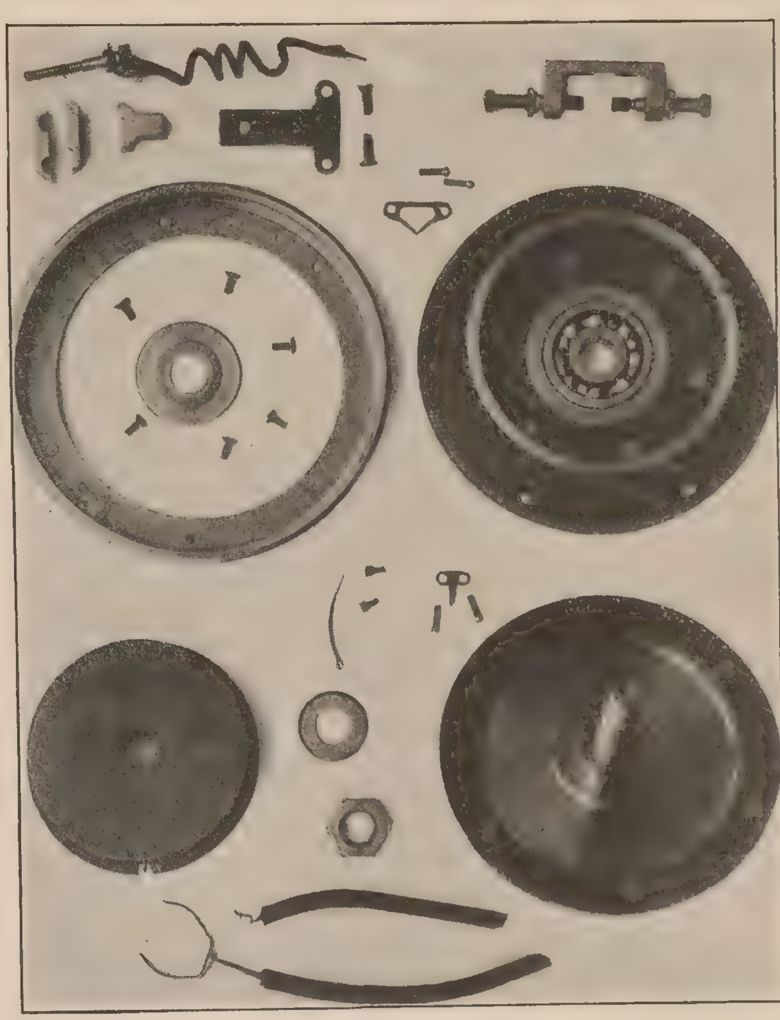


FIG. 5.—TIMER: PARTS.

tories of the Bureau of Standards, the source of high pressure is a tank of compressed air, or liquid CO_2 and that of reduced pressure is a water aspirator. Control is effected by a number of one-eighth-inch needle valves. The measuring instruments are Bourdon tube dial gauges of suitable ranges and a mercury manometer for pressures from subatmospheric to two atmospheres absolute. A standard 100-pound test gauge is used between 15 and 100 pounds per square inch above atmosphere, and a standard 1000-pound test gauge for all pressures from 100 to 1000.

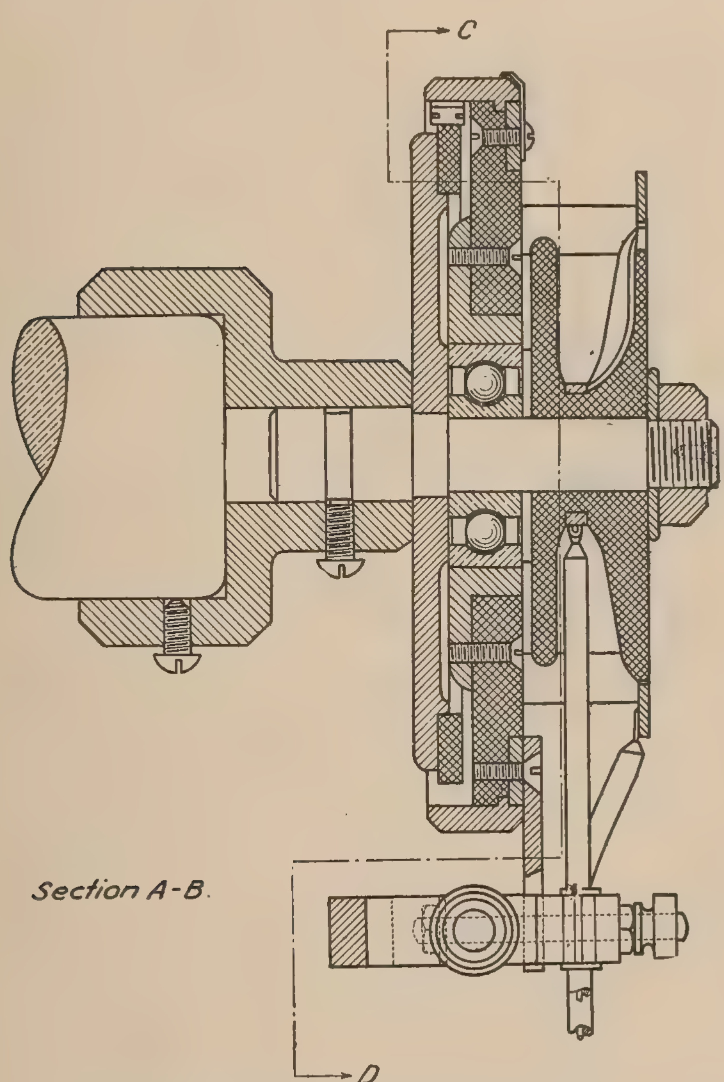


FIG. 6.—Timer: Principal section, parallel to axis.

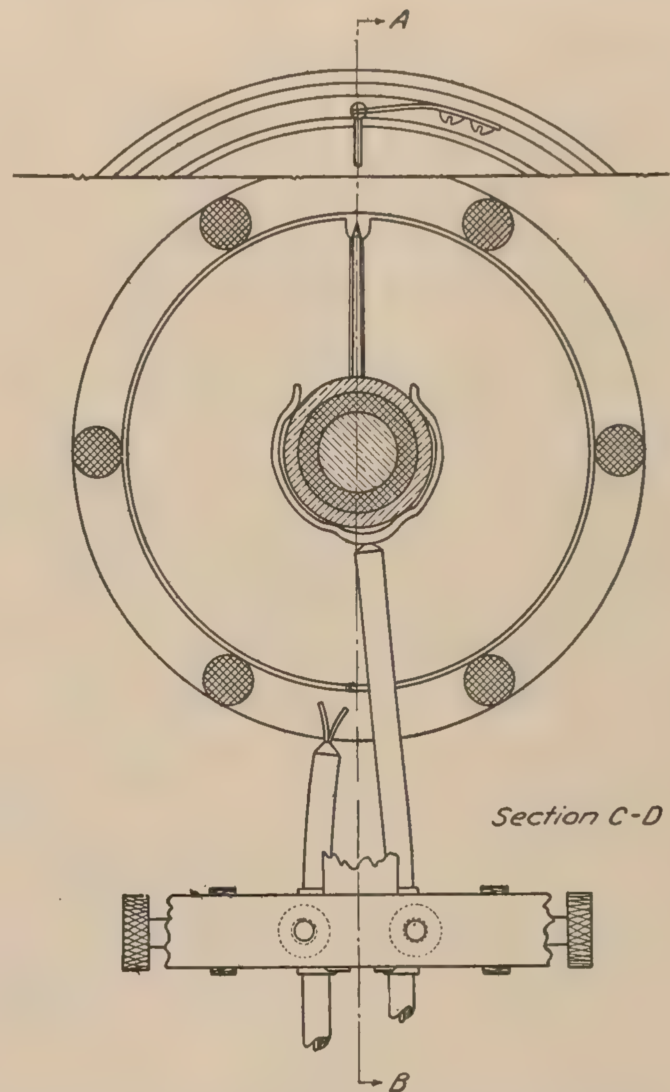


FIG. 7.—Timer: Section showing the segment with contact brush (at the top of the figure), and the rotary spark gap for locating the angle of ignition.

A schematic diagram of the apparatus is given in figure 8. The actual arrangement in the laboratory can be varied as needed to suit conditions. The engine cylinder may be quite remote from the remainder of the apparatus. In figure 8 the small tank near the right is filled to any desired pressure from the supply reservoir and then serves as the source of pressure. The water aspirator is diagramed next to the mercury U gauge. Connected to the same water line may be noted the cooling system for the lower part of the pressure element. The mechanical linkage of timer and piston is shown schematically.

The electrical connections are shown in the left-hand portion of figure 8. The essential elements already described include the telephone detector, the battery, the diaphragm and its contact electrode, and the timer contact. Auxiliaries shown in the diagram are the condenser and numerous switches, likewise the auxiliary circuit for measuring the position in the cycle when ignition occurs (see closing paragraph of the description of the timer). The condenser is used to modify the telephone action as required. When pressures in a high-speed engine are being measured, the timer segment is in contact with the brush so short a time that in a simple circuit the current would not build up in the telephone receiver to a value sufficiently high to cause a click surely audible. In this case a condenser is shunted across the phones so that when the circuit is closed it is charged and can discharge through the phone, intensifying the sound. When maximum pressures are being measured at slow speeds, the snap of the

receivers is disagreeably loud and sharp. To make this sound dull and yet audible, a larger condenser is shunted across the phones. Condensers of one-fourth, one-half, and 1 microfarad capacity are found suited to a circuit which includes a 4.5 or 6 volt battery and a pair of 70 to 80 ohm telephone receivers. For location of faults in the circuit it is very convenient to have switches short circuiting the indicator and the timer in addition to those actually necessary for operation.

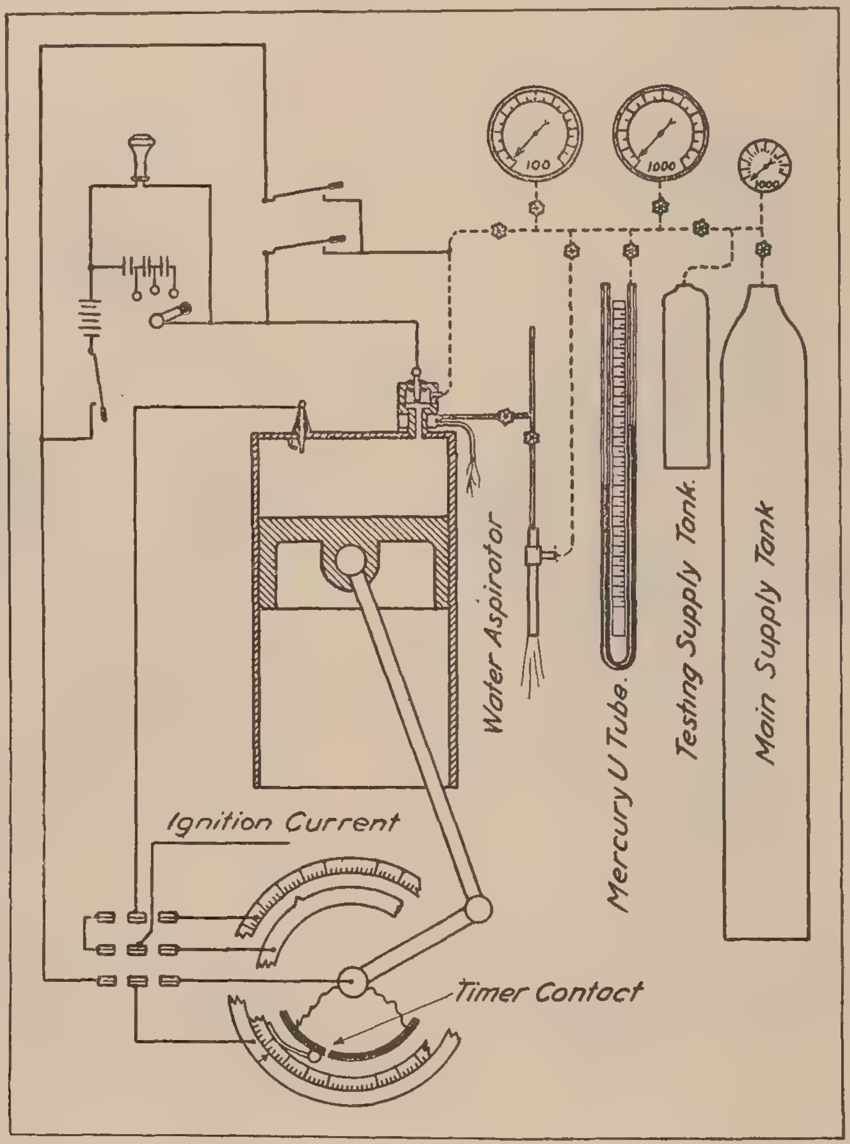


FIG. 8.—Schematic diagram of indicator assembly. Pressure element, timer, pressure system and electrical circuits in their relation to an engine cylinder.

CALIBRATION AND INSTALLATION.

The sensitivity and zero reading of the pressure element should be checked occasionally. By the first is meant the smallest pressure difference which will move the diaphragm sufficiently to cause a telephone click. By the zero reading is meant the difference in the absolute pressures on opposite sides of the diaphragm when it just makes or breaks contact with the electrode. This may be appreciable, owing to distortion of the diaphragm in clamping it into place or to any other lack of symmetry in its elastic behavior.

The zero reading is determined by leaving one side of the diaphragm open to the atmosphere and measuring the pressure on the other side required to obtain a balance. Whenever the zero reading becomes large, or the sensitivity unduly poor, it is time to replace the diaphragm.

The timer scale must be read in relation to engine-crank angle, or preferably set to zero for zero crank angle from piston dead center. A pair of adjusting screws which hold the timer arm in place facilitate this setting.

Leaks in the pressure system will introduce differences between pressures read on the gauges and those actually impressed on the diaphragm, due to pressure drop in the connecting tubes. Frequent tests for leaks are thus a necessary precaution, although leaks must be relatively large to have a significant effect.

DETAILS OF OPERATION.

While the precision of setting of the instruments is approximately equal to the sensitivity of the pressure gauges, the degree of certainty with which an observer can set for the mean cycle pressure at a given phase depends upon the degree of uniformity of the engine performance. This varies widely in different parts of the cycle, since the suction and compression strokes repeat with but small variations, while explosion and expansion pressures may be decidedly erratic.

The simplest measurement with the instrument described in this paper is that of maximum or minimum pressure occurring in the cycle. For such a measurement the timer is short circuited. Suppose, for example, a determination is to be made of the maximum explosion pressure occurring, on the average, in the operation of the cylinder, and suppose, for simplicity, that all cycles were alike—i. e., all these maxima identical in magnitude; then the behavior of the diaphragm will be as follows: When atmospheric pressure is impressed on the top of the diaphragm, the contact with the upper support and electrode will be broken only during a small portion of the cycle. As the pressure is increased, the diaphragm will be forced down earlier and return later in the cycle. When the pressure so applied has been increased to a value near the maximum engine pressure, the diaphragm will remain against the lower support throughout the cycle except for a very small interval near this maximum. Then the diaphragm will come up against the upper electrode, clicking the telephone, and will return almost immediately, clicking it again. There will thus be pairs of clicks a few milliseconds apart, succeeding silent intervals equal to nearly the period of the cycle. As the controlled pressure is raised still further, the pairs of clicks will merge into one, just at the maximum, and disappear above it. By slight variations of the controlled pressure, when near the correct value, it would be easy to make the measurement desired.

The conditions of actual practice preclude the simplifying assumption that the explosion pressures of many cycles are identical, and the range of variation is sometimes rather considerable. Under these circumstances as the measured pressure is increased to a value exceeding the lower values of maximum explosion pressures, the latter contribute no clicks of the telephone. Accordingly, as the pressure is increased from a low value where the telephone receiver clicks twice for every engine cycle, successive pairs of clicks merge and vanish so that the frequency of clicks gradually diminishes until the telephone is completely silent when the measured pressure exceeds the greatest of the explosion pressures. By exercising judgment as to the relative frequency of the telephone clicks with different measured pressures, the observer can reach a close estimate of the average of the maximum explosion pressures.

For measuring a pressure at any part of the engine cycle where the value is neither maximum nor minimum, it is necessary to have the timer in circuit.

As has been explained, the electrical circuit is closed in the indicator head for part of the engine cycle and open for the remainder, the point of closing and opening being dependent upon the gauge pressure applied by the operator. The telephone circuit is, however, closed only at the single instant—i. e., 1° of the timer contact. Thus the timer contact serves as an index to determine whether the indicator circuit is open or closed at a selected phase of the cycle; i. e., to indicate whether at this phase the gauge pressure is above or below the cylinder pressure. Thus if the engine operation is uniform from cycle to cycle, and if the gauge pressure is lower than the cylinder pressure at the phase of the cycle, the diaphragm will be up, the indicator contact closed, and a click in the telephone will occur at each closing of the circuit by the timer. If the gauge pressure is raised above the cylinder pressure at the selected angle, the circuit will not be closed in the pressure element and the timer at the same instant and, therefore, there will be no sound in the telephone.

The point of balance of the two pressures is therefore marked by a definite change in action of the telephone from clicking every engine cycle to complete silence. When the successive cycles do not repeat values of pressure within very close limits, the abrupt change from clicks to silence is replaced by a range of pressure over which clicking becomes irregular as the

gauge pressure is raised, the clicks ceasing entirely at a pressure equal to that of the highest cylinder pressure occurring in any cycle at this phase.

The timer can be revolved at either cam-shaft or crank-shaft speed, the choice being determined to a considerable extent by the construction of the engine with respect to ease of attaching such an extraneous mechanism. The cam-shaft speed is much the simpler for the operator to interpret the telephone clicks. In this case the complete rotation of the timer corresponds to one engine cycle, and the contact segment selects homologous parts of each cycle. When the timer operates at crank-shaft speed it makes contact twice in each cycle,

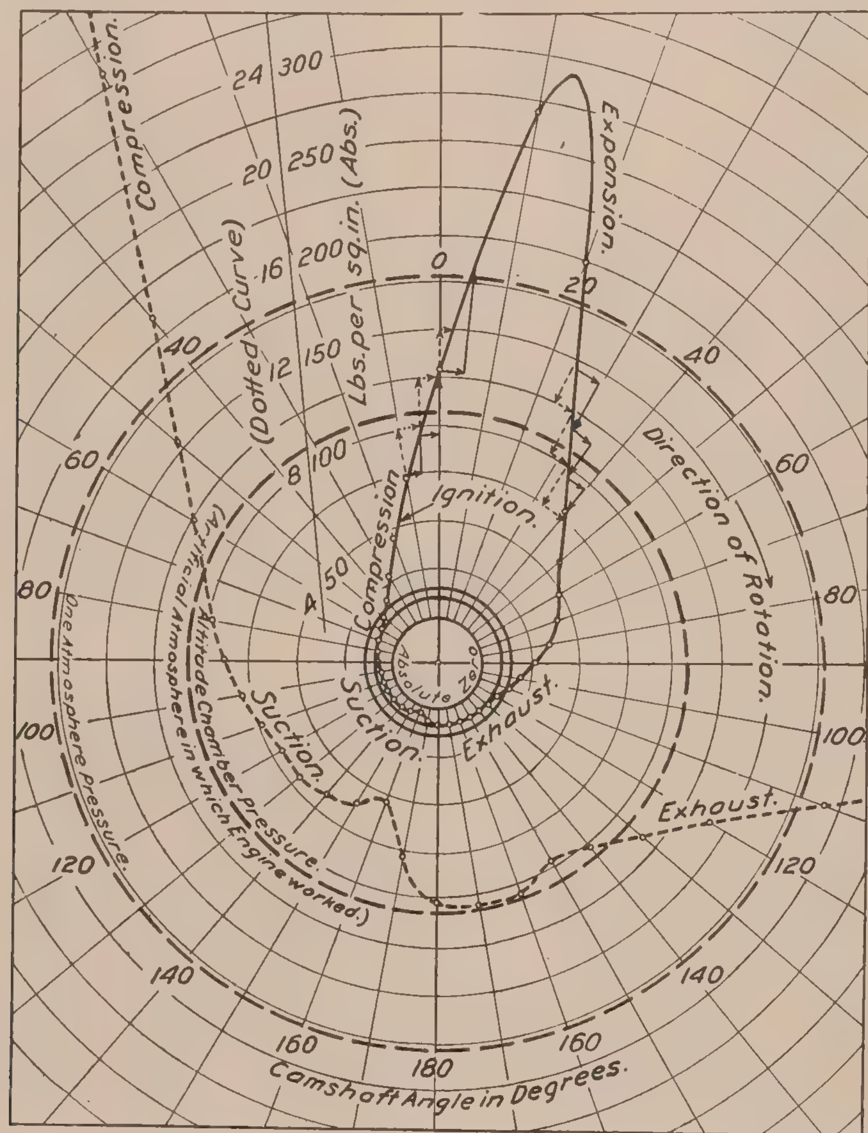


FIG. 9.—Specimen indicator diagram in polar coordinates, pressures *vs.* cam-shaft angle. The diagram is shown by the solid line, to which the right-hand pressure scale applies. A reproduction of that part of the curve near the pole is shown in the dotted curve, to which the $12\frac{1}{2}$ times magnified left-hand scale applies.

and alternate contacts only are homologous. The two homologous sets are represented by alternate clicks of the telephone, and the observer sets for disappearance of each alternate click with about the ease as for disappearance of all clicks. The double value of pressure corresponding to each angle is shown by figures 10 and 12, more fully explained in the next section.

The manipulation of the indicator outfit is as follows: The operator wears a pair of watch-case telephone receivers mounted in the usual switchboard head harness, and keeps his hands on two valves in the pressure line, one admitting compressed air from the reservoir and the other relieving the pressure to the atmosphere or aspirator. With the timer set, say, at zero crank angle, he watches the appropriate gauge and manipulates the valves according to his interpretation of the signals received in the telephones. If there appears to be an appreciable range of pressure over which the clicking is irregular, he estimates the mean of this range and makes a record of the gauge reading. The timer is then set to the next observing point

and the process repeated. The timer may be set and read by the same observer or by a second one. The process of mapping a complete indicator diagram advances so much more rapidly with two as to be desirable, so as to reduce to a minimum the likelihood of large changes in the conditions of engine performance, with the consequent lack of coordination of the earlier and later portions of the diagram.

A convenient record of the quantities observed is a polar graph with degrees of crank angle read directly from the timer and radius vector of pressure read directly from the gauge. This pressure-angle plot can be transformed easily to pressure time or pressure volume, as desired.

SPECIMEN INDICATOR DIAGRAMS.

Representative charts and data sheets are appended to illustrate the results obtained with the instrument. In all the charts two pressure scales are used, one sufficiently close to include the whole range on the sheet and one very much more open, at least 10 times magnified, upon which scale the pressures near zero are plotted to show characteristics quite indistinguishable on the other scale.

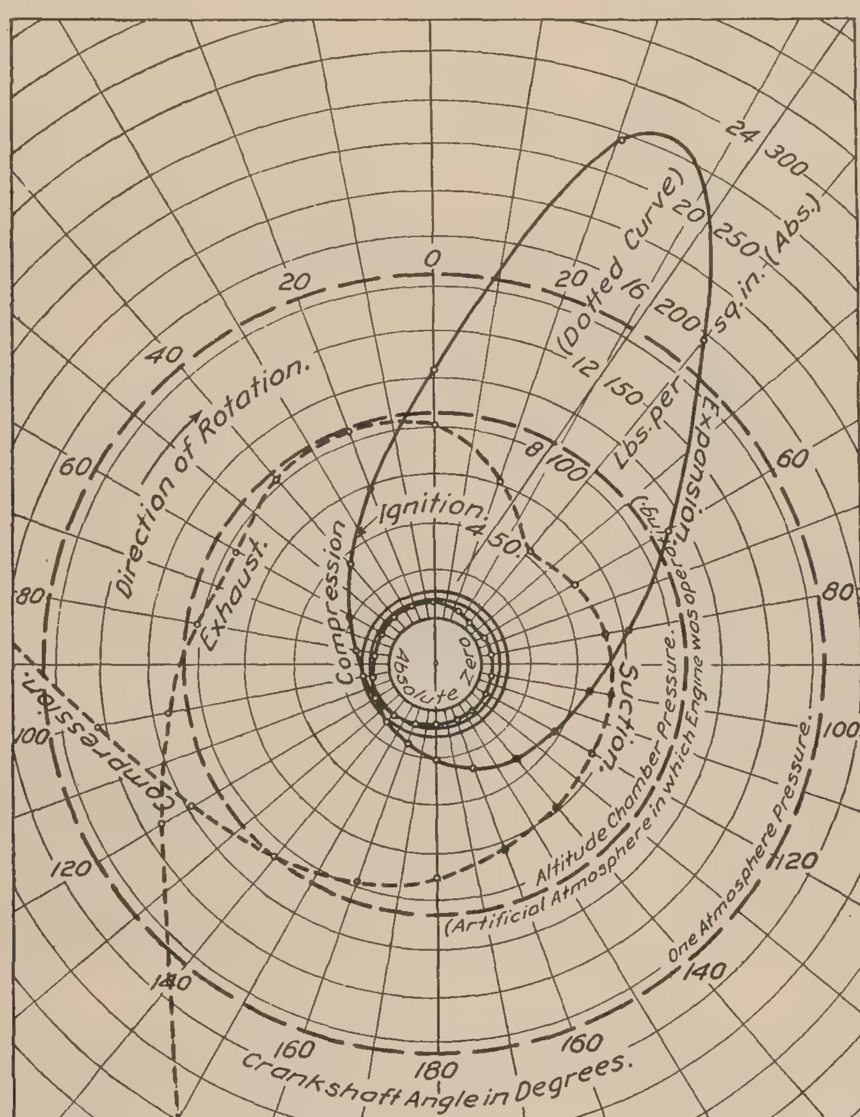


Fig. 10.—Specimen indicator diagram in polar coordinates, pressures *vs.* crankshaft angle. The diagram is shown by solid line, to which the right-hand pressure scale applies. A reproduction of the loop near the pole is shown in the dotted curve, to which the $12\frac{1}{2}$ times magnified left-hand scale applies.

Figures 10 and 12 show diagrams taken with the timer mounted on the crank shaft. It will be noted that the diagram loops twice around the pole, giving two pressure values for each value of crank-shaft angle. The method of securing such a diagram has been discussed in the previous section. The pressure time curves corresponding to figures 9 and 10 are shown in figures 11 and 12. A pressure volume diagram is given in figure 13. A diagram in which the logarithms of the pressures and volumes are plotted as suggested by Clayton,[4] is shown in

[4] A. S. M. E. Journal, Apr., 1912.

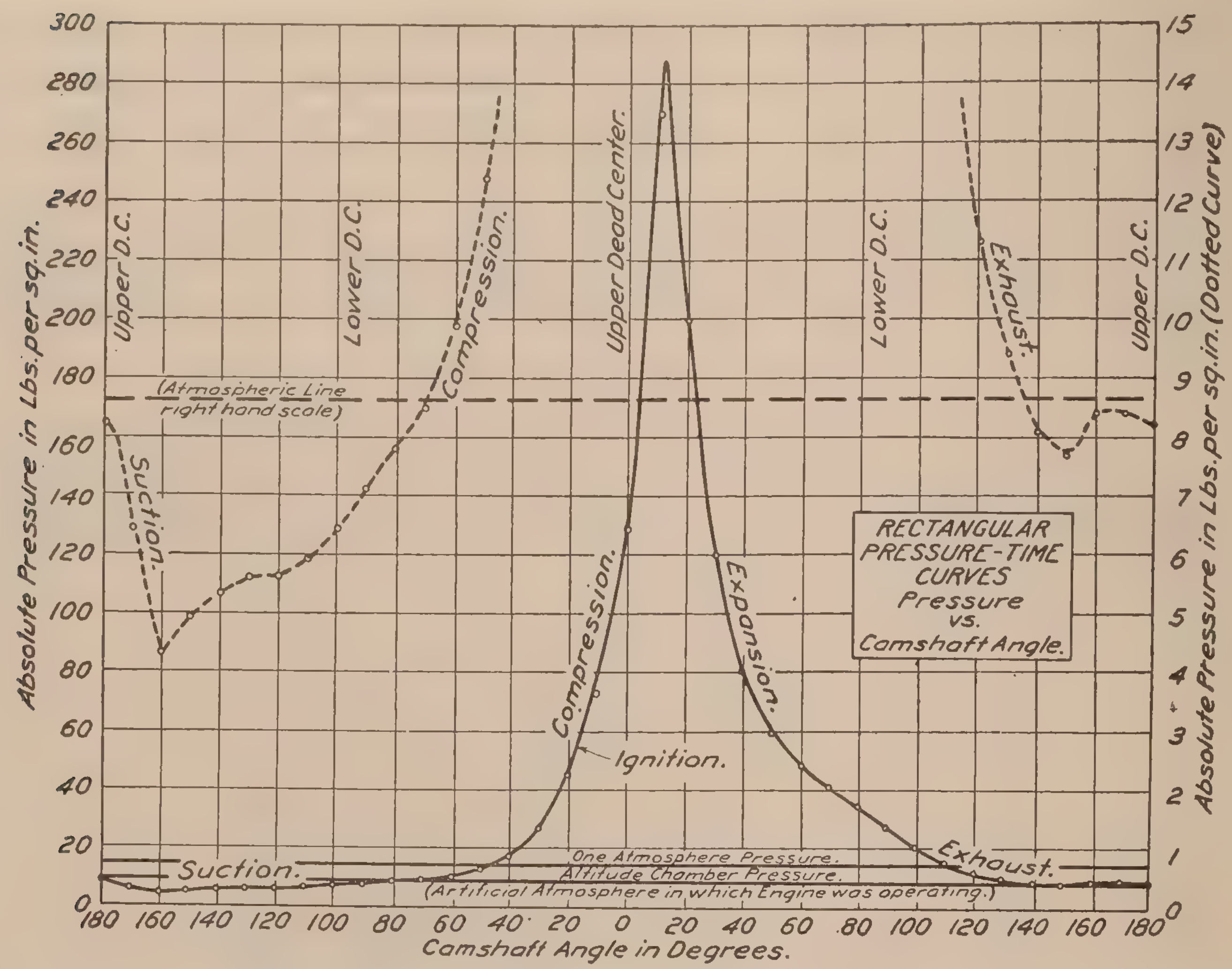


FIG. 11.—Specimen indicator diagram in rectangular coordinates, transformed from figure 9. The diagram is shown by the solid line, the corresponding pressure scale being numbered on the left margin. The portion of the diagram near zero pressure is reproduced in the dotted curve on a scale magnified 20 times, the scale being noted in the right margin.

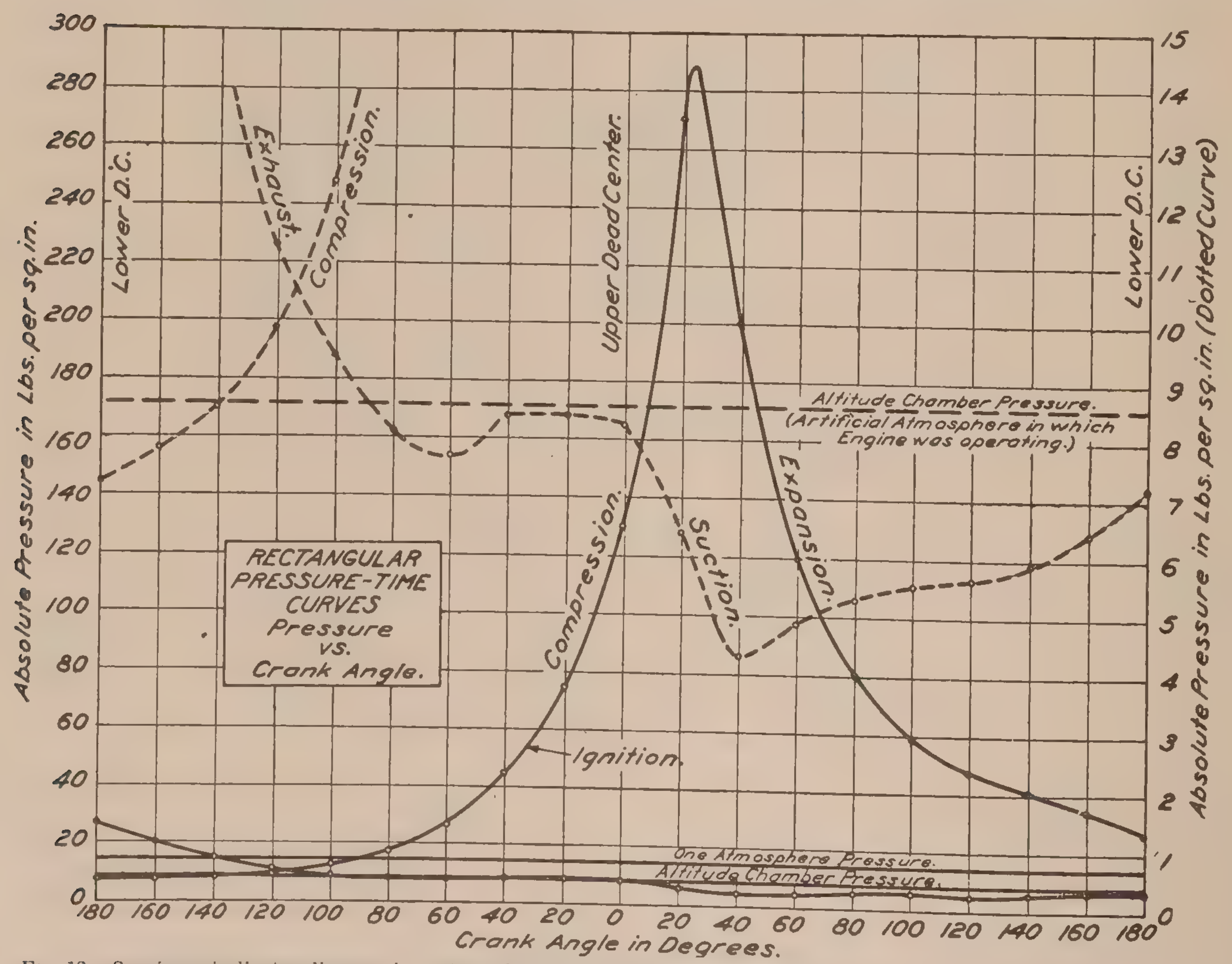


FIG. 12.—Specimen indicator diagram in rectangular coordinates, transformed from figure 10. The diagram is shown by the solid line, the corresponding pressure scale being numbered on the left margin. The portion of the diagram near zero pressure is reproduced in the dotted curve on a scale magnified 20 times, the scale being noted in the right margin.

figure 14. The straight portions of this diagram show the limits between which the gas obeys the following law: PV^n = constant. This plot facilitates the determination of the value of "n," which is the slope of the line.

DATA SHEET.

(Figs. 9 to 14.)

Pressure measurements made with high-speed engine pressure indicator of the thin diaphragm type.

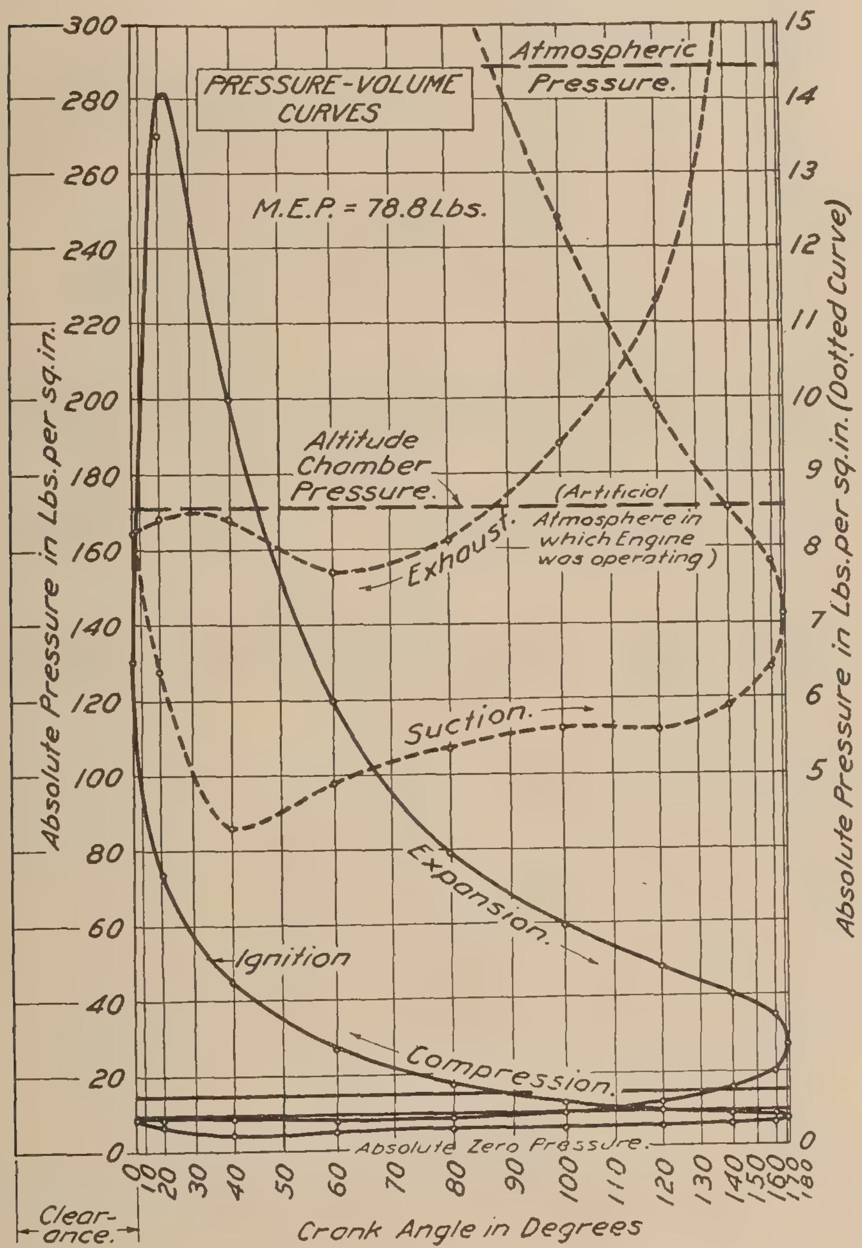


FIG. 13.—Specimen indicator diagram in the usual pressure volume scaling, transformed from data of Figures 9, 10, 11, and 12 (all from same observed data). The lower loop of the diagram is reproduced magnified 20 times, in the dotted curve, to which the scaling in right-hand margin applies.

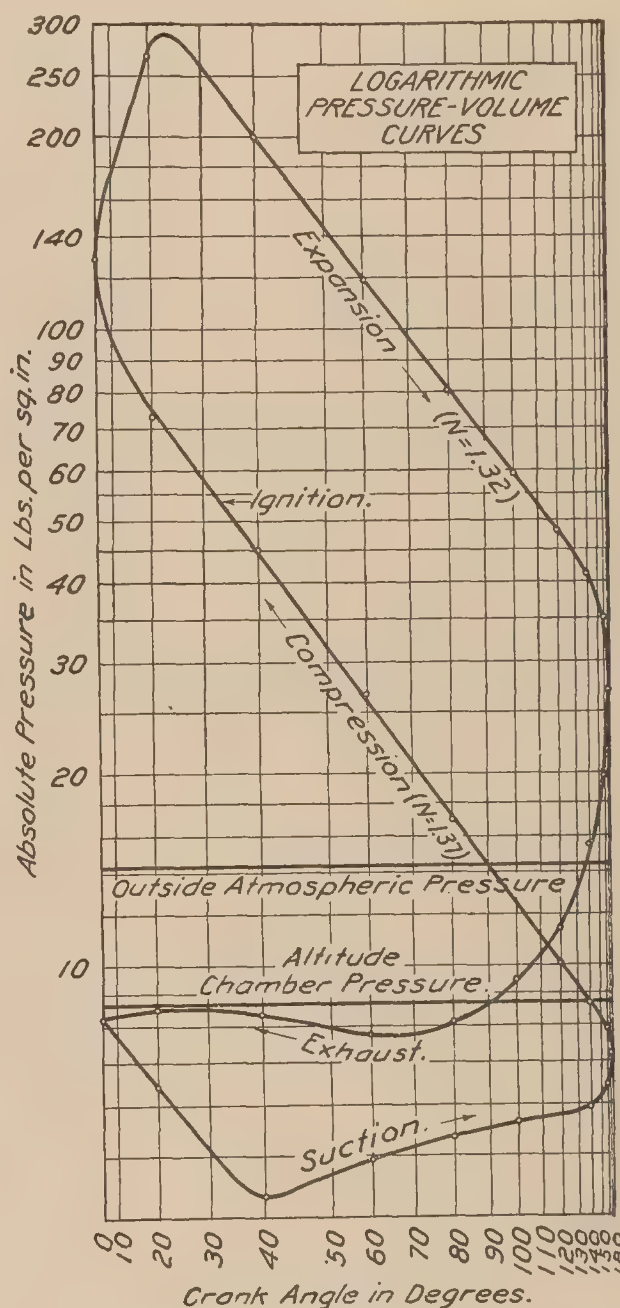


FIG. 14.—Indicator diagram of Figure 13 redrawn in logarithmic scaling. Lack of curvature of the principal parts of compression and expansion strokes shows validity of exponential relation pv^n=constant in representing them.

Hispano Suiza airplane engine running 2,200 revolutions per minute in altitude chamber simulating operating conditions at 15,000 feet altitude. Pressure of this artificial atmosphere 8.6 pounds per square inch.

Engine measurements:

	Inches.
Connecting rod length	8. 93
Bore	4. 72
Stroke	5. 12
Clearance (equivalent of volume in inches of stroke)	0. 97
Compression ratio	6. 3

Indicator zero, 0. 6 pound per square inch.

	Crank angle (°).	Observed pressures (pounds per square inch).		Absolute pressures (pounds per square inch).		Piston stroke—corrected for clearance (inches).	Crank angle (°).
	180	13.0	−6.6	27.0	7.2	6.08	180
	160	6.0	−6.0	19.8	7.8	5.99	160
	140	1.6	−5.3	15.4	8.5	5.64	140
	120	2.5	−3.9	11.3	9.9	5.09	120
	100	−1.4	−4.4	12.4	9.4	4.33	100
	80	−3.0	−5.7	16.8	8.1	3.44	80
	60	12.8	−6.1	26.5	7.7	2.53	60
	40	31.0	−5.4	45.0	8.4	1.72	40
	20	59.5	−5.4	73.5	8.4	1.18	20
	0	115.	−5.6	129.	8.2	.97	0
	20	225.	−7.4	255.	6.4	1.18	20
	40	185.	−9.5	200.	4.3	1.72	40
	60	105.	−8.9	120.	4.9	2.53	60
	80	66.0	−8.5	80.0	5.3	3.44	80
	100	45.0	−8.2	59.0	5.6	4.33	100
	120	34.0	−8.2	48.0	5.6	5.09	120
	140	27.0	−7.9	41.0	5.9	5.64	140
	160	21.0	−7.4	35.0	6.4	5.99	160
	180	13.0	−6.8	27.0	7.0	6.08	180
Maximum pressure: Limits	....	280–230	....	294–244			
Maximum pressure: Average	....	260	....	274			

Readings made to the nearest 0.1 pound per square inch from absolute zero to 20 pounds per square inch gauge; to the nearest 0.5 pound per square inch from 20 to 100 pounds per square inch gauge; and to the nearest 1 pound per square inch for pressure over 100 pounds per square inch.

REPORT No. 108

SOME FACTORS OF AIRPLANE ENGINE PERFORMANCE

By VICTOR R. GAGE
Bureau of Standards

REPORT No. 108.

SOME FACTORS OF AIRPLANE ENGINE PERFORMANCE.

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RÉSUMÉ.

This report was prepared for the National Advisory Committee for Aeronautics and is based upon an analysis of a large number of airplane engine tests made at the Bureau of Standards. It contains the results of a search for fundamental relations between the many variables of engine operation.

The data used came from over 100 groups of tests made upon several engines, primarily for military information. The types of engines were the Liberty 12 and three models of the Hispano-Suiza. The tests were made in the altitude chamber, where conditions simulated altitudes up to about 30,000 feet, with engine speeds ranging from 1,200 to 2,200 r. p. m. The compression ratios of the different engines ranged from under 5 to over 8 to 1. The data taken on the tests were exceptionally complete, including many pressures and temperatures, besides the brake and friction torques, rates of fuel and air consumption, the jacket and exhaust heat losses.

With the Liberty engine, operating at from 500 to 2,000 r. p. m. and with the Hispano-Suiza 300 h. p. operating from 1,400 to 2,200, it is found that the friction torque increases approximately as a linear function of engine speed at a given air density, and approximately as a linear function of density at a constant speed. This means that the friction horsepower increases approximately as the square of the speed. Actually the relation of torque and speed is such that the friction horsepower increases with speed raised to a power between the first and second, this power increasing with speed, approaching the square. The relation depends upon the engine design and speed and density of the air. Any statements as to the distribution of the friction losses are based upon incomplete evidence; the indications are, however, that the pumping losses are about half of the total friction.

There is no doubt that for a given process of combustion and at a constant speed the engine power is directly proportional to the weight of charge supplied; in other words, proportional to the charge density at the beginning of compression. As a consequence, if operating conditions are sensibly constant except for altitude, the engine power will be closely proportional to the air density. The volumetric efficiency increases with increase of air temperature at constant pressure, so that power does not decrease as fast as the air density when the temperature is raised, due to changes in vaporization and heat transfer.

In order to compare the action of the gasoline engine with the theoretically perfect heat engine operating on the same cycle, it is necessary to base the heat balance of the actual engine upon the heat actually made available by combustion, not upon the heat supplied in the fuel. By summing up the exhaust and jacket heat losses with the brake power, an approximation is made of the true heat available, accurate to within perhaps 5 per cent. Basing the heat balance upon the heat thus accounted for, it is fairly well established that the energy distribution is not appreciably altered by change of either altitude or speed. It is, of course, altered by compression ratio changes. The exhaust heat, as per cent of the heat accounted for, is practically the same as the theoretical rejection of heat computed for the same compression ratio.

I. INTRODUCTION.

This report was prepared for and by the direction of the National Advisory Committee for Aeronautics, and is an analysis of a large number of tests made in the altitude laboratory of the Bureau of Standards. Many of the tests used in this report were made upon engines supplied by the Aviation Section, Signal Corps, United States Army, and the Army Air Service, Engineering Division, McCook Field. Much of the work was made possible by the hearty cooperation and aid of the National Advisory Committee for Aeronautics, the Army Air Service, the Navy Department, and the Bureau of Mines.

The very nature of this report depends upon data taken during a long period of time, under the supervision of Messrs. W. S. James and S. W. Sparrow. Dr. H. C. Dickinson and D. R. Harper, 3d, contributed much to the value of the paper. Mr. H. S. White aided in interpreting the various test conditions. Dr. Donald MacKenzie helped by his interest in working up the results. The author wishes to acknowledge his indebtedness to these, and to others with whom he was associated during this work.

The purpose of this study was to glean any general information which could be found from a comparison of many groups of tests, each group having been made for some special purpose, generally for military information. The tests here considered were made upon several engines of the Liberty 12-cylinder and the Hispano-Suiza 8-cylinder type. Various compression ratios were used. All of the tests were made in the altitude chamber, and the determinations included brake torque, rates of fuel and air consumption, jacket and exhaust heat losses, as well as pressures and temperatures at many points in and about the engines. These tests were made at several speeds and with air pressures corresponding to different altitudes, but generally with a constant air temperature. With some exceptions, the mixture ratio and time of ignition were adjusted to give maximum power, as most of the tests were made in the war period, when maximum power was always the goal. Certain operating conditions have been selected as laboratory standards, such as air supply temperature, although the standard jacket water temperature is not the same for the Liberty as for the Hispano-Suiza. Some later tests have been made to determine the effect on horsepower and general performance of special variations, such as of mixture ratio, air temperatures, jacket temperatures, different oils, etc. Generally, but not always, some means was provided for reducing the gasoline supply sufficiently to secure a proper mixture ratio at the extreme altitudes. An auxiliary device of this kind is necessary because the altitude adjustment on stock carburetors is inadequate above about 45 cm. barometer (15,000 feet). Complete determinations of friction torque at various speeds and altitudes were made as a part of the later tests. During these friction runs no fuel was supplied, and there was no ignition spark passing in the cylinders, but otherwise the operating conditions of air, oil, water, etc., were the same as in the power runs. On the earlier tests not enough power was available to operate the larger engines at normal speeds, so the friction runs cover only the lower speeds.

In any analysis of engine performance the change of indicated power rather than brake power is the fundamental relation to be considered, and the first step in obtaining the indicated power is to study the friction losses, which are the subject of Part II. By adding the friction and brake torques the indicated torque is obtained and its relation to the engine speed and air density are studied in Parts III and IV. Other variables introduce so many complications that it is necessary to first obtain a general idea of the relations of torque, speed, and density with other conditions fairly constant, as in Parts III and IV, before taking up the effect of other variables. Two of the latter, viz, air-supply temperature and mixture ratio, are considered in Parts V and VI, respectively.

An internal-combustion engine is essentially a "heat engine," and a knowledge of the distribution of the energy supplied to it is desirable. The energy which might be obtained by the combustion of the fuel should be considered in two parts, viz, that which is not developed by the combustion as it takes place in the engine and that which is so developed and rendered available for transformation into useful work. The latter must then be further divided into

the two parts of that actually transformed to work and that rejected as heat. In this manner a logical comparison can be made between the actual engine performance and that to be expected from the theoretical cycle.

II. FRICTION LOSSES.

INTRODUCTORY REMARKS.

Brake power is the ultimate and practical measure of the usefulness of an engine. Indicated power, or brake plus friction, is the real measure of the performance of an engine. Friction losses are a necessary evil, and must be reduced to as small a quantity as is practicable. For intelligent improvement of mechanical efficiency, it is important to know the relative magnitude of the various factors which make up the friction losses. The study of friction losses is here made for these two purposes, namely, (1) as a means of obtaining the indicated power, and (2) to learn as much as possible concerning the relative distribution of the various items which compose the "friction."

In testing airplane engines in the altitude laboratory of the Bureau of Standards, each engine is usually put through a "friction run," during which the ignition and fuel are shut off and the engine is turned over by operating the electric dynamometer as a motor. The torque required to turn the engine over is measured at each of several speeds and altitudes.

During the friction runs the air, oil, water, and general engine temperatures and conditions are maintained as nearly as possible the same as in the power runs. Unless otherwise specified, all friction runs were made with throttle fully opened.

The friction losses of the Liberty 12-cylinder aviation engines were obtained from altitude laboratory tests Nos. 141 to 144, 146, 148, 152, 153, and 160. The engines used in these tests had compression ratios of approximately 5.6 to 1, except in tests Nos. 152 and 153, where it was 7.2, and in test No. 160, where it was 5.4. The friction losses of the Hispano-Suiza, model H, 300 h. p., 8-cylinder engine were obtained from altitude laboratory tests Nos. 161 and 162, on an engine with compression ratio of 5.3.

METHODS AND DETAILS OF HANDLING TEST DATA.

The data used in obtaining the relations between friction mean effective pressure and speed or density were selected for about normal operating conditions. In tests where conditions were abnormal the data were rejected, or, if plotted, used solely as a qualitative indication. The air temperature was generally maintained constant, so, for convenience, the barometer has been used as equivalent to density. The friction mean effective pressure values were grouped and plotted in two ways—(1) for nearly constant density plotted versus r. p. m., and (2) for nearly constant speed plotted versus density, thus accomplishing the same result as plotting to the three dimensions of f. m. e. p., r. p. m., and density. Figures 1–H and 2–H are examples of this method. In locating the faired curves all three dimensions have to be considered, as, for example, in figure 1–H the curve for f. m. e. p. versus r. p. m. at 64 cm. barometer is placed above the average of the points because in figure 2–H the f. m. e. p. values at this barometer are evidently too small at nearly all speeds.

Figures 1–L, 1–H, and 2–H show the original data upon which the friction results and conclusions are based, the letter L after a figure number signifying Liberty engines, H, Hispano-Suiza engines. The subsequent figures 3 and 4 are reproduced from the faired curves of figures 1–L, 1–H, 2–H, and the omitted 2–L curve, but collected upon one sheet and with the points omitted. The curves have been extrapolated for speeds below 600 r. p. m. in figure 4–L and below 1,400 r. p. m. in figure 4–H. The curves have also been extrapolated for barometers less than 30 cm. in figures 3, except for one estimate based on a closed throttle run on the Hispano-Suiza (fig. 2–H, 1,800 r. p. m.). In figure 3 the curves have been extrapolated to their intercepts on zero barometer; the estimate of pumping losses presented in figures 5 and 6, and discussed under a separate heading, being based upon the intercepts thus found. These inter-

cepts have also been used to estimate the friction of the engine exclusive of the pumping work, presented as the zero barometer curves on figure 7, f. h. p. versus r. p. m.

Only one set of friction runs wherein the engine was throttled was available. This was made upon a Hispano-Suiza engine, 300 h. p., and is shown in figure 2–H, 1,800 r. p. m. Such results were converted to approximately the equivalent value with open throttle by subtracting from the observed throttled friction m. e. p. the change of pressure difference between the exhaust and intake manifolds. This, on an indicator card, would amount to placing the suction line across the full length of the card, at the manifold pressure under open throttle conditions. As the suction line is in fact less than the full length of the card, and as the manifold pressure is greater than the suction pressure in the cylinder, it is possible that the net result of this approximation may not be far from the truth. There are also unknown changes of area of the lower loop of the card, due to change of compression line, of fuel vaporization, etc., with change of throttle position. The above method has been used in a later chapter (see fig. 4, Part III) for converting the mean effective pressure at throttle to an equivalent value at open throttle, and gives results which correspond, within experimental limits, with the mean effective pressures which were obtained with the same weight of air when the throttle was open, and with the air at reduced density. When no altitude chamber is available this method may be used to approximate the power output of an engine at altitude from throttled runs.

ESTIMATE OF PUMPING LOSSES.

By extending the curves of friction versus barometer to zero density, as on figures 3, intercepts are obtained which may and probably do fairly accurately represent the friction mean effective pressure of the engine when there is no air to pump. The difference between the friction losses at zero density and those at another density, but at the same speed, is an approximation of the pumping loss. The pumping loss thus derived includes any effects of change of gas pressure upon piston friction, but later tests have shown that these are practically negligible.

The mean effective pressure required to pump the charge apparently varies directly with the air density at a given engine speed, as shown on figures 5, agreeing with the hypothesis that the weight of air handled at constant speed, and hence the pumping work, should vary directly with the density.

The proportion of the total friction which is used in overcoming the pumping losses is shown for the Liberty engine on figure 6–L, and for the Hispano-Suiza 300 h. p. engine on figure 6–H. It is evident that the two engines have very different pumping characteristics.

CONCLUSIONS RESPECTING FRICTION LOSSES.

An empirical and approximate statement, applicable to the speed and density ranges herein covered, is that the friction torque (m. e. p.) increases as a linear function of speed at a constant air density, and as a linear function of air density at a constant speed. This applies to both the Liberty 12-cylinder and the Hispano-Suiza 8-cylinder engines and is shown on figures 3 and 4. Actually, the true relations of friction torque, engine speed, and air density are much more complex and can not be deduced from the available information. It is known, however, that the friction torque is such that the friction horsepower will increase with engine speed raised to between the first and second power, approaching the square of the speed at the higher speeds, figure 7. The exact relation is dependent on the engine, as well as on the speed and air density. Also, at a given speed, the friction horsepower increases with some varying power of the air density, because of the change of the pumping work. The friction horsepower at constant speed is found to increase slightly more rapidly than does the density.

No systematic change of friction could be connected with change of compression ratio, from the data used in this report, although a slight change is to be expected.

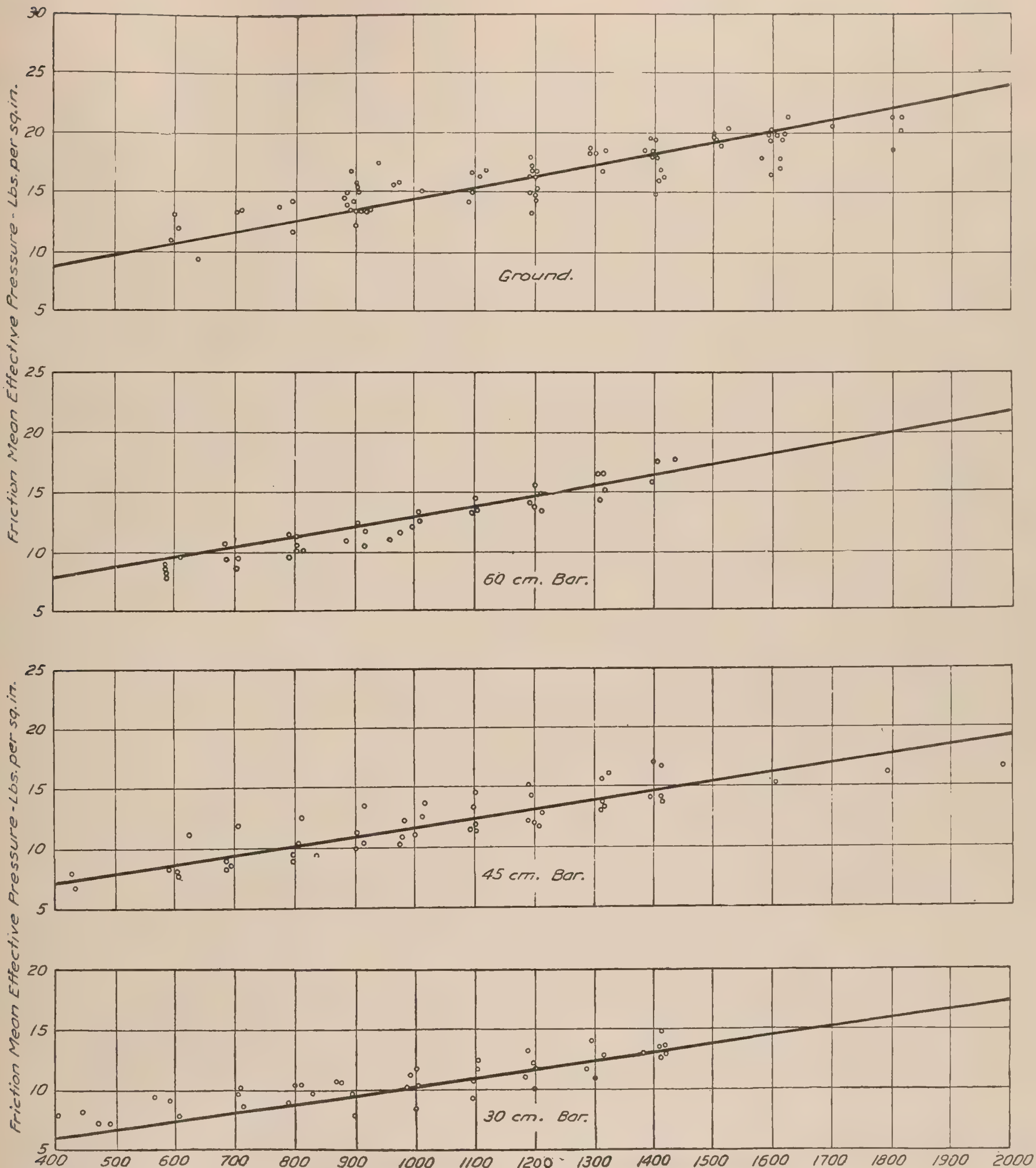

Friction Mean Effective Pressure - Lbs. per sq. in.
Ground.
60 cm. Bar.
45 cm. Bar.
30 cm. Bar.
R. P. M.

Fig. 1-L.

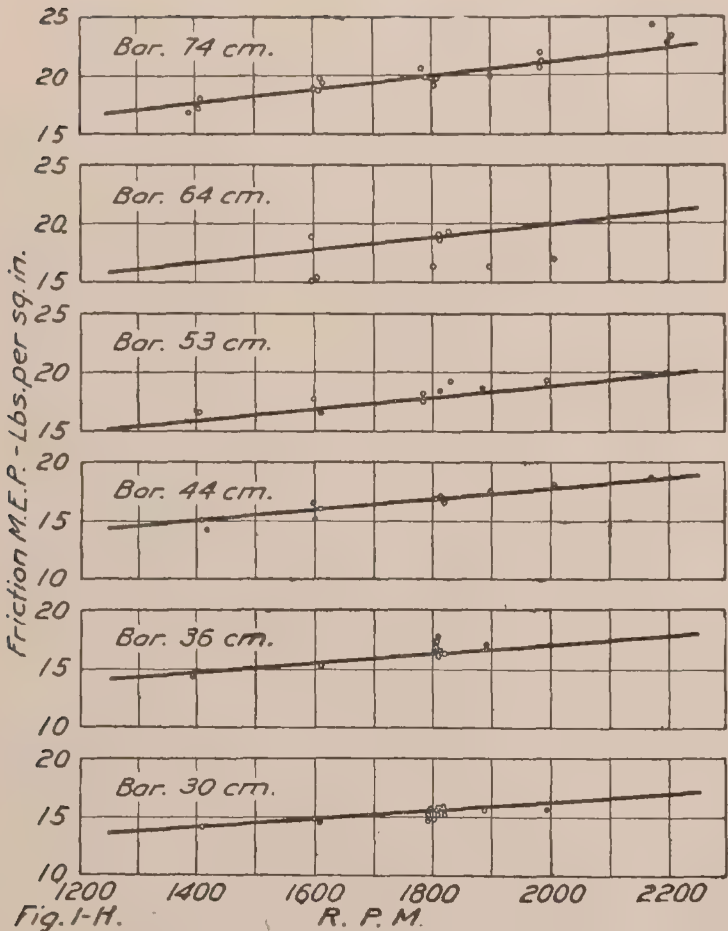


Fig. 1-H.

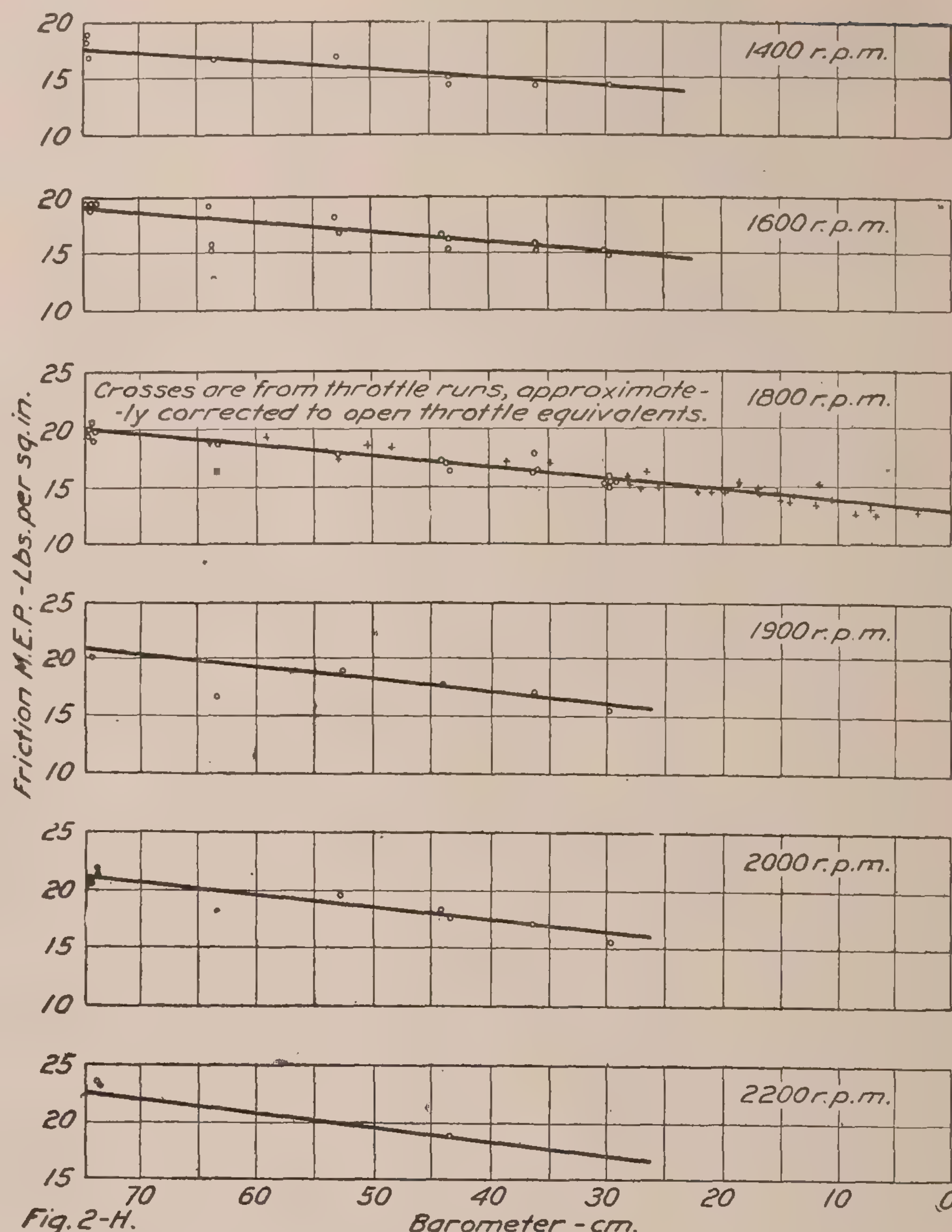


Fig. 2-H.

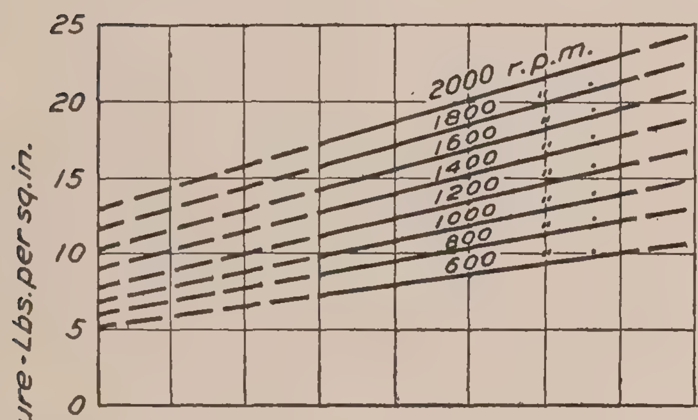


Fig. 3-L.

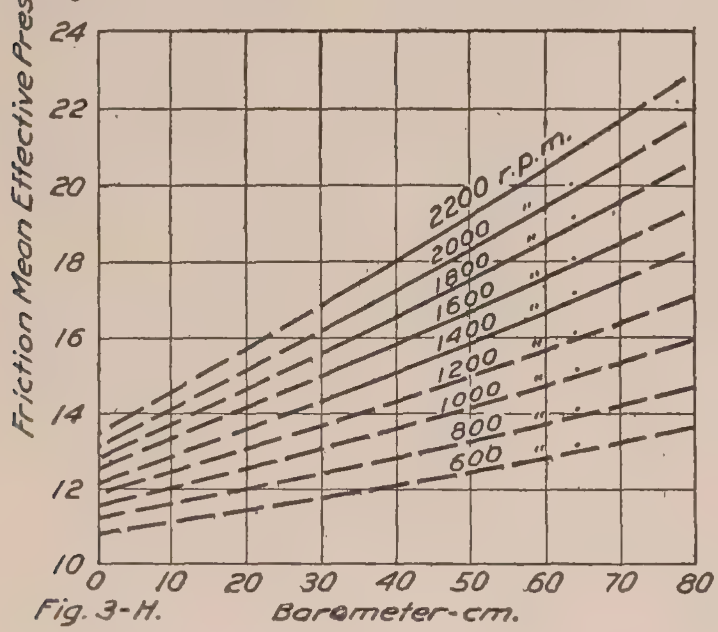


Fig. 3-H.

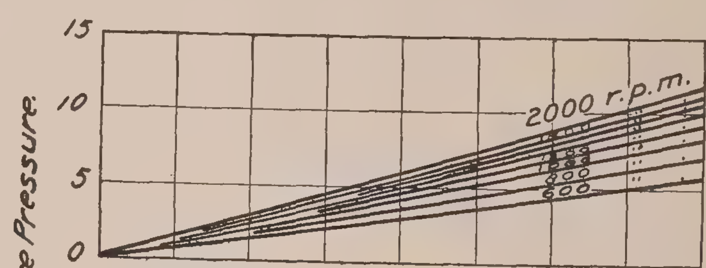


Fig. 5-L.

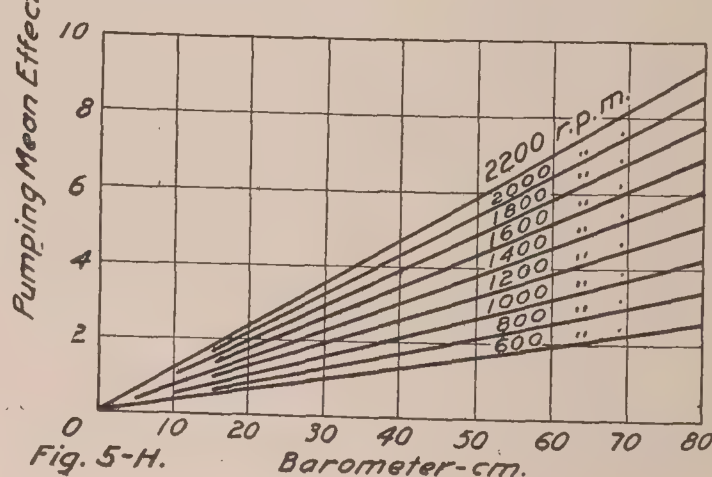


Fig. 5-H.

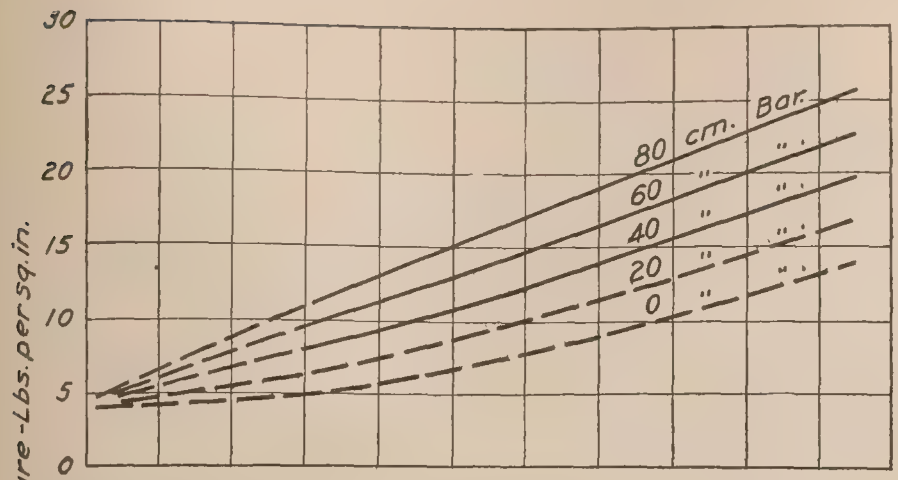


Fig. 4-L.

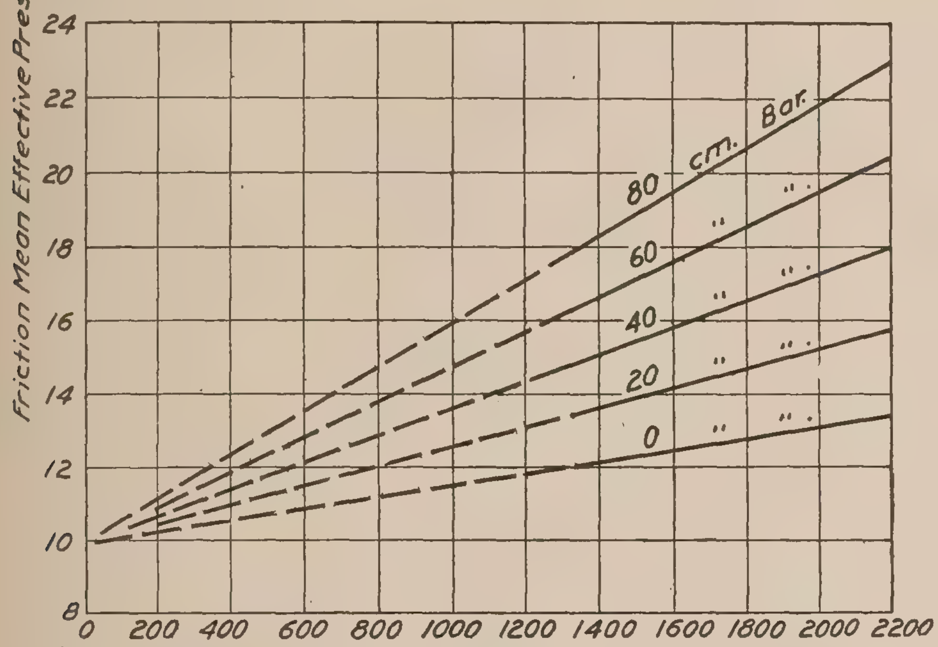


Fig. 4-H.

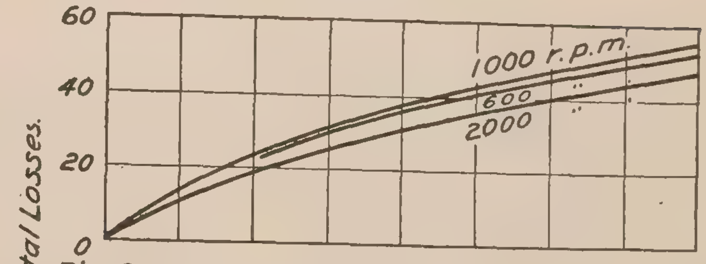


Fig. 6-L.

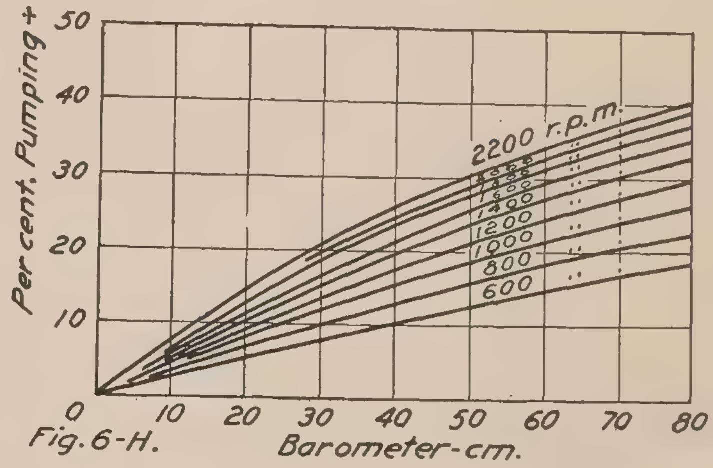


Fig. 6-H.

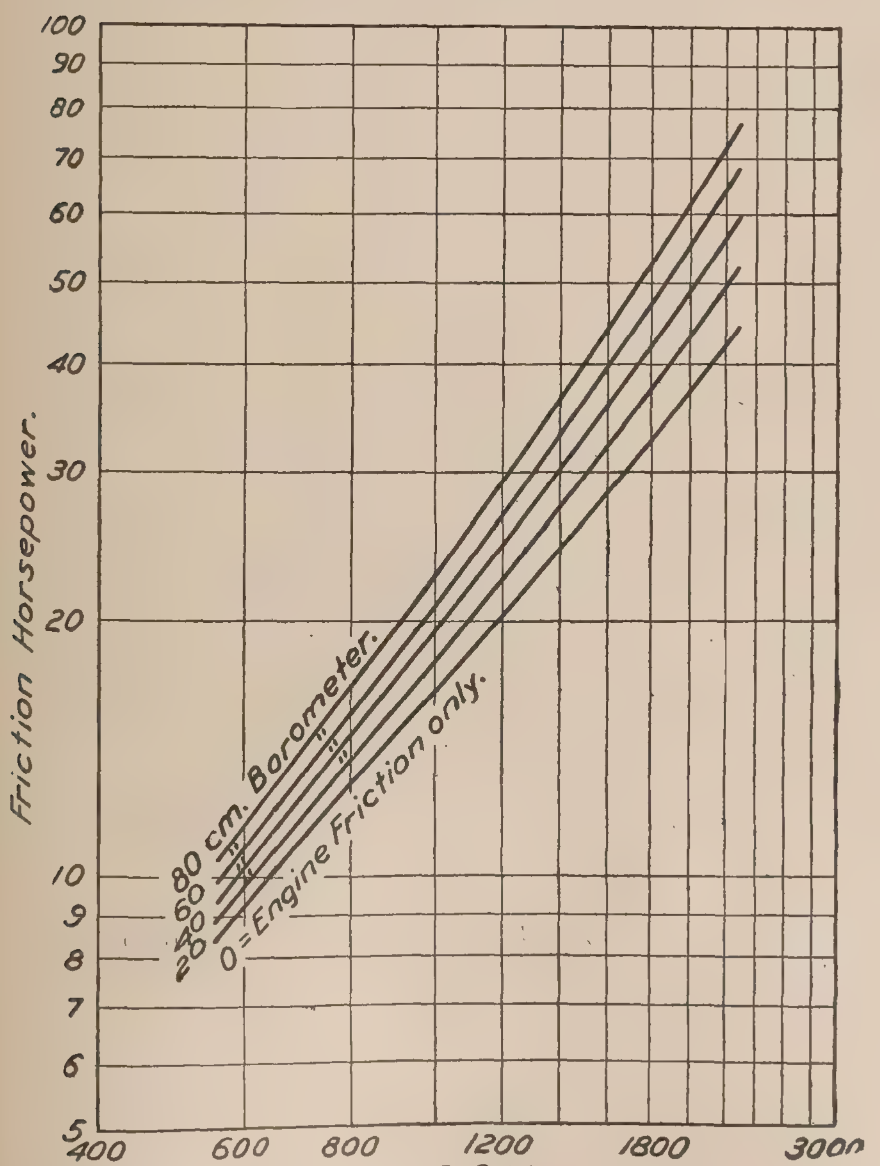


Fig. 7-H.

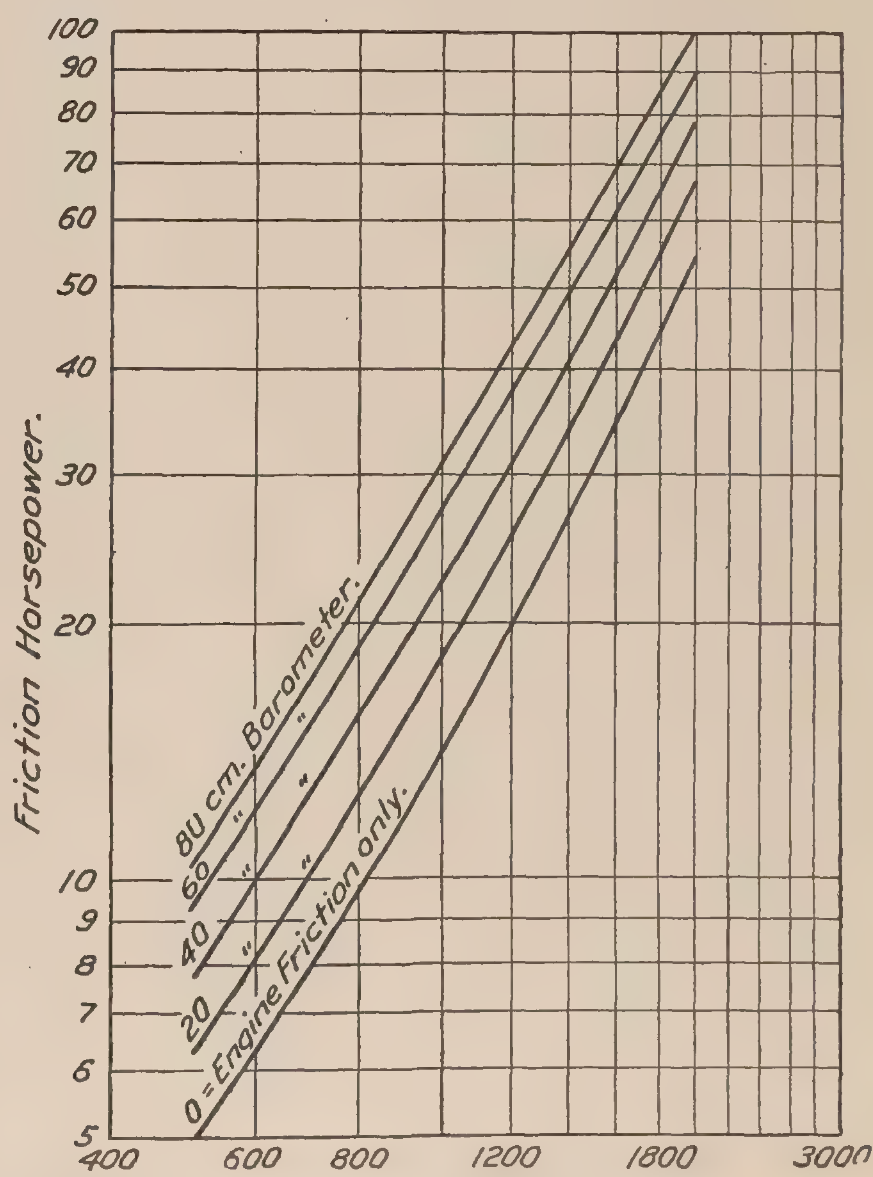


Fig. 7-L.

III. EFFECT OF DENSITY OF AIR UPON POWER.

INTRODUCTORY REMARKS.

The indicated power of a four-stroke cycle internal combustion engine may be defined as the power developed inside the cylinders derived from the combustion of the fuel, and is obtained by adding the friction power (including pumping losses) to the brake power. Change of piston friction due to the increased pressures in the cylinder when the engine is operating seemed to be the most probable source of error when determining the indicated power by this method. However, a consideration of all the evidence, including actual indicator diagrams, leads to the conviction that indicated power is correct when obtained from brake plus friction.

CHANGE OF INDICATED MEAN EFFECTIVE PRESSURE WITH BAROMETRIC PRESSURE.

For a given engine the indicated horsepower may be expressed as i. m. e. p. times speed times a constant. It is shown later that the work obtainable from unit weight of air is independent of speed but is dependent upon conditions of operation, for example, mixture ratio. Hence, at a given density, variation in i. m. e. p. must be due either to a difference in the amount of air taken in per stroke or to a difference in the degree to which it is utilized. Figures 1 and 2 are presented, as a composite photograph, to show the magnitude of such variations under the conditions noted.

Values of brake mean effective pressures used for computing indicated mean effective pressure on figure 1 are from tests on Liberty 12-cylinder aviation engines with 5.4, 5.6, and 7.2 compression ratios (altitude laboratory tests, Nos. 144, 147, 155, 156, 157, 159, and 160). Similar values used on figure 2 are from tests on a Hispano-Suiza, model H, 300 h. p. engine with compression ratio of 5.3. (Tests Nos. 161 and 162.) These engines were operated at speeds of from 1,200 to 2,000 r. p. m. for the Liberty and from 1,400 to 2,200 r. p. m. for the Hispano-Suiza. The barometric pressure ranged from ground (75 cm.) to that corresponding to about 25,000 feet altitude (30 cm.). Only wide-open throttle runs were used on figures 1 and 2. In all cases the fuel was an aviation gasoline which meets the specifications of the Aircraft Production Board for export to the American Expeditionary Forces.[1]

The indicated mean effective pressure versus altitude curves of the two engines, figures 1 and 2, are not comparable above about 15,000 feet (45 cm. barometer), because the Liberty engine tests, except No. 160, were made with special altitude adjustment on carbureter, whereas the Hispano-Suiza tests were made to determine the engine performance when equipped with stock carbureters without special altitude adjustment.

CHANGE OF INDICATED MEAN EFFECTIVE PRESSURE WITH DENSITY AT BEGINNING OF COMPRESSION.

The factor which determines the weight of charge in a given engine cylinder is the density at the beginning of compression. This density is a function of the pressure and the temperature of the gases at this time. The volume occupied by the charge is the volume of the cylinder less the volume occupied by the residual gases held in the clearance space. (At normal speeds the exhaust valve is supposed to be closed in time to prevent the escape of incoming charge, and the end of the charging period and the beginning of compression may be considered as coincident.)

The pressure and temperature of the charge at the beginning of compression and the volume occupied by the products of the previous combustion are unknown. However, certain well justified assumptions can be used in connection with measured quantities to compute the approximate density of the charge at the beginning of compression. This method will be described later.

[1] For properties and distillation curve of this fuel, see Report No. 47 of the Fourth Annual Report of the National Advisory Committee for Aeronautics, Power Characteristics of Fuels for Aircraft Engines, p. 6 and Plot 1.

Some of the tests included propeller load runs in which the engine was throttled. It was considered of interest to investigate the relative change of mean effective pressure in the two methods of reducing the density, one by changing the operating pressure surrounding the engine, and the other by throttling the incoming charge. When the engine is throttled the suction pressure is caused to fall considerably below the atmospheric pressure, with the result diagrammatically indicated on figure 4 as the change from the solid to the dotted lines, the exhaust pressure remaining substantially in the same relation with respect to atmospheric pressure. When the density is equally reduced by increase of altitude, the changes in exhaust, suction, and atmospheric pressures are such that all three remain in about the same relation as before the change of altitude was made. In order to approximate the equivalent indicated mean effective pressure at open throttle from tests with the throttle partly closed, the value of friction mean effective pressure at open throttle was increased by the mean effective pressure representing the increase in the lower loop of the indicator diagram, figure 4, and this total friction was added to the observed brake mean effective pressure at partially closed throttle positions.

The density of the charge at the beginning of compression was approximated by computations based on the following premises: (1) The total weight of charge per cylinder is the sum of the weights of air and of gasoline supplied per cycle. (2) The total volume of the charge at the beginning of compression may be considered as the total cylinder volume less the volume occupied by the residual gases expanding from clearance volume and exhaust pressure to the suction pressure. (3) The density at the beginning of compression is the weight of charge per cylinder per cycle (1) divided by the volume of charge per cylinder per cycle (2).

In detail, the assumptions made in order to compute the data for plotting figure 3 include the following:

(1) That the exhaust pressure in the cylinder is atmospheric pressure.

(2) That the suction pressure inside the cylinder, *a* or *b* on figure 4, is the pressure measured in the intake manifold close to the intake valves (actually this intake pressure as measured has too great an absolute value and would lie above the suction pressure on the card, thus indicating a smaller lower loop than would be shown by an indicator).

(3) That the exhaust gases filling the clearance volume of the cylinder would expand from the exhaust pressure to the suction pressure according to the relation $PV^{1.3}=C$.

(4) That the corners of the lower loop card were sharp, as in figure 4, and not rounded as they are in fact. (This assumption gives too large an area for the lower loop; assumption 2 gives too small an area, so the errors tend to neutralize each other.)

(5) That all of the fuel supplied was completely vaporized at the beginning of compression.

The data used in studying the change of indicated mean effective pressure with density at the beginning of compression, shown on figure 3, were from a test (No. 160) which included both open throttle and "propeller" or throttle runs, and which was made upon a Liberty 12-cylinder aviation engine with 5.4 compression.

CONCLUSIONS.

Change of indicated mean effective pressure can be taken as directly proportional to change of density of the air supplied when the conditions of operation are constant, except for density changes produced by pressure, and of the best. When the conditions are not the best, the power usually drops off slightly faster than the density. One of the essential conditions for best operation is that the mixture of air and gasoline be in proper portions. The carburetors ordinarily supplied with the engines have insufficient altitude adjustment, so the mixture is too rich above about 15,000 feet (45 cm. barometer).

The change from 5.4 to a 7.2 compression increases the mean effective pressure at a given density, although the change is not very obvious on figure 1. The data are too incomplete to warrant a statement of the amount of increase, as the 7.2 compression engine was tested only at altitudes of 15,000 and 25,000 feet (45 and 30 cm. barometers). Increase of mean effective

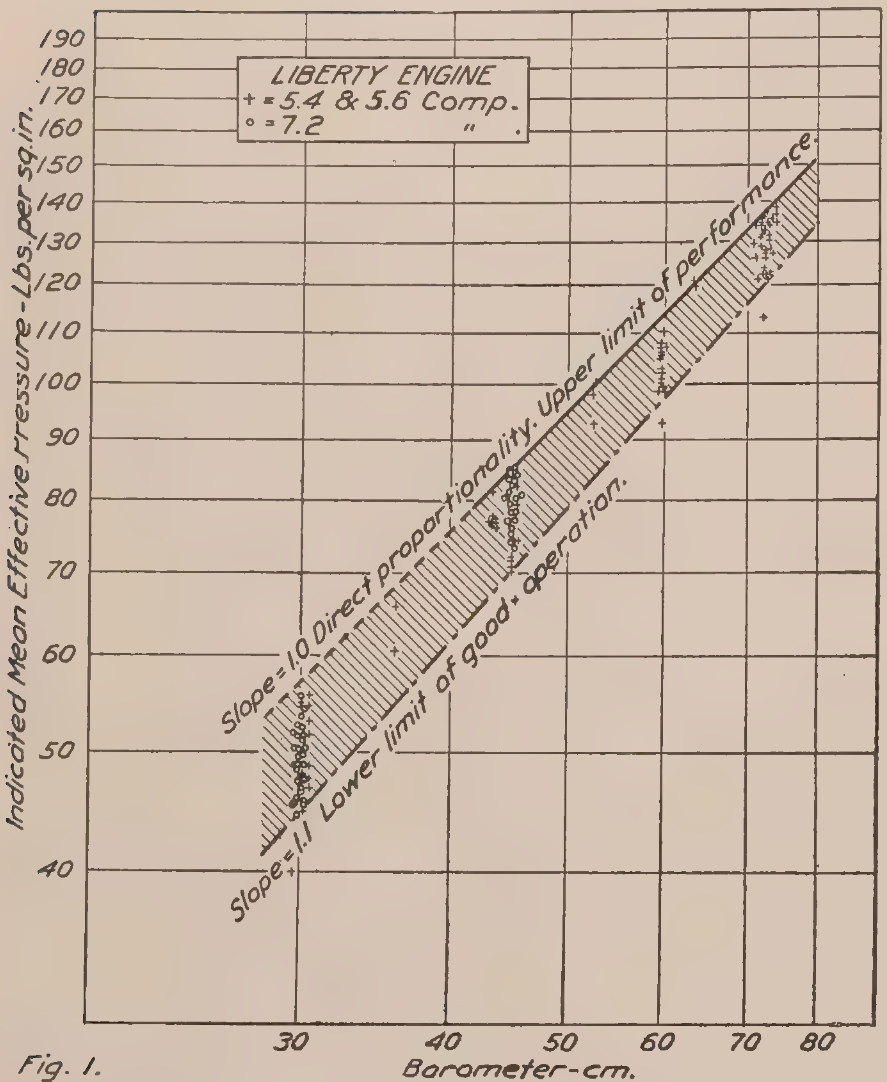


Fig. 1.

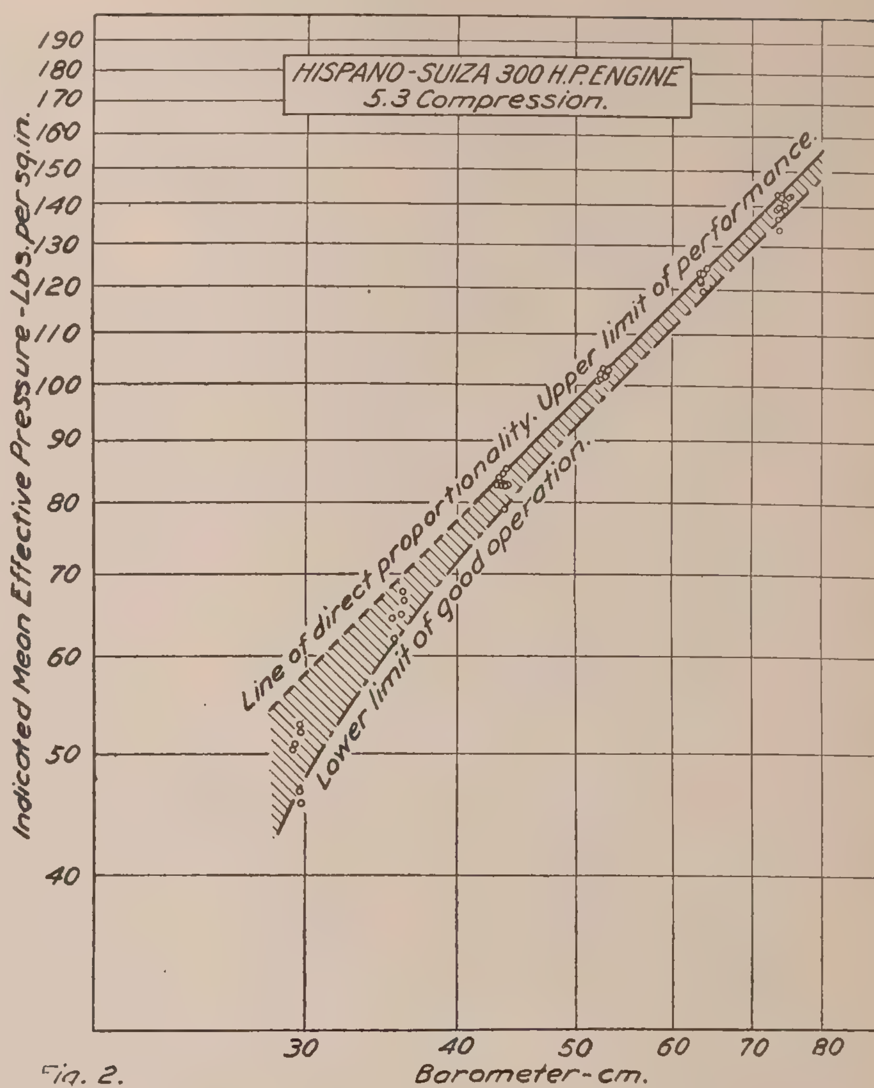


Fig. 2.

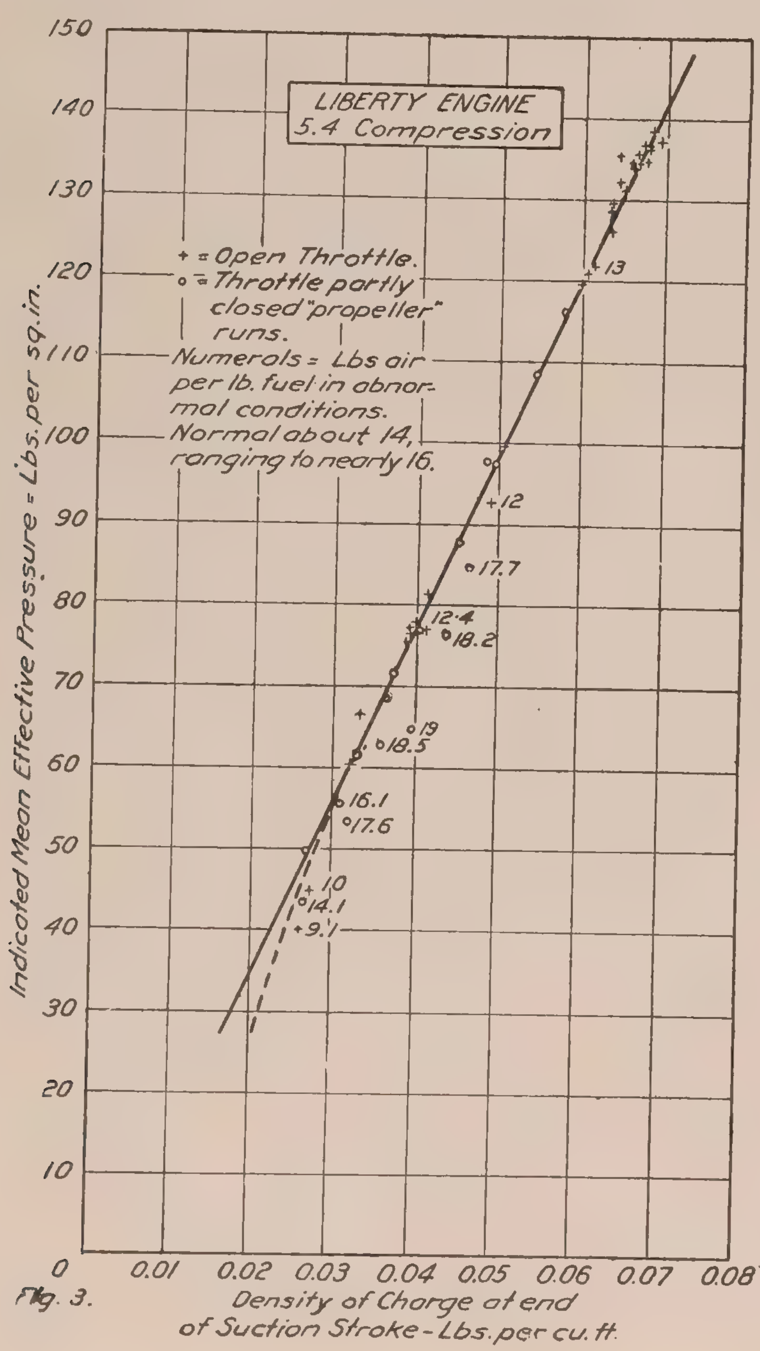


Fig. 3.

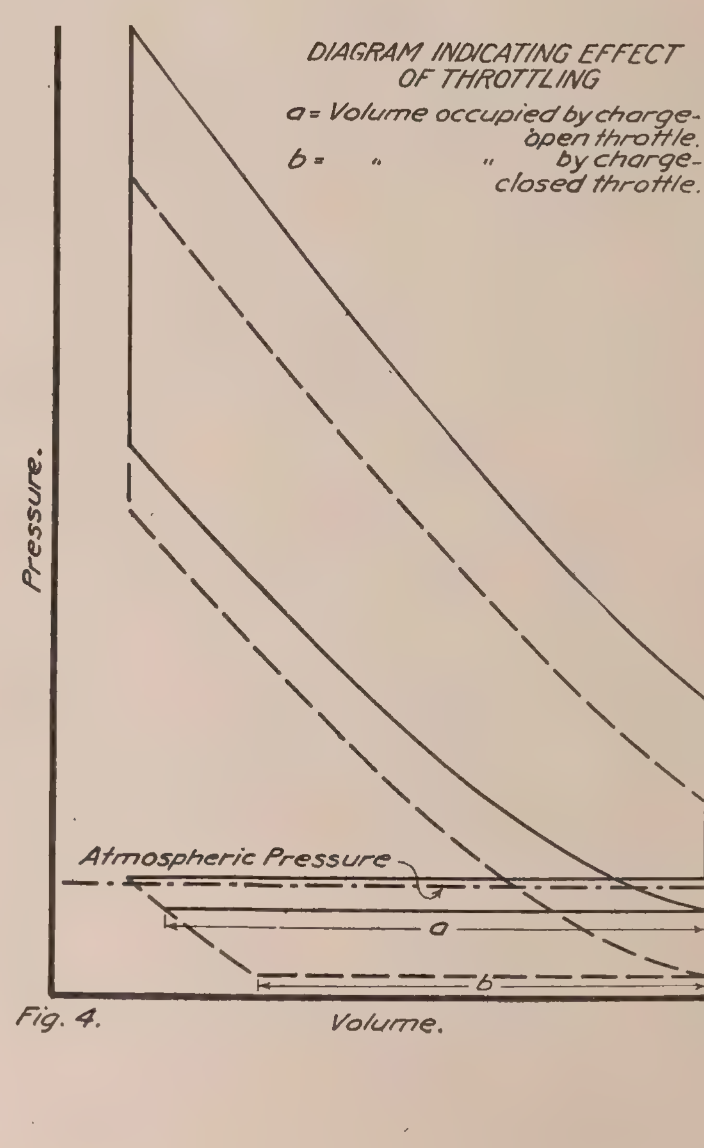


Fig. 4.

pressure with increase of compression ratio is expected because of greater ratio of expansion, and a possible improvement in combustion.

A comparison of figures 1 and 2 shows that the indicated mean effective pressure of the Hispano-Suiza engine is slightly greater than that of the Liberty, at the same air density and compression ratio. This is explained by the less amount of heating with the Hispano-Suiza manifold jackets. When the air supply is at 0° C. (32° F.) the manifold temperature of the Hispano-Suiza is about $-10°$ C. (14° F.), while that of the Liberty is about $+5°$ C. (41° F.). For a given temperature and a given pressure of air supply, the actual density of the air entering the cylinders is reduced by heating, although it is also increased by the cooling resulting from the evaporation of the gasoline. Using the manifold density instead of the supply density, the two engines give the same mean effective pressure for the same density.

Apparently the relation between indicated mean effective pressure and air supply density is based upon the fact that the change of indicated mean effective pressure is proportional to the change of density of the charge at the beginning of compression, irrespective of whether the density is changed by altitude or by throttling.

IV. WORK PER UNIT WEIGHT OF AIR.

One method of rating engines is by the horsepower per cubic inch or liter of piston displacement. This may be somewhat misleading, as it is based on the volume, not the weight, of charge. The piston displacement of an engine is constant, but the weight of charge drawn in on the suction stroke is controlled by the density of the air and by the volumetric efficiency. For example, with a given engine operating with fixed air pressure and temperature, it is possible to alter greatly the horsepower per cubic inch displacement by simply changing the "choke" of the carburetor. The indicated work obtained from unit weight of charge was selected as a means of showing how well the engine or engines utilized the energy supplied, independently of the amount drawn in.

For a first approximation the weight of air was substituted for the weight of charge. The results show that, except for changes due to different compressions and different mixture ratios, the indicated work obtained from unit weight of air is practically constant, and that it makes little difference whether the weight of air per unit time is changed by speed, altitude, air temperature, or volumetric efficiency (throttling the engine changes volumetric efficiency).

The subject of volumetric efficiency is taken up in Part V in connection with the effects of change of air temperature upon engine performance, but it is not out of place to consider it here, in connection with possible increase of engine output by increase of air supplied. The volumetric efficiency is controlled by many different and independent factors, among which are engine speed, temperature, and pressure changes in the air from entrance to cylinder, as well as the design of the whole induction system, including intake valves and ports, valve timing, and manifolds. The magnitude of these factors is generally unknown, except for their combined effect upon volumetric efficiency.

These various reasons for change of volumetric efficiency were considered, and an attempt was made to analyze the magnitude of the effects of some of them. Apparently the change in total pressure drop is relatively unimportant at open throttle. The temperature change from entrance to inside the cylinder is, however, a very important factor. Generally some means is provided for heating the air supply to an engine which of itself reduces the volumetric efficiency, but is counteracted to a greater or less extent by the heat absorption due to evaporation of the gasoline. The net result may be either way, with an added effect of heat transfer to or from the jacket water and piston when the gas is inside the cylinder.

The friction losses in the induction system may be subdivided into those due to (*a*) rubbing friction, (*b*) losses incident to transformations from one form of energy to another. The author has been unable to make the laws of gas friction (flow through pipes) account for changes of volumetric efficiency, and is inclined to believe that a great portion of the friction loss is incidental to transformations from potential to kinetic energy and the reverse. On the whole, the flow through the induction system is essentially a modified "throttling" or "constant heat" process, adiabatic but not isentropic.

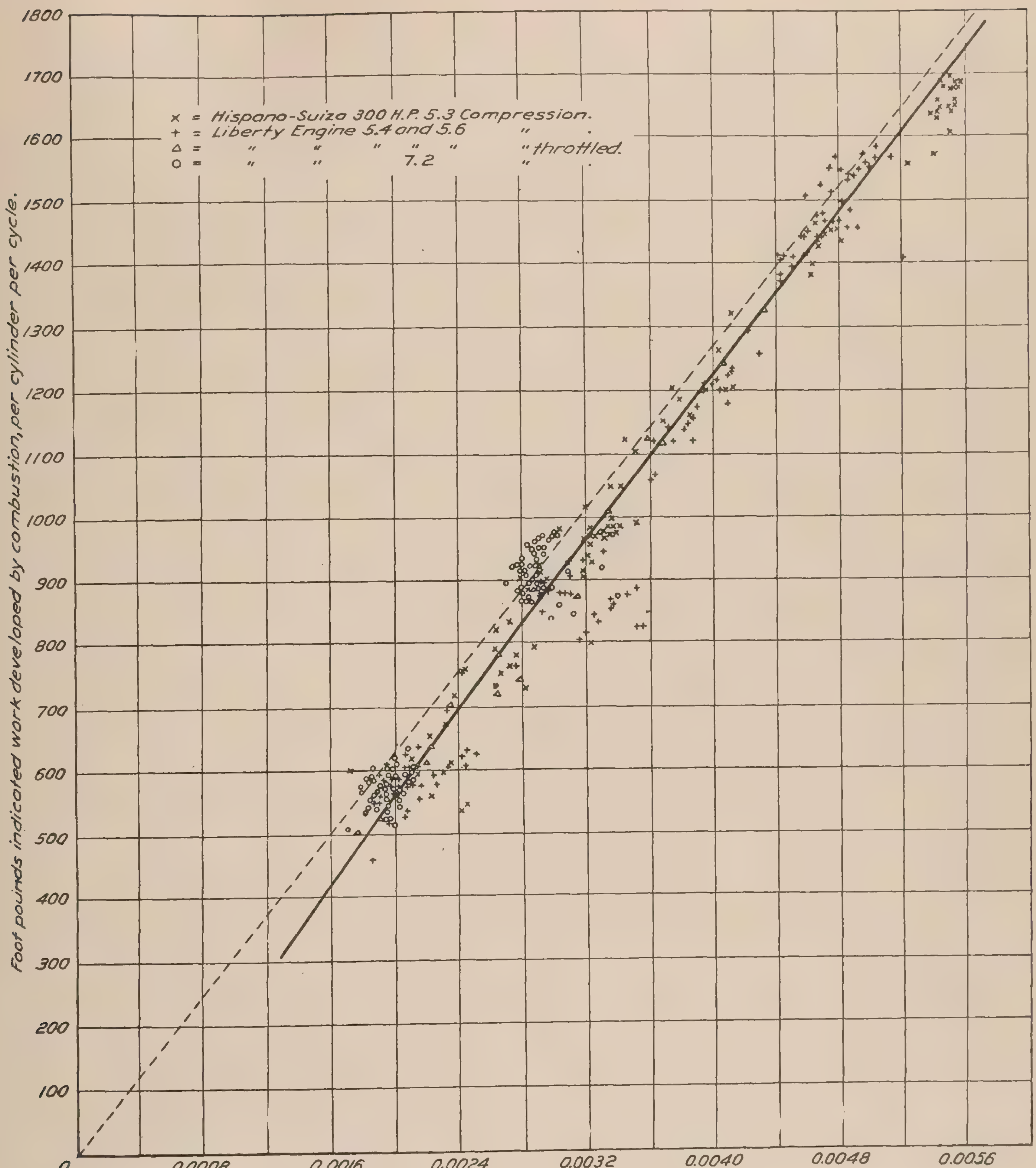
Returning to the subject of the indicated work from unit weight of air, the density of the charge at the beginning of compression is a direct measure of the weight of air drawn into each cylinder at each suction stroke. The use of the weight of air per cylinder per cycle avoids the tedious computations which are incident to obtaining the charge density. In the event that only one engine is considered, the process is further abbreviated by taking weight of air per revolution, which is a constant times the weight per cylinder per cycle.

The points in figure 1 represent the indicated work (ordinates) plotted versus the weight of air (abscissæ), or, more strictly, the foot-pounds of indicated work per cylinder per cycle versus the corresponding weight of air used per cylinder per cycle. Figure 1 was plotted as a generalization, using data from many tests of both the Liberty and the Hispano-Suiza engines. The tests plotted include all of the several compression ratios previously mentioned, with the engines operating at the several speeds, altitudes, mixture ratios, and throttle position. On the whole, the points indicate a definite relation between the indicated work and the weight of air supplied, although there is a considerable scattering. The line shown as the curve is really the center of a zone or band, and was located without reference to the points (circles) representing the 7.2 compression data.

As there were many variables included in the tests shown in figure 1, data from one engine, a Liberty 12 with 5.4 compression, was plotted on figure 2, using coordinates proportional to those of figure 1. That is, the indicated mean effective pressure is used in place of the foot-pounds of work per cylinder per cycle, and pounds of air divided by r. p. m. are employed as equivalent to the weight of air per cylinder per cycle. On figure 2 all the variables are eliminated except mixture ratios and possible changes in the efficiency of combustion at different densities. The mixture ratio would be expected to have considerable influence upon the work obtained from a pound of air (when mixed with gasoline), hence the points where the data would plot are marked by numerals which are the values of the corresponding mixture ratios. On the tests plotted in figure 2 the weight of air supplied per cycle was changed mainly by alterations of the density through change of pressure, although changes of speed from 1,200 to 2,000 r. p. m. caused smaller changes in the air weight per cycle at such altitude, and on some of the runs the engine was throttled. In the latter case the changes due to change in the lower loop of the indicator diagram have been handled as previously explained.

The indicated mean effective pressure is found to be nearly, but not quite, proportional to the weight of air supplied per cycle. A reduction in the density of the air causes a slightly more than proportionate decrease in the mean effective pressure, especially at the smaller densities. The mixture proportions have considerable effect upon the work to be obtained from a pound of air, other conditions being constant, and this subject of effect of mixture ratio is discussed in another part of this report. The speed of the engine has no effect, at least for the speed range covered by the data on figure 2.

The explanation of the more rapid falling off of indicated mean effective pressure per pound of air at the greater altitudes was not obvious. It was suspected that this might be due to less complete combustion at reduced densities. Some few exhaust gas analyses had previously been made on a similar engine which was being tested in the altitude chamber. These analyses were examined to see if they showed any change in efficiency of combustion with change of altitude. The determinations of oxygen in the exhaust gas, as shown by these analyses, are plotted versus barometer on figure 3, and apparently indicate a greater amount of uncombined oxygen at the higher altitudes. In obtaining the samples of exhaust gas reasonable precautions were taken to prevent air leaking in, since the result of a leak would be to increase the oxygen content of the sample as the pressure was reduced in the altitude chamber. The excess oxygen in the exhaust at altitude would also be a result of a decrease in the efficiency of combustion. Evidence from recent tests with a *constant* mixture ratio hardly support the assumption that the efficiency of combustion changes appreciably with altitude. But, even so, it would not follow that, with mixture adjusted for *maximum power*, there may not be a change in the process of combustion.


× = Hispano-Suiza 300 H.P. 5.3 Compression.
+ = Liberty Engine 5.4 and 5.6 " .
△ = " " " " " " throttled.
○ = " " 7.2 " .
Foot pounds indicated work developed by combustion, per cylinder per cycle.
Pounds of Air per cylinder per cycle.
1800
1700
1600
1500
1400
1300
1200
1100
1000
900
800
700
600
500
400
300
200
100
0
0.0008
0.0016
0.0024
0.0032
0.0040
0.0048
0.0056
Fig. 1.

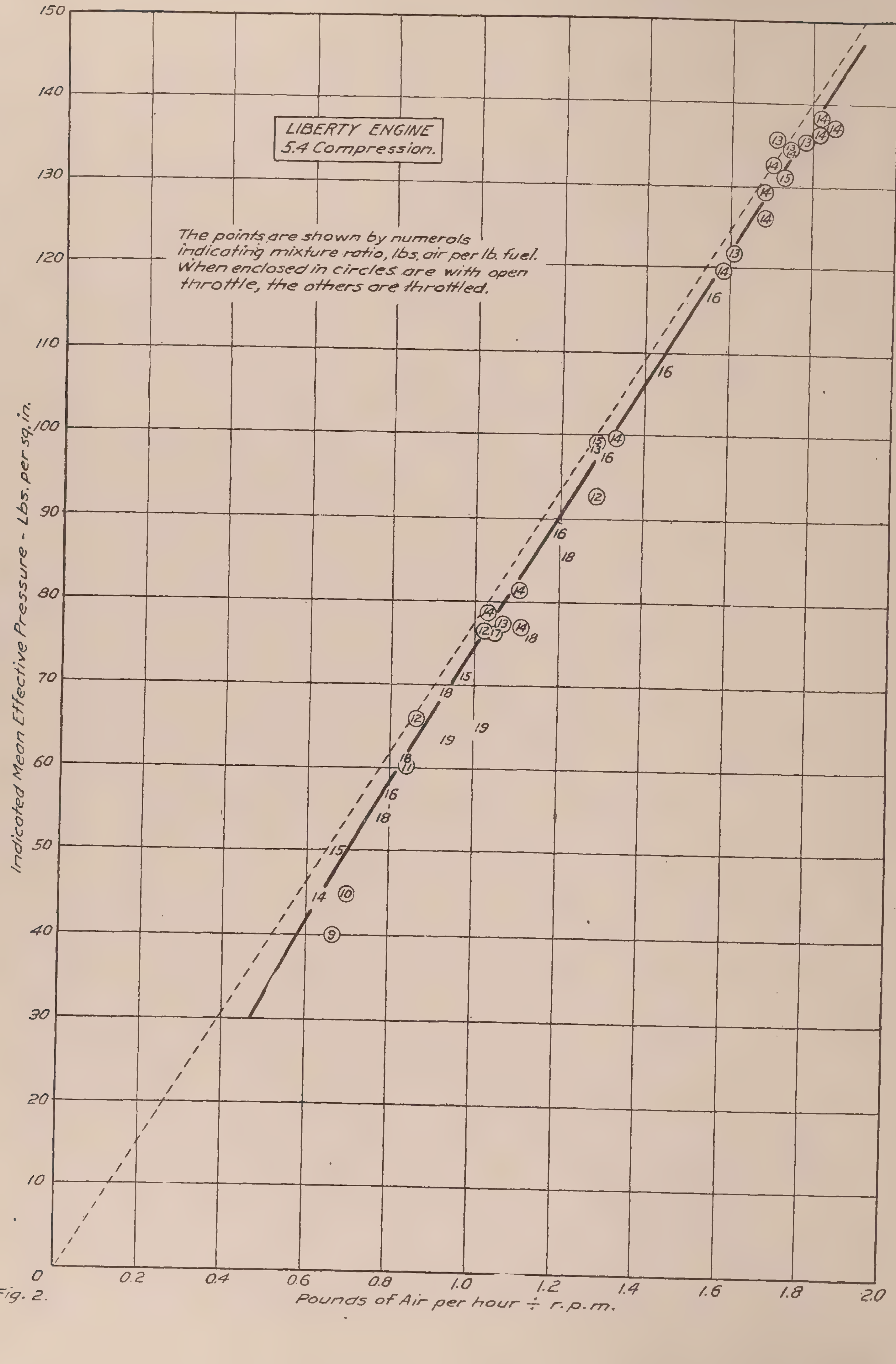

LIBERTY ENGINE
5.4 Compression.
The points are shown by numerals indicating mixture ratio, lbs. air per lb. fuel. When enclosed in circles are with open throttle, the others are throttled.
Indicated Mean Effective Pressure - Lbs. per sq. in.
Pounds of Air per hour ÷ r.p.m.
0
0.2
0.4
0.6
0.8
1.0
1.2
1.4
1.6
1.8
2.0
10
20
30
40
50
60
70
80
90
100
110
120
130
140
150

Fig. 2.

An approximation of the "Coefficient of utilization of the air" was computed by the formula:

$$\frac{20.9-O_2}{20.9}$$

in which 20.9 is the per cent by volume of oxygen in air and O_2 is the per cent by volume of oxygen in the exhaust gas. The efficiency of utilization as computed from the curve of figure 3 is given on figure 4. Apparently about 5 per cent less of the oxygen is used at 30,000 feet altitude than at the ground when the mixtures is adjusted for maximum power. As to the unburnt fuel, there were few determinations of hydrocarbons in the exhaust, and such as were obtained were vitiated by the presence of derivatives from the lubricating oil.

If the weight of air supplied to the engine per cycle at any given barometric pressure be multiplied by the appropriate coefficient of utilization estimated from figure 4, it will cause the

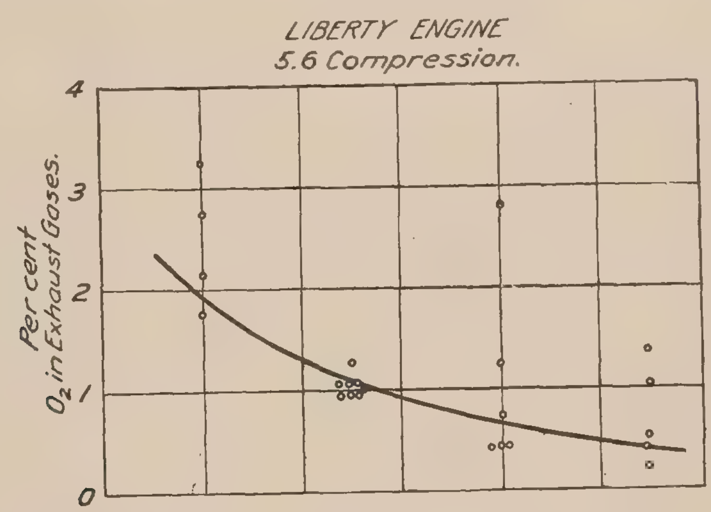


Fig. 3.

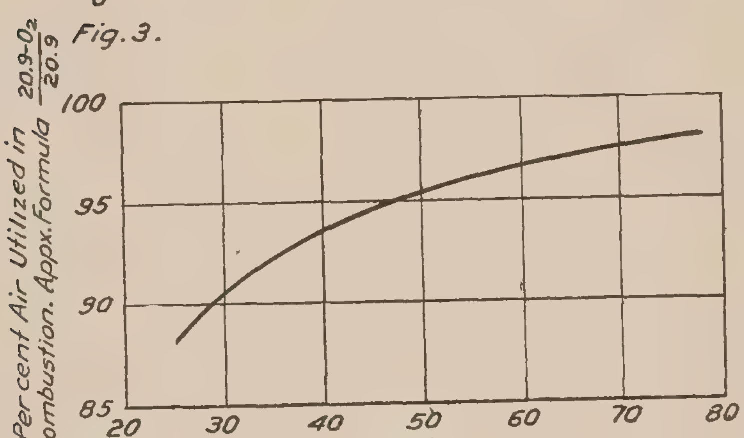


Fig. 4.

curve of figure 2 to become the dotted straight line passing through the origin. This indicates that the indicated mean effective pressure is directly proportional to the weight of air actually utilized in combustion when the mixture is adjusted for maximum power. For other mixtures the line of proportionality probably will not pass through the origin, but will have an intercept on the abscissæ; in other words, some air will necessarily be unused if there is an excess of air, or may be only partially effective if rich mixtures cause incomplete combustion.

The conclusion drawn from the data reviewed in this chapter is that it is possible to use the weight of air supplied per stroke as a measure of the indicated mean effective pressure, or vice versa, over ranges of considerable barometric pressure, provided the mixture ratio is maintained sensibly constant, and, of course, with a constant compression. As for the basic relationship, air utilized versus engine power, it is appreciated that more work must be done in the field of gas analysis before absolute conclusions can be stated, but the relation indicated by the data here considered is the same as that which would be expected from theoretical considerations, namely, that indicated torque is proportional to the weight of air utilized in the combustion, entirely independent of altitude, and approximately independent of speed or of small changes in mixture ratio.

V. EFFECTS OF CHANGE OF TEMPERATURE

It is customary to supply heat to the mixture of air and gasoline in order to aid carburetion and distribution, although this heating reduces the engine power at open throttle. Heating the air supply at constant pressure does not reduce power in proportion to change of air-supply density. It has previously been shown that engine power is proportional to air density when the pressure is changed at constant temperature. These apparently contradictory facts led to the work on which this chapter is based, involving the change of volumetric efficiency with air temperature, and from which it is concluded that the engine power is directly proportional (when conditions are normal) to the weight, or density, of the charge in the cylinder at the beginning of compression.

The outline followed in studying the effects of varying air temperatures was to first find whether the air temperature affected the indicated work obtainable from unit weight of air or of gasoline. Under the conditions and within the limits of these tests no change was discovered. Continuing the study, it was found that the volumetric efficiency increases as the air temperature is increased.

Perhaps it will be well to say that volumetric efficiency as used in this report is defined as the weight of air actually drawn into the engine in a given time, divided by the weight of air required to fill the piston displacement for the same time interval, this latter weight being computed for the density existing at the entrance to the carburetor. When volumetric efficiency is figured in this manner, it gives the net results of the engine performance just the same as does the engine power, and both are affected by any changes in the induction system. For example, a refrigerating coil could be placed in the intake manifold and (assuming the fuel to be noncondensable) with absolutely no other change the engine power could be greatly increased as a result of the greater weight of charge drawn in. The volumetric efficiency could thus be made greater than 100 per cent.

It was necessary to find the reasons for the improvement in the volumetric efficiency with increase of air temperature, so a study was made of the temperature changes in the intake manifold, revealing that there is greater cooling by evaporation of fuel and less heating by jackets as the air temperature is increased. Both of these effects tend toward a higher volumetric efficiency, or, in other words, an increase of relative density inside the cylinder at the higher air temperatures.

One of the suggestions arising from this work is that it may be possible to break up a heavy fuel by intense heating of the air, subsequently cooling the charge in order to minimize the loss of power.

In testing in the altitude chamber, the air temperature is generally maintained constant for each and the same for all tests. However, there are tests in which the air supply temperature was varied, using a Liberty 12-cylinder aviation engine with 7.2 compression ratio, running at 1,700 r. p. m. with open throttle at altitudes of 14,000 and 25,000 feet, and using "X" gasoline for fuel. The intake manifolds had the usual water jackets.

Figure 1, work per pound gasoline versus temperature, is plotted from all the data of these variable temperature tests, divided into groups, each group consisting of a certain range of mixture ratios. Figure 2, work per pound air versus temperature, is plotted from data carefully selected for about maximum power mixtures and for constant relation between the pressures of carburetor supply, engine exhaust, and altitude chamber.

The abscissæ of figures 1 and 2 are manifold temperatures, which could be converted to air supply temperatures by changing the scaling. As the indicated work per pound of gasoline (fig. 1) and per pound of air (fig. 2) does not change with temperature, the numerical values of the temperature scale are immaterial. A comparison of the three ranges of mixture ratio, shown by the three sections of figure 1, indicates that the work obtained per pound of gasoline increases as the mixture becomes leaner. This result is also shown in the chapter on mixture ratios. In the lower section of figure 1, mixture ratios of from 18 to 22, the points indicate a possibility that the work may increase slightly at high temperatures with these very lean mix-

tures, but as this plot includes quite a range of mixtures, it is also possible that there may be some accidental coincidence, such as a grouping of the leaner mixtures at the higher temperatures, which would give the same appearance. Data sufficient to settle this point are not at present available.

Another feature which should not be overlooked when studying this subject is the existence of what may be termed a critical temperature for each gasoline. To explain this, it has been found that "X" gasoline does not carburet well if the air supply is much cooler than the freezing point of water. Reports from sources outside of the Bureau of Standards indicate that a temperature of at least 15 or 20° C. (60 or 70° F.) is necesaary to handle commercial gasoline. The tests considered in this chapter are not below the "critical temperature" of the fuel used.

Figures 1 and 2 show that the indicated work obtainable from unit weight of charge is not varied by change of temperature. As the volume of an engine cylinder is constant, and as the weight of charge is proportional to its density, and as heating the air does not decrease power as fast as it does density, it follows that the air supply density is not proportional to the charge density inside the cylinder.

Density changes of charge, due to changes in temperature, and perhaps pressure, after the air has entered the carburetor, may be expected to explain these changes of volumetric efficiency. Such changes in charge temperature will not be completed in the manifold, but will continue inside the cylinder, and will alter the volumetric efficiency until the intake valve is closed. Manifold temperatures and pressures indicate the direction and magnitude of the density changes, although the information afforded by them is incomplete, in that the final conditions inside the cylinder are unknown, except indirectly by means of volumetric efficiency. This development is illustrated by figures 3 to 10, inclusive, is outlined in the following discussion, and is based upon data from a Liberty engine of 7.2 compression ratio when conditions were as follows: 1,700 r. p. m., 14,000 feet altitude, 1 part by weight of "X" gasoline with from 14.6 to 16.9 parts of air, spark advance 21.5°, jacket water in about 65° C. (150° F.), out about 73° C. (160° F.), except for three of the several runs at higher air temperatures when jacket was varied, both above and below normal.

The manifold temperature, measured at the intake valve (fig. 3) is higher than the air supply temperature when the latter is below about −7° C. (+20° F.), because the jackets are giving more heat to the charge than is abstracted by evaporation of gasoline. With air supply temperatures above this the heat transfer from the jackets becomes less and less as the air supply temperature approaches the jacket temperature, while at the same time more and more of the gasoline evaporates in the manifold, withdrawing increasingly greater quantities of heat. Figure 4 shows the ratio of absolute temperature of air supply to that of the manifold $\frac{(T_c)}{(T_m)}$ plotted against air supply temperature. If there were no change in pressure drop from entrance to manifold, this ratio would be a measure of the density change. Figure 5 is the plot of absolute pressures of air supply and of manifold, and figure 6 is their ratio $\frac{(P_m)}{(P_c)}$, all versus entrance air temperature. The pressures and their ratios change but little with air temperature, so that the ratio of density in manifold to that at carburetor $\frac{(d_m)}{(d_c)}$ (fig. 7) is practically the same as the absolute temperature ratio $\frac{(T_c)}{(T_m)}$ (fig. 4), and both have the same form, but with a little more than half the magnitude of the volumetric efficiency change (fig. 9, *a*). The actual engine performance, brake mean effective pressure versus air temperature, is shown on figure 8, lower *a* curve, and the indicated m. e. p. is the upper *a* curve. Curves *b* are computed by taking the m. e. p. at 0° F. as a starting point and assuming the m. e. p. to change in direct proportion to the density of air supply. It is seen that the m. e. p. (power) does not decrease as fast as the air supply density when air temperature is changed at constant pressure. Curves *c* are computed by taking the m. e. p. at 0° F. as a starting point and assuming the m. e. p. to vary in direct proportion to the density in the manifold. This assumption (*c* curves) approaches the

actual performance (*a* curves) more closely than does the assumption of power varying with air supply density (*b* curves). If the density inside the cylinder were known, it is quite probable that the indicated m. e. p. would be found to vary directly with the density at beginning of compression. Under the conditions of these tests the volumetric efficiency should be a measure of the actual total density changes. Figure 9, *a* is the actual observed volumetric efficiency; *b* and *c* are computed on the same basis as the *b* and *c* curves of figure 8.

It was of interest to see how the indicated mean effective pressure would vary with air supply temperature if volumetric efficiency could be kept constant at 100 per cent. In order to imitate this condition the actual indicated m. e. p. values at the several air temperatures were divided by the corresponding observed volumetric efficiencies, the results being shown by crosses on figure 10. To make a comparison with the assumption that power varies with density, the value of $\frac{\text{i. m. e. p.}}{\text{vol. eff.}}$ at 0° F. was selected as a starting point, and the $\frac{\text{i. m. e. p.}}{\text{vol. eff.}}$ values at other air temperatures were computed upon the assumption that m. e. p. varies directly with the density, density being changed only by change of temperature. The results are shown by circles on figure 10 and they coincide with the points (crosses) from the observed data. This confirms the statement that a change of temperature of entering air alters the power of an engine only as it changes the density of the charge at the beginning of compression, provided that the fuel is properly vaporized or pulverized at the time of ignition. But the reduction of power is not directly proportional to the increase of absolute temperature of the entering air because at the same time there is a reduction of heat transfer from the surrounding media and also a greater cooling effect because more of the fuel is vaporized before the intake valve closes. The extent of these two modifying factors is determined by the design of the engine, the conditions of operation, and the nature of the fuel employed.

There can be no formula of universal application which will convert the engine power or torque actually obtained at one air supply temperature to that which would be obtained by test at another temperature. However, for one engine, operated under constant conditions except air temperature, and using one fuel, an empirical formula may be devised to convert engine power at one air temperature to the power that would be obtained at another air temperature.

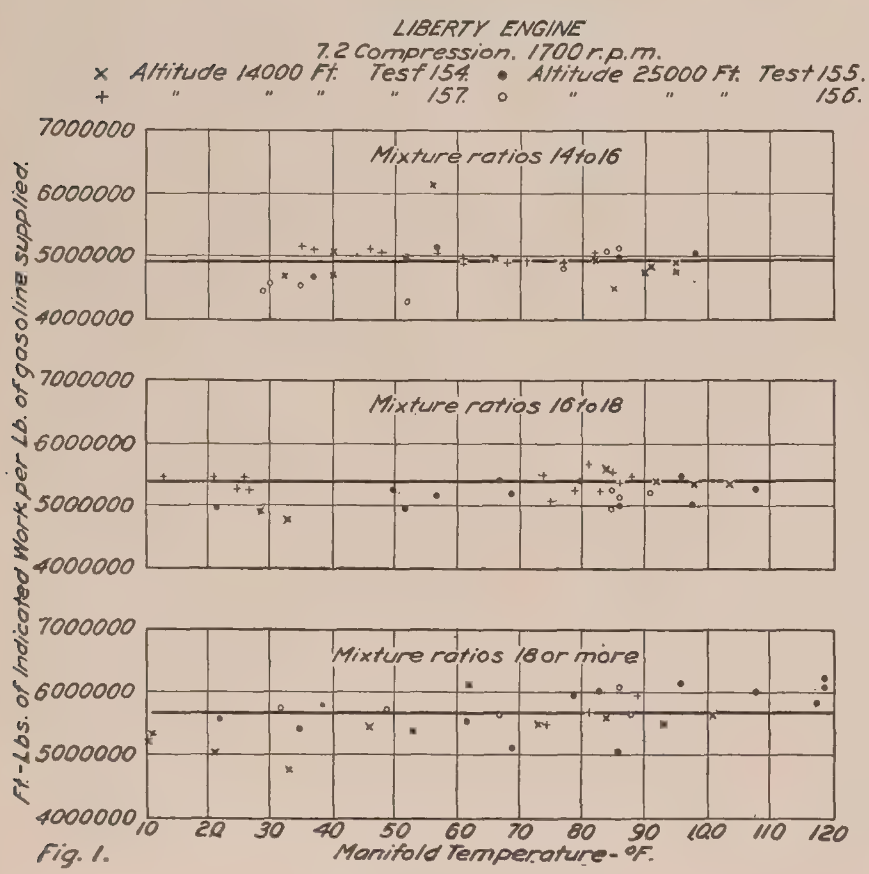


Fig. 1.

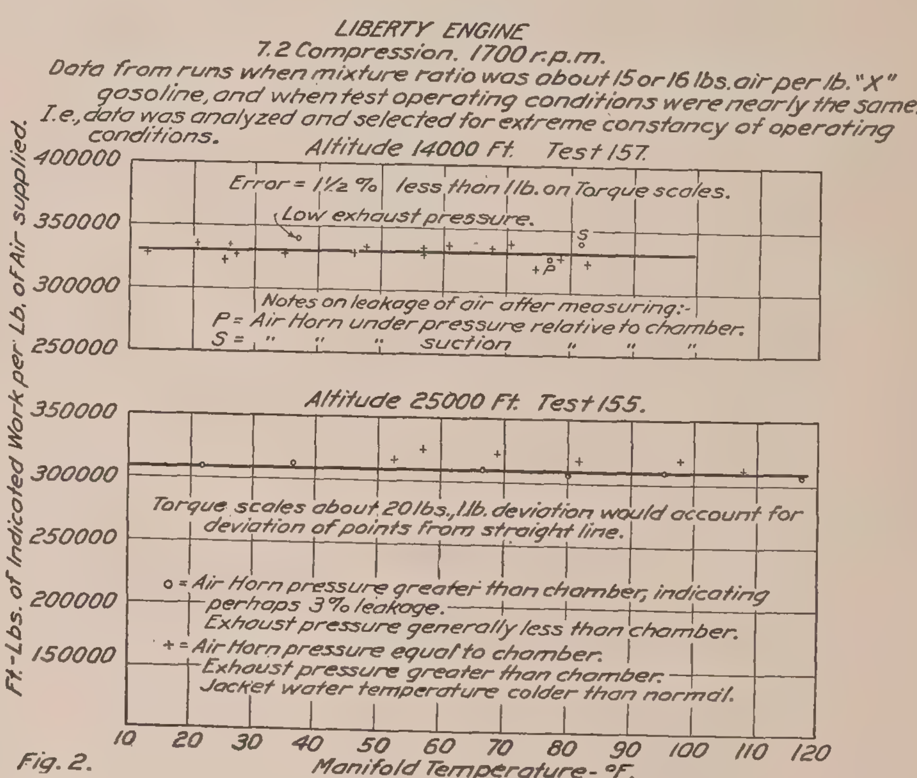


Fig. 2.

LIBERTY ENGINE
7.2 Compression 1700 r.p.m. "X" Gasoline.
Altitude 14000 Ft. Test 157.
Mixture Ratios from 14.6 to 16.9 only. Spark advance 21.5°. Jacket Water in 145 to 150 °F., out 160 to 165 °F., except with Carbureter Air temperatures of 95, 107 & 108° when Jacket out was 129, 144, 176,° respectively.

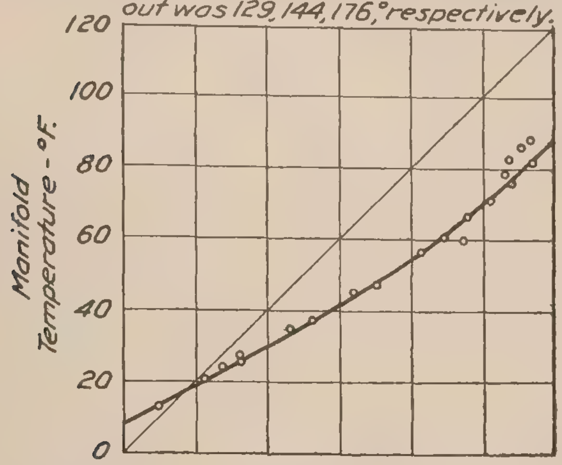


Fig. 3

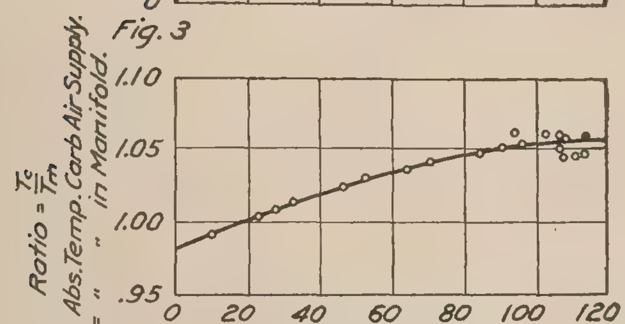


Fig. 4.

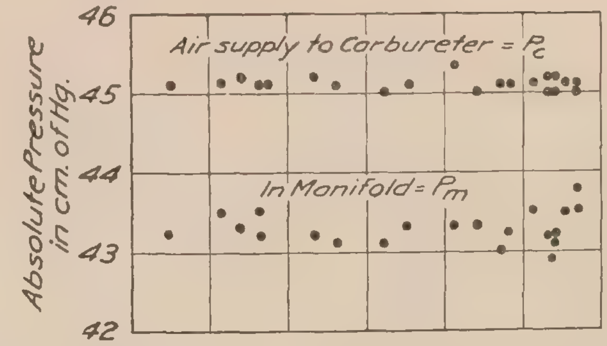


Fig. 5

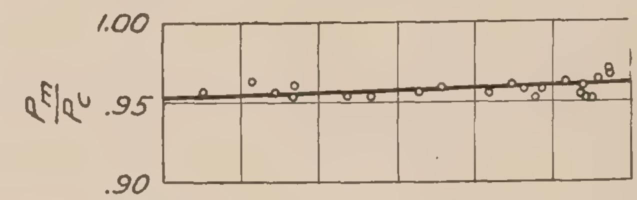


Fig. 6.

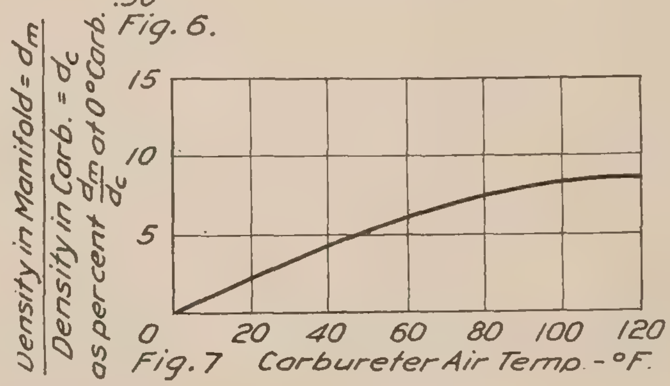


Fig. 7

a = Test results.
b = Assumption that value is proportional to density of Carbureter supply air.
c = Assumption that value is proportional to density in manifold.

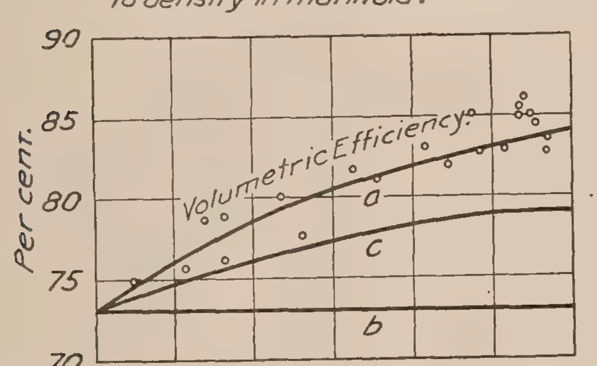


Fig. 9.

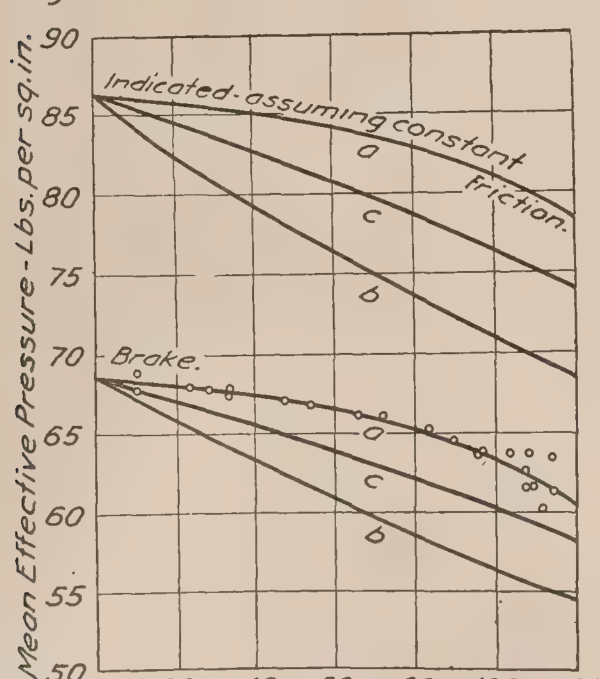


Fig. 8

+ = I.M.E.P. test values, divided by corresponding Volumetric Efficiency.
∘ = (I.M.E.P.$_o$ + V.E.$_o$) × (T_c ÷ T_o)

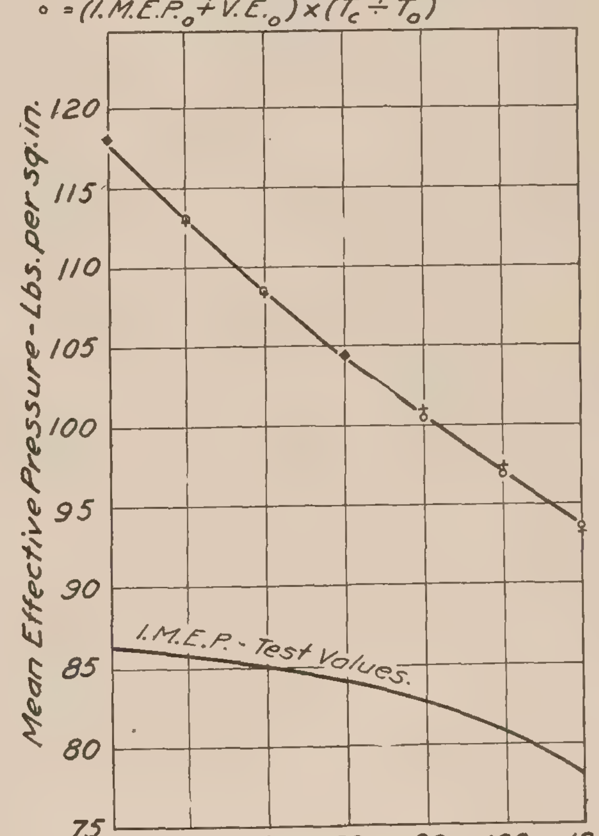


Fig. 10.

VI. EFFECTS OF CHANGE OF MIXTURE RATIO.

The subject of mixture ratios is receiving considerable attention at the present time. The mixture of air and gasoline required for maximum power is known to be slightly richer than that for maximum economy. Either too rich or too lean a mixture results in a loss of both power and economy.

The effect of change of altitude and compression ratio upon the proportion of air and fuel for, say, maximum power is not definitely known. It is possible that the proportion giving maximum economy at the ground may not be the same as that giving maximum economy at another altitude, or with another compression ratio, or at another throttle position. The same statement applies to the mixture for maximum power. If altitude and compression ratio change the required proportion for a desired result, such change may be explained by the effects of changed density and heat content of the charge at the time of ignition upon the process of combustion. Although no evidence of the influence of engine speed could be detected on these tests, at speeds from 1,200 to 2,200 r. p. m., still it is possible that the relation of the velocity of flame to piston travel may have some influence upon power.

This subject of mixture ratio involves the question of how much energy is liberated by the combustion of the fuel under the several conditions. Exhaust gas analysis could well be used in the search for more light upon the subjects. Another point to be considered is the possible difference between the actual mixture ratio, defined as the measured air to measured gasoline, and the effective mixture ratio, defined as the proportions of air and gasoline actually combining inside the cylinder.

At the present time data are being obtained in the altitude chamber upon the effects of change of mixture ratio, but among the older tests there are only a few with much range of mixture, as most of the runs were made with the carburetor adjusted for the best economy consistent with the condition of maximum power. However, the tests upon the 7.2 compression engine also included runs with the mixture purposely made considerably leaner than the maximum power adjustment, and some runs on the engines with about 5.4 compression also included rich mixtures as well as maximum power mixtures.

A general survey was made of the data from practically all of the older tests, using basic relationships. By basic is meant the fundamental relations between mixture ratio and the indicated work resulting from unit weight of (1) gasoline and (2) air supplied. The use of indicated instead of brake work placed the data from all the engines on a more comparable basis and removed some inconsistencies resulting from variations of volumetric efficiency. The results of the general survey indicated that the work obtainable from unit weight of gasoline steadily increased as the mixture was made leaner until the ratio was about 18 or 20 to 1, when the engine performance became erratic, probably due to misfiring. This erratic behavior did not always begin at the same mixture ratio for the various altitudes, compressions, etc. The survey also indicated that the work obtainable from unit weight of air followed the form into which the usual curves of power versus ratio would convert; that is, the maximum power is obtained from unit weight of air with a mixture perhaps slightly richer than the theoretical combining proportion of 15 weights of air to 1 of gasoline.

Guided by these general relations, data were selected so as to segregate the various variables as much as possible, still retaining comparable tests with varying mixtures. These are shown in figures 1 and 2, plotted to the coordinates previously mentioned, namely, mixture ratio as abscissæ for both plots, and ordinates of indicated work per pound of gasoline for figure 1 and per pound of air for figure 2. One plot or section of figure 1 is made with data from a Liberty engine with 7.2 compression, another from a Liberty with 5.4 compression, and the third from a Hispano-Suiza 300-horsepower engine with 5.3 compression. All of the plots of figure 2 are from a Liberty 7.2 compression engine, at different times and different altitudes. In order to reach absolute conclusions it will be necessary to cover the whole range of mixture ratios at each compression, as this is a major factor. It is evident that altitude is also a factor, probably of minor importance, and that speed changes of the magnitude of from 1,200

to 2,200 r. p. m. are absolutely negligible in altering the work obtainable from unit weight of either air or fuel at a given mixture ratio.

The numerical values of figures 1 and 2 should not be used for commercial gasoline, as the fuel used on all these tests was one of high volatility, complying with the specifications for aviation gasoline used during the war for export to the American Expeditionary Forces.

The oil which enters into combustion in the aviation engines could not be considered, although heat balances and gas analyses both indicated that in some cases the oil is a considerable source of fuel.

The tentative conclusions drawn from the study of mixture ratios are that the indicated work obtainable from unit weight of fuel (the indicated thermal efficiency) increases as the mixture is made leaner up to the point when weak mixtures cause erratic engine performance; and that the indicated work obtainable from unit weight of air is at a maximum (meaning that maximum power is obtained) at about the theoretical combining proportions of 15 parts by weight of air to 1 of gasoline, decreasing with change of mixture in either direction, but being fairly constant from perhaps 14 to 16.

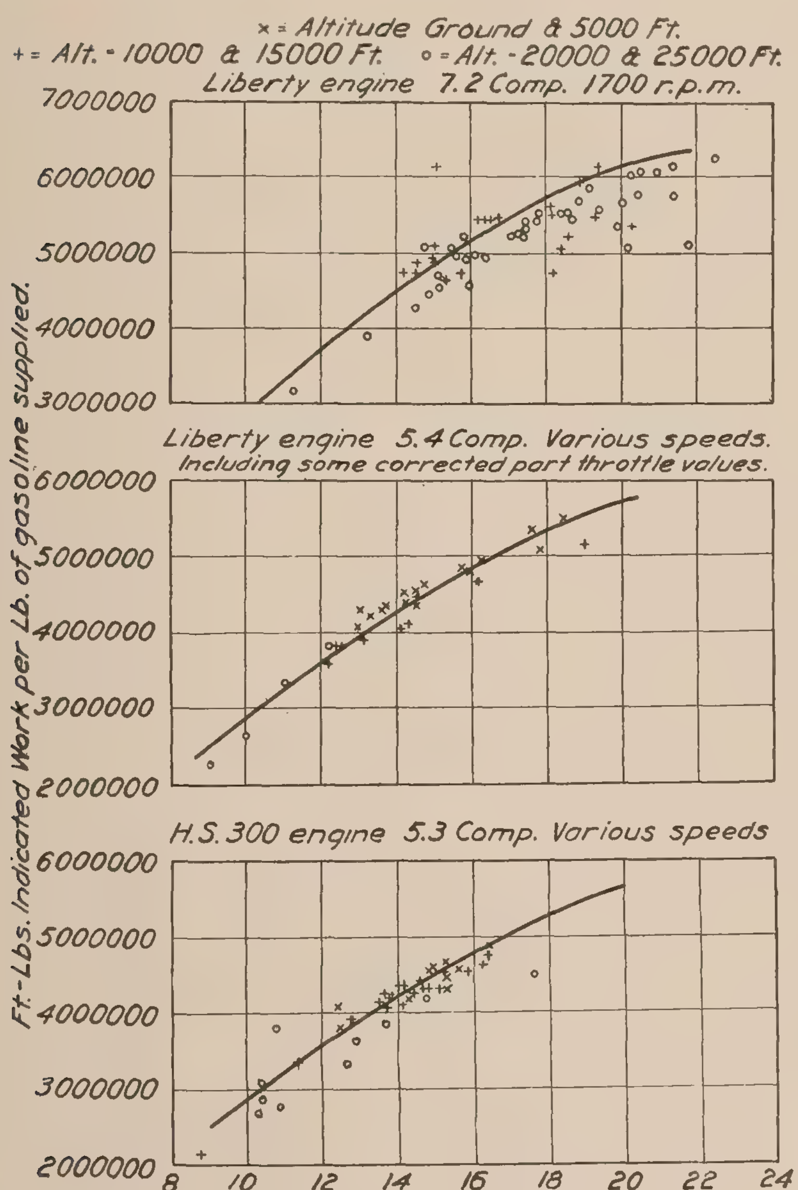


Fig. 1.

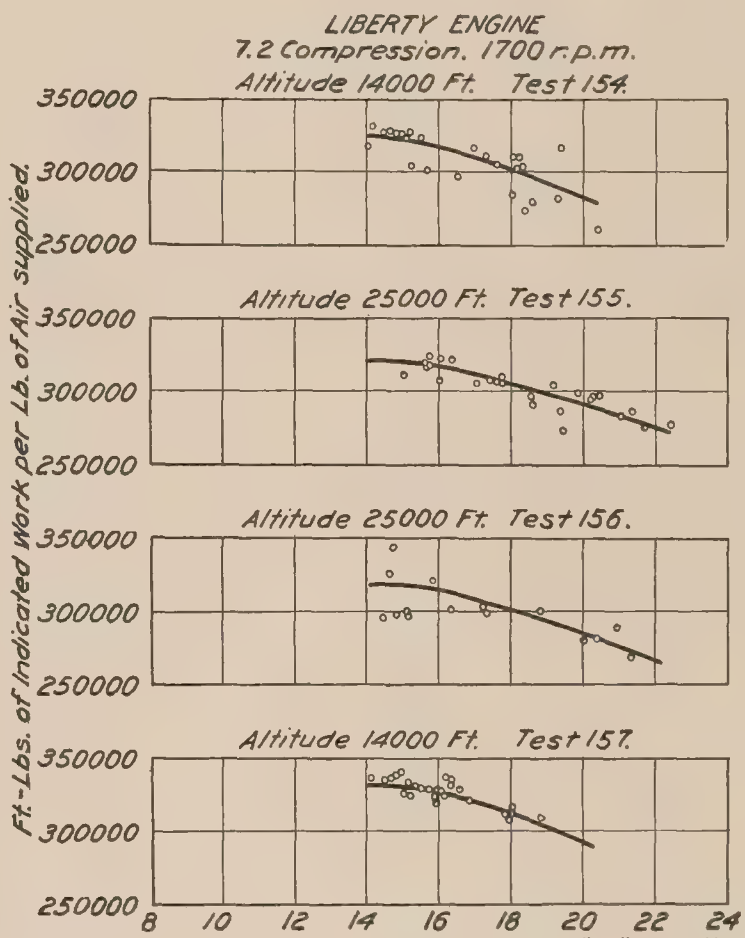


Fig. 2.

VII. HEAT BALANCES.

The heat balance of an engine shows the disposition made of the energy supplied, and is a valuable aid in making improvements in engine performance. The data obtained in the altitude laboratory permit accounting for most of the energy supplied, as it is exceptionally complete in all respects. In this section a comparison has been made between the heat balance obtained from actual engine tests and that computed by thermodynamics from the corresponding theoretical cycle, indicating that the greatest possible improvement in thermal efficiency can be made by increasing the compression ratio.

A major portion of the potential energy supplied is actually made available to the engine by the chemical process of combustion, but not all, as some of the elements or compounds usually escape uncombined. And unburned fuel in the exhaust may be of a different chemical nature from that supplied, in which case chemical changes represent energy changes. For example, if the exhaust contains carbon monoxide, alcohols, or aldehydes, they represent chemical changes and also energy losses.

The thermal efficiency of an engine is usually computed as the actual work divided by the chemical energy supplied. This fails to distinguish between the losses due to incomplete combustion and the losses due to incomplete transformation from heat energy to work. For purposes of analysis of engine performance it would be well to separate the efficiency of combustion from the efficiency of transformation of the energy actually made available to the mechanism, the former probably varying more with change of operating conditions than does the latter.

One method of approximating the efficiency of combustion is by assuming that all the heat made available by combustion appears in the total heat accounted for during an engine test. In these tests the energy accounted for is in (1) the brake power, (2) the exhaust heat, and (3) the jacket water. Going more into detail concerning these items: (1) The delivered work is the useful result obtained from the engine and needs no comment. (2) The exhaust heat was measured by water cooling the exhaust gases to their original temperature and pressure, and measuring the heat thus absorbed. Therefore, the exhaust heat does not include any loss due to inefficiency of combustion, as it is made up entirely of the sensible heat of all the products of combustion, together with the latent and superheat of the water vapor. (3) The jacket losses were obtained from the quantity and the temperature rise of the cooling water. The heat thus represented includes that taken directly from the combustion, from compression, and from piston friction. As a great part of the friction loss is due to piston friction, a large portion of the friction will appear as heat in the jacket water. It should be noted that the sum of brake thermal efficiency, jacket, exhaust, and radiation is 100 per cent of the heat made available, and that if indicated is substituted for brake thermal efficiency in obtaining the heat available, then some of the friction will appear twice in the total. "Radiation, etc.," includes heat losses from the engine system by conduction, convection, and radiation. The "radiation, etc.," excluding losses due to unburned fuel, is probably of the order of 5 per cent of the heat made available by combustion. If unburned fuel is included, the losses are called "residual heat."

The energy accounted for in exhaust, jacket, and brake (an approximation of the heat made available by combustion) is used as a basis for computing heat balances in this chapter, as well as the usual basis of "heat" supplied in the fuel. Both types of heat balance were compared and used as a means for studying the effects of altitude and speed upon the energy distribution in airplane engines. Table 1 presents data from the Liberty 12-cylinder engine and Table 2 from the Hispano-Suiza model H, 300 h. p., 8-cylinder engine. The tests covered the usual range of engine speeds at each of several altitudes. Both engines had about the same compression ratio, a little more than 5 to 1. The tests were made under the usual standard conditions of the altitude laboratory described in previous chapters. The heat balances using heat supplied in fuel as 100 per cent are based upon the higher heating value of the fuel. Whenever possible and desirable, the indicated thermal efficiency is used in place of the brake thermal efficiency, as it gives informaton concerning the internal thermodynamic performance of the engine uninfluenced by the unstable relation of brake power to friction.

Tables 1 and 2 present the items of the heat balances computed upon the two bases previously mentioned, i. e., per cents of heat supplied in gasoline and of heat accounted for in exhaust, jacket, and brake. The last column of Table 1 is an estimate of the quantity of heat removed in the lubricating oil on the Liberty engine, obtained by noting the temperature rise of the oil from inlet to outlet during the test, and subsequently determining the rate of flow of the oil at various engine speeds. Less than one-half of 1 per cent of the heat supplied is removed by cooling the oil.

Exhaust heat and indicated horsepower, as per cent of heat accounted for, taken from Tables 1 and 2, are plotted against air density (at constant temperature) on the upper section of figure 1. Data from the Liberty engine are denoted by circles, and that from the Hispano-Suiza engine by crosses. In the preliminary plotting of data with these same coordinates it had been found that there was no systematic variation with speed, so the different speeds are not distinguished from one another. The lower section of figure 1 is similar to the upper and is plotted from the same data, the difference being that the exhaust and indicated horsepower at the several densities are plotted against the engine speed. Evidently the two engines are thermodynamically alike, and, within the limits of speed, density, and experimental error of these tests, there is no change in the thermal utilization of the heat made available to the engines.

HEAT REJECTED AND COMPRESSION RATIOS.

Tables 1 and 2 and figure 1 show that the heat rejected in the exhaust of the actual engines nearly coincides with the theoretical heat rejection of a perfect engine operating on the same cycle with the same compression ratio. On these engines there is little chance for the water jackets to cool the exhaust after it leaves the cylinder. The heat rejected from the actual engines, expressed as per cent of the heat accounted for, is practically constant for the several densities and speeds. This agreement between theory and test results led to the tabulation of data from tests of engines with several different compression ratios, including some of the first tests made in the altitude chamber. Heat balances as computed from these tests with several different compressions are given in Table 3, wherein the items are as follows:

EJB is the sum of the heat accounted for in the exhaust, jacket, and brake, expressed as per cent of the heat supplied in gasoline.

E is the heat rejected in exhaust, expressed as per cent of heat accounted for in exhaust, jacket, and brake.

J is the heat removed in jacket water, expressed as per cent of heat accounted for.

B is the brake or dynamometer horsepower, expressed as per cent of heat accounted for.

As friction determinations were not complete on some of the older tests because of lack of power, so the indicated thermal efficiencies are not given for this group of tests. In tabulating the data for Table 3, a speed common to all tests was sought, but some of the tests were made at 1,500 r. p. m., while others were made at 1,600. On figure 2 the brake thermal efficiencies at a given range of altitude are shown to increase with increase of compression, but not quite as much as would be expected from the theoretical cycle. A comparison of the three sections of this figure shows that the highest thermal efficiency occurs at the ground. This is because the mechanical efficiency decreases with increase of altitude, so the brake thermal efficiency must decrease likewise, although previous work indicates that the indicated thermal efficiency would be constant if based on heat available.

By expressing the heat rejected in the exhaust as per cent of the heat accounted for, as in Table 3, the data taken at different altitudes are reduced to a common basis for direct comparison. These data are shown as points superposed upon the theoretical heat rejection versus compression ratio curve on figure 3. Evidently the heat lost in exhaust of actual engines is very closely related to the theoretical heat rejection of the Otto cycle. The points plotted on figure 3, being per cent of exhaust, jacket, and brake, will have values slightly greater than if expressed as per cent of true heat available, because the radiation has been omitted from the former basis.

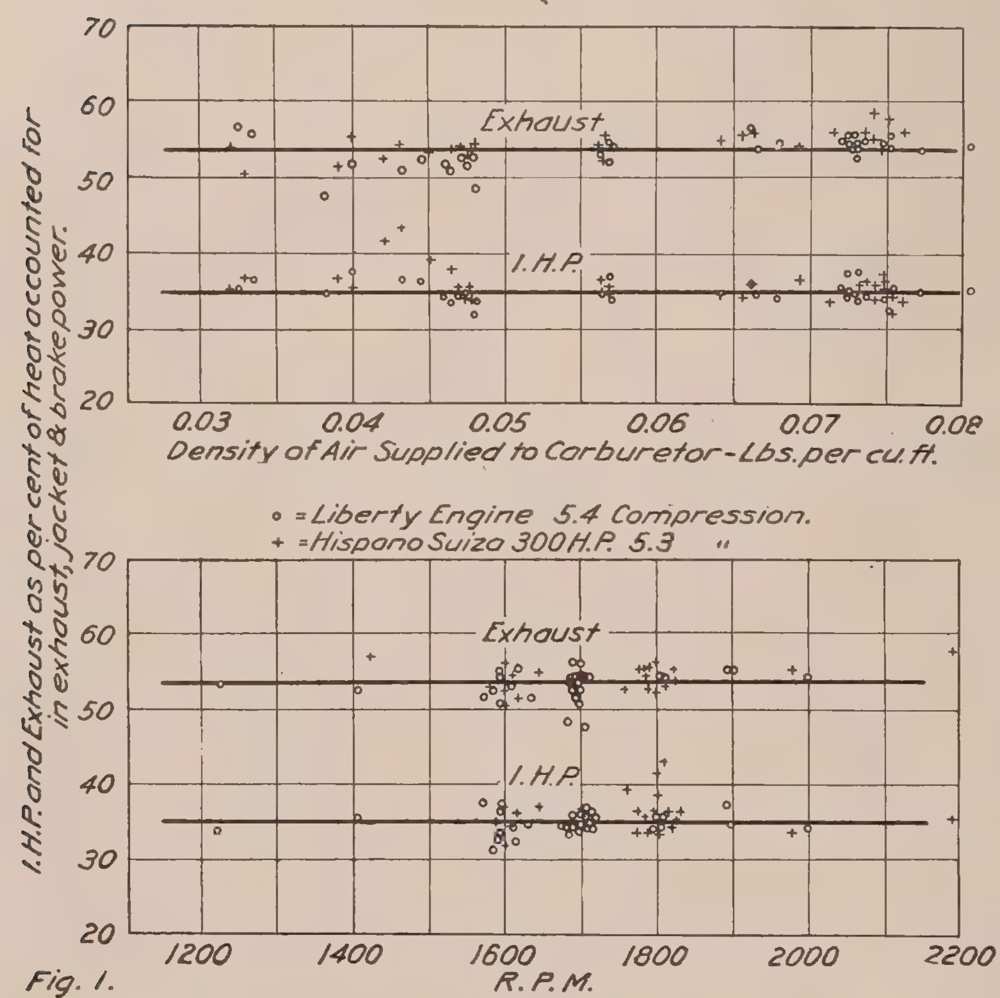


Fig. 1.

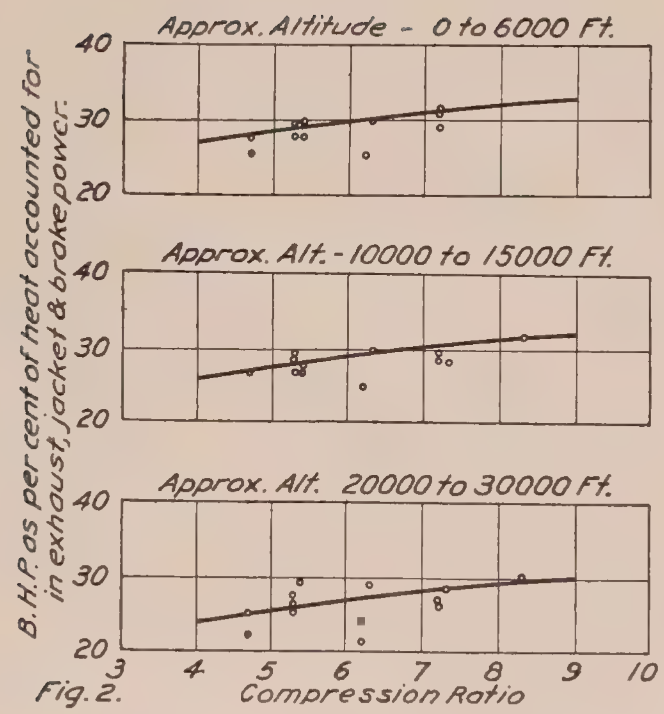


Fig. 2.

"Air Standard" Efficiency $= 1 - \left(\frac{1}{\text{Comp. Ratio}}\right)^{0.408} =$ the efficiency of the Otto Cycle, using air, assuming constant specific heat. $\left(\frac{1}{\text{Comp. Ratio}}\right)^{0.408} =$ heat rejected during the exhaust of this cycle.

The heat rejected in exhaust of actual engines expressed as per cent of heat accounted for (from Table III) is shown by the points superposed upon the theoretical curve.

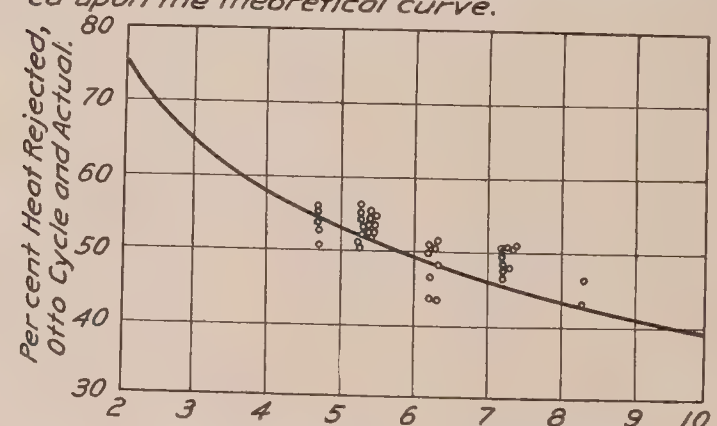


Fig. 3. Compression Ratio $= \frac{\text{Total Volume}}{\text{Clearance Volume}}$

CONCLUSIONS.

Of the heat accounted for about 54 per cent is exhausted and 35 per cent is transformed into useful work with an engine of about 5.3 compression ratio. This distribution apparently is not influenced by altitude nor engine speed.

Of the heat made available by combustion, it is estimated that about 52 per cent is exhausted, the same as the theoretical rejection for this compression. The indicated work accounts for about 33 per cent and the jackets remove about 15 per cent not including friction, or about 20 per cent including friction. It is also estimated that the losses due to imperfect combustion are about 10 per cent or more of the energy supplied when the mixture is adjusted for maximum power, increasing at altitude, perhaps because of richer mixtures, perhaps because of less complete combustion.

Increasing the compression ratio is the only method of reducing the exhaust losses, and offers opportunity for a greater gain in efficiency than any other means, although it introduces many problems of its own, including an increased jacket loss as well as the fuel problem.

TABLE 1.—*Heat balance.*

LIBERTY AVIATION ENGINE.

[Test No. 160—Liberty 12-cylinder engine, compression 5.4.]

Run No.	Altitude (approximate).		Air supply density.		Engine speed.	Heat accounted for in exhaust jacket and brake.		Indicated horsepower as per cent of		Exhaust as per cent of		Jacket as per cent of		Residual heat as per cent of heat supplied.	Heat in oil as per cent of heat supplied.
								Heat accounted for.	Heat supplied.	Heat accounted for.	Heat supplied.	Heat accounted for.	Heat supplied.		
	Km.	*Feet.*	*Gms./liter.*	*Lbs./ft.³*	*R.p.m.*	*Kg. cal./sec.*	*Btu./hr.*								
A- 5	Ground		1.17	0.0731	1,225	180	2,571,000	33.9	26.6	53.2	41.6	17.0	13.3	21.5	0.2
6	do.		1.17	.0732	1,406	199	2,830,000	35.8	27.4	52.6	40.2	16.4	12.6	23.4	.2
7	do.		1.17	.0731	1,598	211	3,015,000	37.6	27.1	53.2	38.3	14.7	10.6	27.7	.3
8	do.		1.15	.0719	1,802	244	3,480,000	35.5	27.6	54.8	42.7	15.7	12.2	21.6	.4
9	do.		1.16	.0726	1,794	251	3,590,000	34.6	29.0	54.8	46.0	16.3	13.7	15.7	.4
10	do.		1.16	.0724	1,893	236	3,375,000	37.4	27.4	55.4	40.6	13.8	10.1	26.2	.4
B- 1	do.		1.17	.0729	1,802	253	3,610,000	35.0	28.4	54.5	44.3	16.3	13.3	18.2	.4
2	do.		1.16	.0727	1,898	260	3,710,000	35.0	28.3	55.2	44.8	16.0	13.0	18.5	.4
3	do.		1.16	.0725	1,999	262	3,740,000	34.4	25.5	54.6	40.5	17.7	13.1	25.3	.5
C- 5	do.		1.20	.0751	1,617	253	3,610,000	32.6	27.7	55.1	47.0	17.3	14.7	14.5	.2
6	do.		1.20	.0748	1,699	255	3,650,000	34.1	28.8	54.5	46.2	16.6	14.1	15.0	.3
E-18	do.		1.29	.0808	1,702	241	3,450,000	35.3	26.9	54.5	41.7	15.7	12.0	23.1	.3
19	do.		1.23	.0771	1,697	243	3,470,000	35.0	27.3	53.6	41.9	16.9	13.3	21.3	.3
20	do.		1.21	.0753	1,700	241	3,450,000	35.6	28.9	53.9	44.0	16.1	13.2	17.9	.3
21	do.		1.18	.0735	1,682	246	3,520,000	34.3	29.1	54.4	46.3	16.7	14.2	14.5	.3
C- 7	1.5	5,000	1.09	.0678	1,692	221	3,150,000	34.0	28.1	54.2	44.9	17.4	14.4	16.9	.3
8			1.06	.0664	1,610	207	2,960,000	34.8	25.5	53.2	39.0	17.7	13.0	26.2	.3
E-12			1.06	.0661	1,690	201	2,875,000	36.0	30.5	56.4	47.8	13.6	11.5	14.9	.4
C- 9	3.0	10,000	.91	.0569	1,596	176	2,510,000	33.7	26.3	54.1	42.2	18.3	14.2	21.6	.4
10			.90	.0565	1,688	168	2,400,000	35.9	23.5	52.4	34.4	19.8	13.0	34.0	.3
D- 1			.91	.0568	1,707	167	2,380,000	37.0	28.3	54.5	41.8	15.4	11.8	22.9	.4
E- 6			.90	.0564	1,694	179	2,555,000	34.4	29.3	53.2	40.8	18.8	14.4	22.9	.4
D- 2	4.6	15,000	.77	.0481	1,686	143	2,040,000	33.7	24.2	48.6	34.9	25.5	18.3	27.5	.5
3			.77	.0481	1,589	150	2,145,000	31.9	25.7	52.7	45.1		18.6	13.9	.5
8			.74	.0466	1,595	136	1,940,000	33.3	23.9	50.8	34.1	21.6	14.5	32.3	.4
E- 1			.76	.0474	1,632	124	1,767,000	34.8	28.1	51.3	41.6	22.1	17.9	18.2	.5
22			.75	.0471	1,690	141	2,015,000	34.2	26.3	52.4	39.8	21.1	16.0	23.4	.5
23			.74	.0464	1,696	141	2,020,000	34.6	23.8	51.5	35.8	21.7	15.0	30.0	.4
24			.71	.0445	1,697	135	1,930,000	36.0	24.6	52.2	35.8	19.8	13.6	30.9	.5
25			.69	.0433	1,702	134	1,915,000	36.3	25.8	50.4	35.9	21.6	15.4	30.1	.5
D- 4	6.1	20,000	.64	.0402	1,572	101	1,450,000	37.8	24.1	51.8	32.9	19.0	12.1	35.8	.5
5			.61	.0382	1,704	110	1,575,000	34.8	20.9	47.6	28.9	27.0	16.3	38.9	.5
D- 6	7.6	25,000	.52	.0324	1,702	71	1,020,000	35.4	14.3	56.4	22.8	21.7	8.5	59.3	.5
7			.54	.0335	1,595	73	1,040,000	36.4	16.5	55.2	25.3	19.8	9.0	54.0	.5

TABLE 2.—*Heat balance.*

HISPANO-SUIZA AIRPLANE ENGINE.

[Test No. 162—Model H engine, compression ratio 5.3.]

Run No.	Altitude (approximate).		Air supply density.		Engine speed.	Heat to jacket.		Heat accounted for in exhaust, jacket, and brake.		Indicated horsepower as per cent of		Exhaust as per cent of		Residual heat as per cent of heat supplied.
										Heat accounted for.	Heat supplied.	Heat accounted for.	Heat supplied.	
			Grams/liter.	*Lbs./ft.*	*R.p.m.*	*Kg. cal./sec.*	*B.t.u./hr.*	*Kg. cal./sec.*	*B.t.u./hr.*					
A- 1	Ground		1.20	0.0752	1,421	18.5	264,000	146	2,090,000	34.6	27.7	57.2	46.0	19.6
2	do		1.20	.0748	1,642	20.7	295,000	161	2,300,000	37.2	28.4	55.0	42.1	23.5
3	do		1.20	.0747	1,814	27.2	388,000	179	2,560,000	36.7	27.7	53.2	40.2	22.7
4	do		1.19	.0744	1,979	32.5	464,000	205	2,930,000	34.0	29.0	55.5	47.4	14.6
5	do		1.19	.0741	2,192	23.8	340,000	205	2,935,000	35.9	29.0	58.6	47.2	19.4
11	do		1.20	.0752	1,600	29.9	426,000	182	2,600,000	32.0	27.2	55.9	47.7	14.6
12	do		1.19	.0746	1,796	25.9	369,000	185	2,640,000	35.4	26.8	55.5	42.1	24.2
B-25	do		1.22	.0760	1,793	30.2	431,000	191	2,735,000	33.5	27.9	55.6	46.5	16.3
26	do		1.18	.0734	1,774	23.2	331,000	174	2,485,000	36.4	31.3	55.4	47.8	13.7
27	do		1.14	.0712	1,800	30.2	430,000	190	2,710,000	33.4	31.3	55.5	52.1	6.2
28	do		1.11	.0693	1,817	26.1	372,000	172	2,460,000	36.6	31.2	53.8	45.9	14.6
	Km.	*Feet.*												
A-13	1.5	5,000	1.02	.0639	1,610	23.5	336,000	146	2,085,000	34.4	27.3	54.4	43.3	20.5
14			1.06	.0661	1,792	21.3	304,000	154	2,205,000	36.0	28.3	55.8	42.7	23.4
B-11			1.05	.0657	1,780	26.0	371,000	163	2,325,000	33.8	29.3	55.5	48.3	13.0
A-15	3.0	10,000	.90	.0564	1,601	22.2	317,000	119	1,700,000	34.8	25.2	52.5	38.1	27.4
16			.90	.0565	1,808	21.9	313,000	132	1,880,000	35.6	26.9	54.0	40.8	24.7
B- 6			.91	.0566	1,793	19.5	278,000	129	1,845,000	35.9	27.7	55.3	42.8	22.7
A-17	4.6	15,000	.76	.0474	1,591	18.0	256,000	97	1,383,000	35.2	25.9	53.0	38.1	28.0
18			.75	.0472	1,793	19.6	280,000	107	1,530,000	35.6	26.5	53.4	39.7	25.6
B- 1			.76	.0476	1,786	21.4	306,000	111	1,590,000	33.8	26.5	53.7	42.0	21.8
19			.77	.0481	1,783	21.0	300,000	112	1,600,000	33.8	25.7	54.2	41.4	23.7
21			.74	.0465	1,802	15.9	227,000	101	1,445,000	38.3	29.1	53.6	40.7	24.0
22			.72	.0450	1,757	14.3	204,000	94	1,345,000	39.8	24.6	52.9	32.8	38.0
23			.69	.0432	1,809	13.5	193,000	87	1,245,000	43.4	27.2	53.9	33.8	34.9
24			.68	.0422	1,797	13.5	192,000	90	1,280,000	41.5	26.7	52.2	33.6	35.6
A-19	6.1	20,000	.63	.0393	1,620	14.6	208,000	69	990,000	36.6	19.4	51.6	27.4	46.8
20			.64	.0400	1,818	15.9	227,000	88	1,265,000	35.2	23.8	55.3	37.3	32.5
A-21	7.6	25,000	.51	.0321	1,782	13.0	186,000	58	821,000	35.7	16.1	53.8	24.3	54.9
22			.53	.0329	1,599	12.5	179,000	51	724,000	37.2	16.5	50.2	22.3	55.4

TABLE 3.—*Summary of approximate heat balances.*

EJB=per cent of heat supplied in fuel which is accounted for in exhaust, jacket, and brake power.
E=per cent of EJB which is in exhaust.
J=per cent of EJB which is in jacket.
B=per cent of EJB which is in brake power.

Compression ratio.	Items.	Approximate altitude.							Engine—Hispano-Suiza or Liberty.	Test No.	Engine speed (approximate).	Fuel.
		Thousands of feet.										
		0	5-6	10-12	14-15	20	25	30				
		Kilometers.										
		0	1.7	3.5	4.5	6.1	7.6	9.2			*R. M. P.*	
4.7	EJB	91.5	84.0	85.6	....	86.3	....	97.0	H.-S. 150	31	1,500	X gas.
	E	50.3	55.2	55.4	....	52.2	....	53.0				
	J	23.9	17.3	18.1	....	22.6	....	24.8				
	B	25.8	27.6	26.5	....	25.1	....	22.2				
5.3	EJB	85.4	79.5	72.6	72.0	53.2	44.6	....	H.-S. 300	162	1,600	X gas.
	E	55.9	54.4	52.5	53.0	51.6	50.2					
	J	16.4	16.1	18.6	18.5	22.0	24.7					
	B	27.8	29.5	29.0	28.7	27.5	25.2					
5.3	EJB	....	79.3	....	87.6	....	80.9	....	H.-S. 180	163	1,600	X gas.
	E	....	52.2	....	51.0	....	55.3					
	J	....	18.4	....	22.4	....	18.3					
	B	....	29.4	....	26.8	....	26.5					
5.4	EJB	85.5	73.8	78.4	81.8	64.2	46.0	....	Liberty	160	1,600	X gas.
	E	55.1	53.2	54.1	51.3	51.8	55.2					
	J	17.3	17.7	18.3	22.1	19.0	19.8					
	B	27.6	29.2	27.6	26.7	29.2	24.4					
6.2	EJB	....	101.1	90.6	....	93.4	....	77.5	H.-S. 150	89	1,500	X gas.
	E	....	50.8	49.6	....	46.6	....	43.3				
	J	....	24.1	25.5	....	29.6	....	35.3				
	B	....	25.1	24.9	....	24.0	....	21.5				
6.3	EJB	....	73.3	....	71.1	....	69.7	....	H.-S. 180	166	1,600	X gas.
	E	....	51.5	....	48.0	....	43.2					
	J	....	18.7	....	22.4	....	28.0					
	B	....	29.9	....	29.7	....	28.9					
7.2	EJB	....	90.1	....	98.4	....	89.3	....	Liberty	152	1,600	X gas.
	E	....	49.2	....	50.4	....	47.4					
	J	....	22.0	....	20.9	....	25.9					
	B	....	29.0	....	28.8	....	26.8					
7.2	EJB	85.0	85.7	....	82.8	....	82.1	....	Liberty	152	1,600	Hecter.
	E	50.4	50.4	....	48.4	....	46.5					
	J	18.7	18.6	....	22.4	....	26.6					
	B	31.0	31.1	....	29.4	....	27.1					
7.3	EJB	....	....	....	94.0	....	79.7	....	H.-S. 180	165	1,600	X gas.
	E	....	....	....	50.2	....	47.8					
	J	....	....	....	21.8	....	23.7					
	B	....	....	....	28.1	....	28.5					
8.3	EJB	....	....	....	77.3	....	78.7	....	H.-S. 180	164	1,600	X gas.
	E	....	....	....	46.6	....	43.0					
	J	....	....	....	22.0	....	27.0					
	B	....	....	....	31.5	....	30.2					

REPORT No. 109

EXPERIMENTAL RESEARCH ON AIR PROPELLERS, IV

By W. F. DURAND and E. P. LESLEY
Leland Stanford Junior University, California

REPORT No. 109.

EXPERIMENTAL RESEARCH ON AIR PROPELLERS IV.

By W. F. Durand and E. P. Lesley, Leland Stanford Junior University.

INTRODUCTION.

The purpose of the investigations on the performance of air propellers described in the following report was the extension in certain directions of the field covered by the previous investigations (Reports Nos. 14, 30, 64) and the testing of certain special forms.

The forms included under the present report are characterized as follows:

Number.	Diameter.	Pitch.	Mean blade width.	Shape of blade.	Blade section.
	Inches.	*Inches.*			
139	36	10.8	0.15r	2	Noncambered.
142	36	32.4	.15r	Special.	Do.
143	36	32.4	.15r	Special.	Do.
144	36	10.8	.20r	2	Do.
145	36	10.8	.15r	1	Do.
146	36	10.8	.20r	1	Do.
148	36	25.2	.25r	2	Do.
150	36	25.2	.25r	2	Do.
151	36	(1)	.15r	2	Do.
152	36	(2)	.15r	2	Do.

1 Plane driving face 17°–10 feet. 2 Plane driving face 21°–40 feet.

It will be noted that propellers 139, 144, 145, 146 are similar in all respects to Nos. 3, 4, 1, 2 of Report No. 14 except in pitch ratio which is here 0.3. These forms thus serve to extend to this pitch ratio the various results covering thrust and torque coefficients and efficiency, for the four families representing the combination of these two blade forms and blade areas. These results, together with those previously tested, thus give a series of values in each of the four families for the six nominal pitch ratios, 0.3, 0.5, 0.7, 0.9, 1.1, 1.3.

Propeller 148 is of form No. 2 (Report No. 14), or with curved and somewhat tapering outline. Its mean width, however, is 0.25r or 25 per cent greater than the A_2 of Report No. 14. Propeller 150 is of form No. 1 (Report No. 14), or with nearly straight parallel sides, and has also a mean width of 0.25r. These two propellers serve, with the two mean widths previously reported on, to give two series of values for varying blade area corresponding to mean widths, respectively, of 0.15r, 0.20r, 0.25r.

Propellers 151, 152 are of form and area the same as F, A, of Report No. 14, but with face flat or unwarped. For propeller 151, the blade is set at such an angle as to give, at the 13-inch radius, a pitch of 25.2 inches, or the same as a 0.7 pitch ratio propeller of 36 inches diameter. For propeller 152 the blade is set to give similarly, at the 13-inch radius, a pitch of 32.4 inches, or the same as a 0.9 pitch ratio propeller of 36 inches diameter.

Propellers 142 and 143 have a section and pitch identical with propeller No. 1 of Report No. 14. They have contours, however, as shown on plates I, II, the sections being distributed along a cycloidal curve instead of a radial line, as in the case of No. 1. These forms were intended to test certain ideas submitted to the committee by Brig. Gen. H. H. C. Dunwoody (retired).

RESULTS OF TESTS.

The various results are given in the form of curves of the thrust and torque coefficients T and Q and efficiency ρ.

The thrust and torque coefficients are defined by the equations

$$T_c = \frac{gT}{\Delta V^2 D^2} \tag{1}$$

$$Q_c = \frac{gQ}{\Delta V^2 D^3} \tag{2}$$

$$\rho = \frac{TV}{2\pi NQ} \tag{3}$$

The introduction of the factor g serves to render the coefficients nondimensional and thus independent of the system of units employed, provided they are homogeneous.

The values of T_c, Q_c, and ρ thus defined are given on curve sheet diagrams Nos. III to XII.

DISCUSSION OF RESULTS.

As would be expected, the efficiencies given by the propellers of 0.3 pitch ratio are so displaced relative to the values of V/ND that for low values of this abscissa the efficiency is higher and for high values lower, as compared with the efficiencies with forms of higher pitch ratio. The maximum efficiency reached, however, is definitely less than those for higher pitch ratios, thus falling in general line with the results for the series of pitch ratios 0.5, 0.7, 0.9, 1.1, 1.3.

The wide blade forms 148 and 150 have efficiency curves falling somewhat below those with lesser area but with similar forms otherwise, thus indicating the increasing influence due to skin resistance. These efficiency curves have, however, a somewhat lengthened range, corresponding to the increased value of the dynamic pitch.

Forms 151 and 152 show efficiencies definitely below those for the most nearly related helicoidal forms. Over the working range, however, the differences are relatively small and the tests indicate the close degree of approach to helicoidal forms realized by these nonwarped blades.

Propellers 142 and 143 show efficiency torque and thrust curves very similar to those for propeller No. 1. Propeller No. 142 having a convex leading edge, bends under load in such manner as to decrease the pitch. This operates to give, as it should, a somewhat lower maximum efficiency than for No. 1, but somewhat higher efficiencies for large values of the slip (low values of V/ND).

On the other hand, propeller No. 143 with concave leading edge increases in pitch when under load with the corresponding result of showing a somewhat higher maximum efficiency as compared with No. 1, but definitely lower efficiencies over the working range of moderate to small values of V/ND.

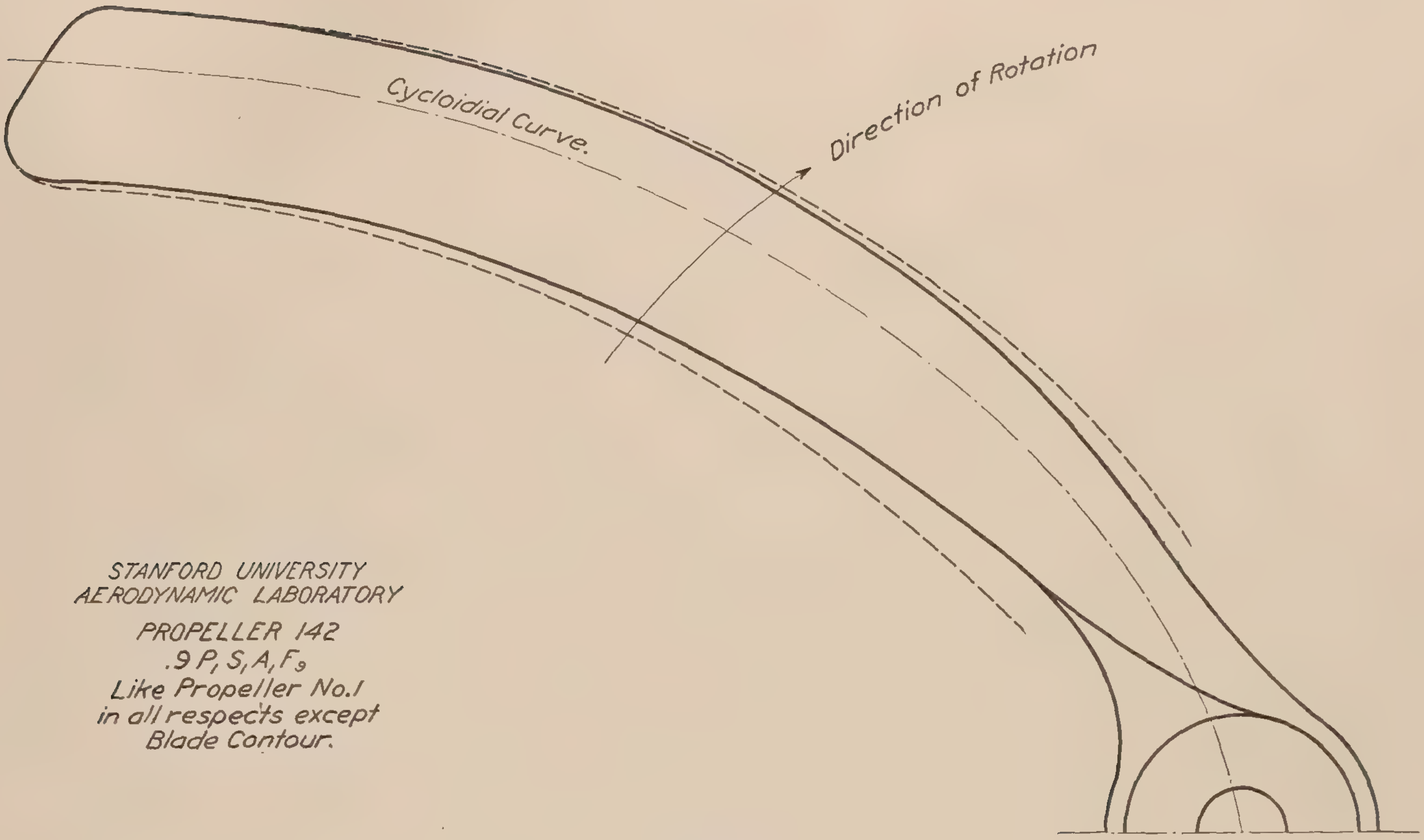


PLATE I.

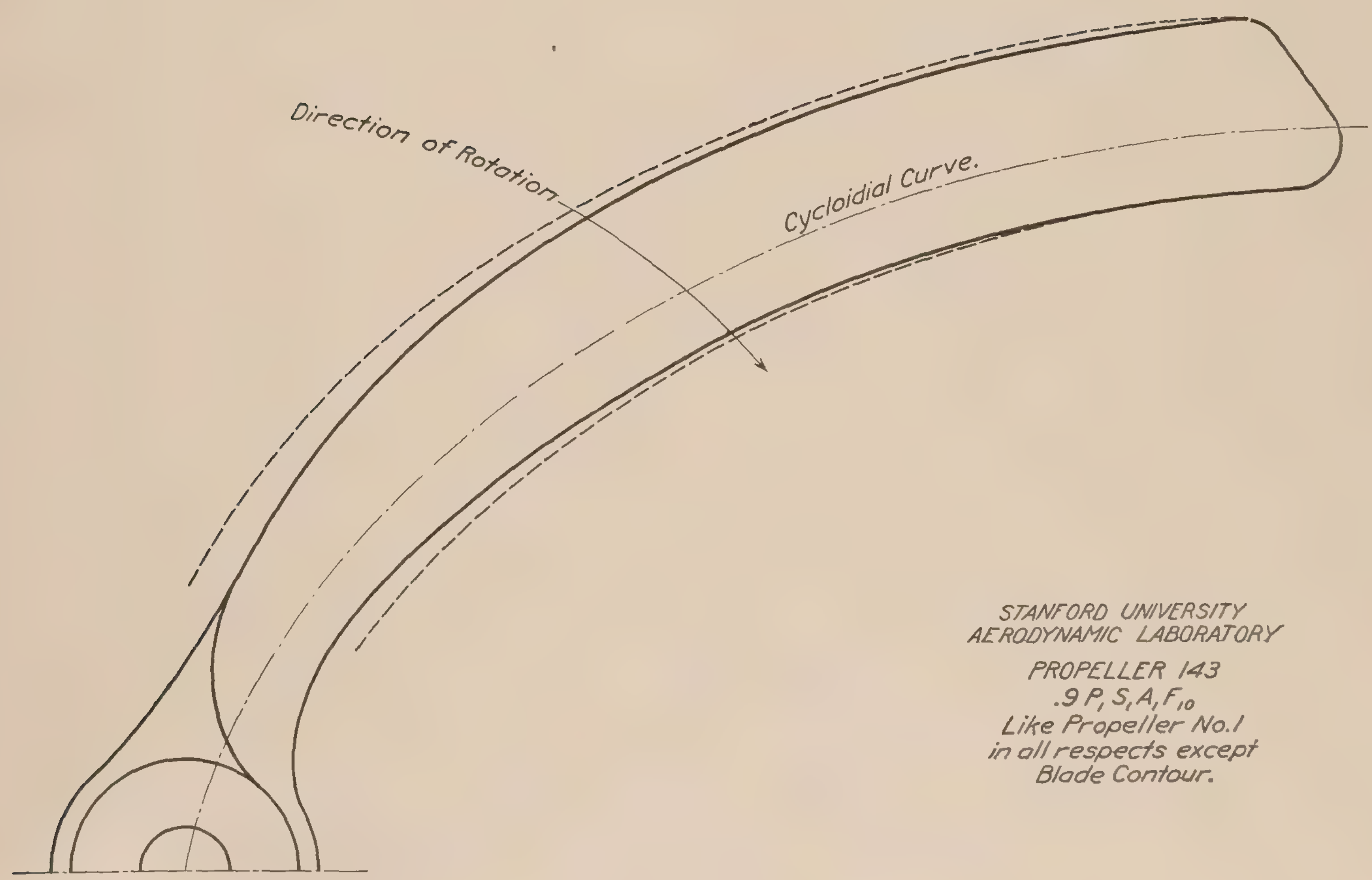


PLATE II.

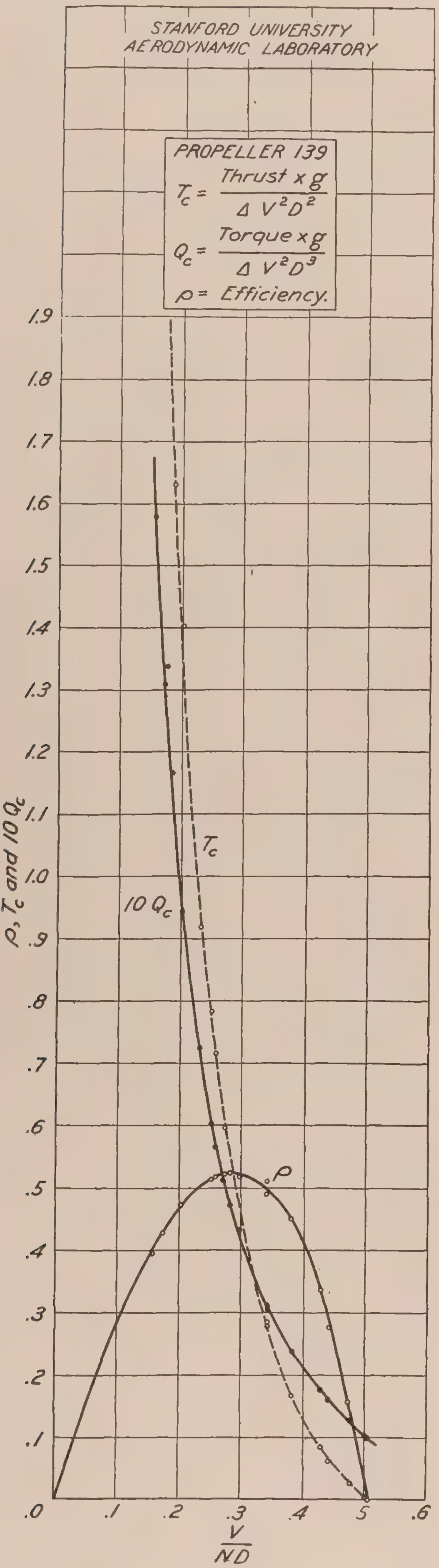


PLATE III.

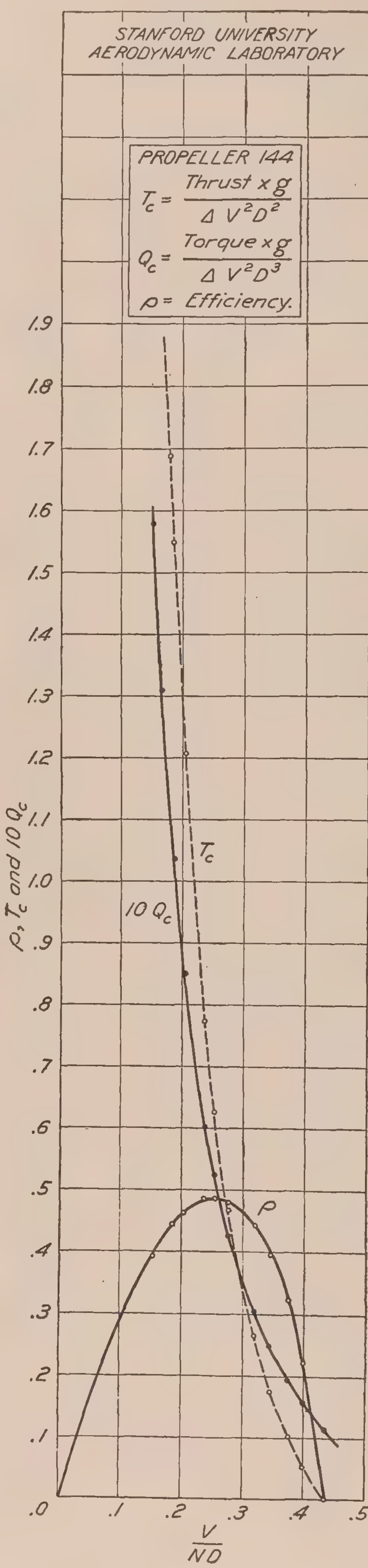


PLATE IV.

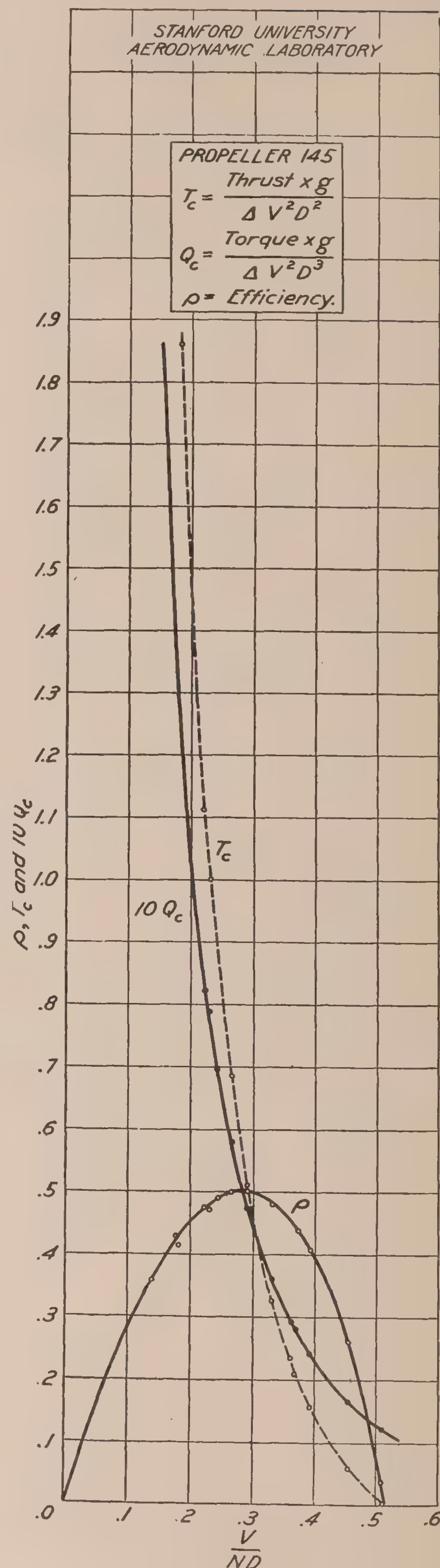


PLATE V.

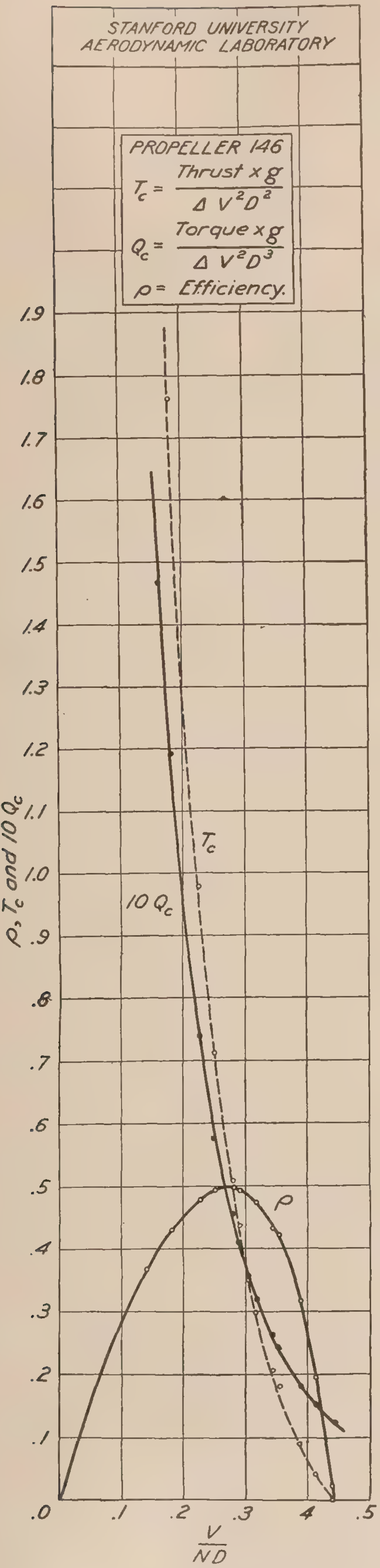


PLATE VI.

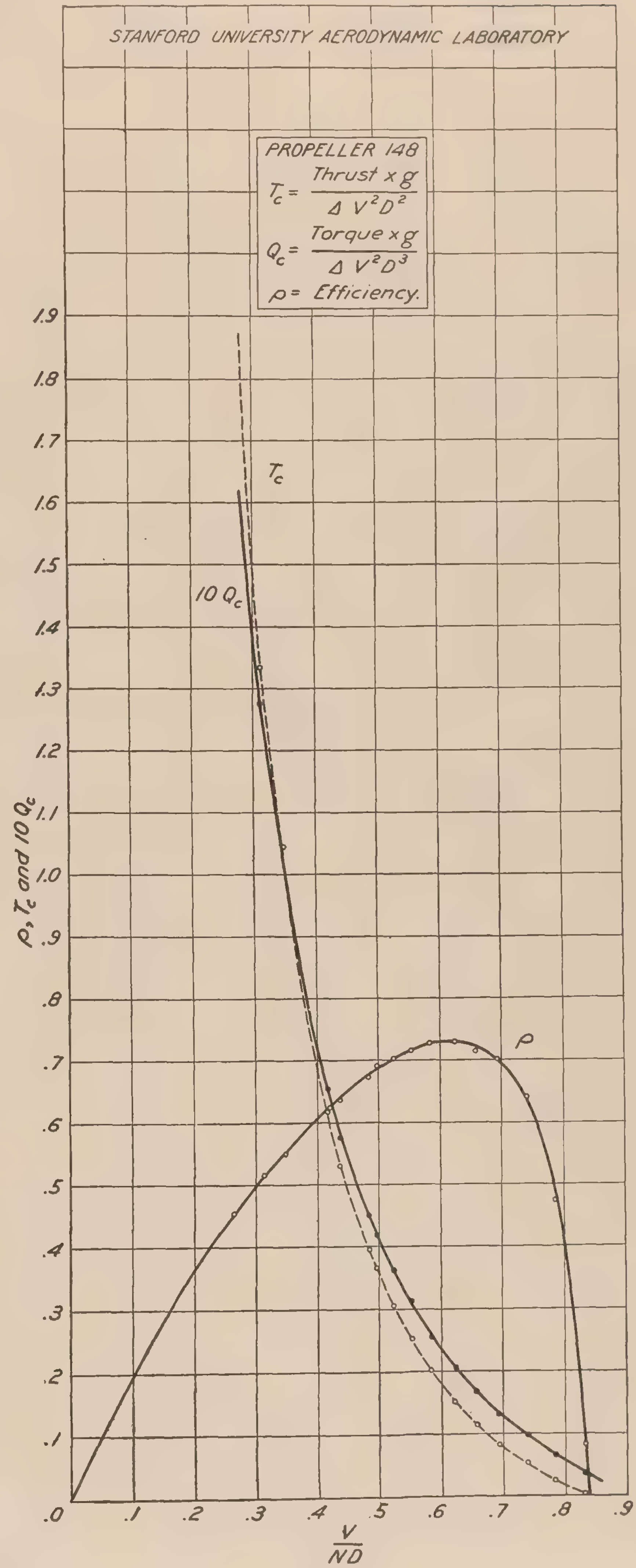


PLATE VII.

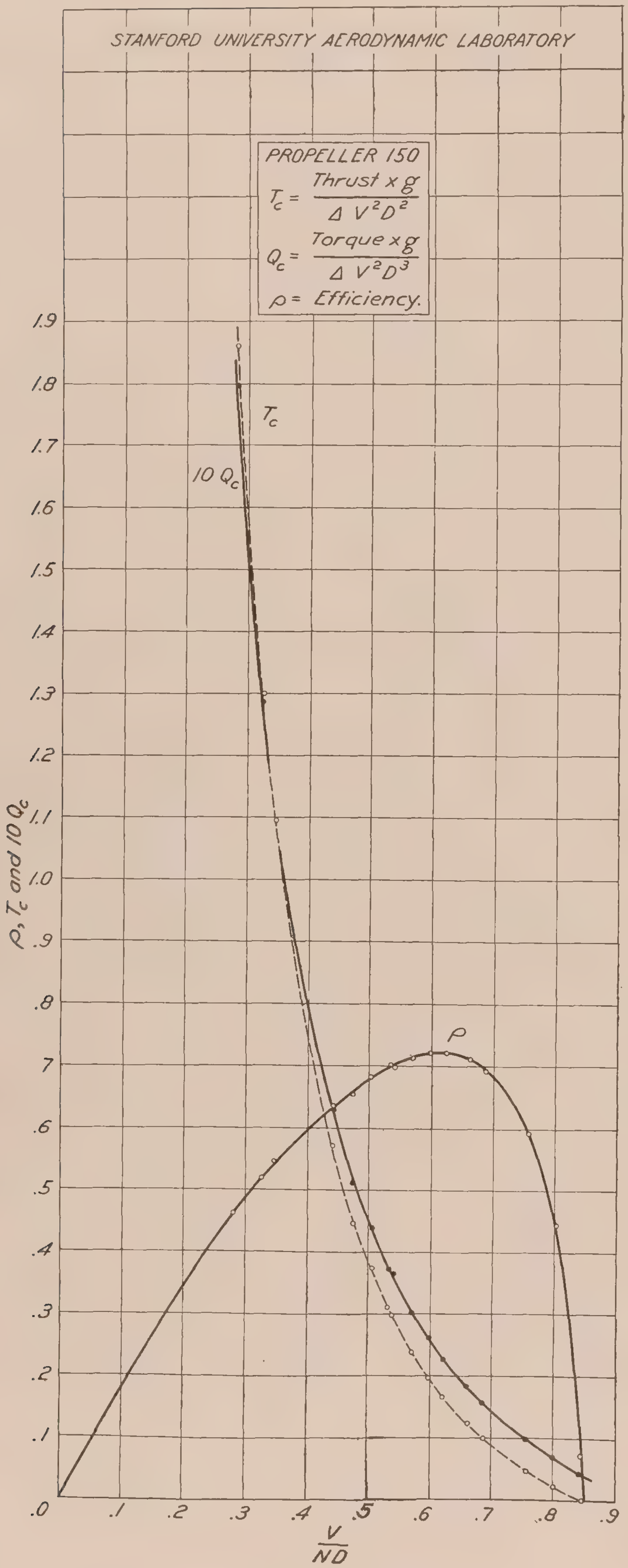


PLATE VIII.

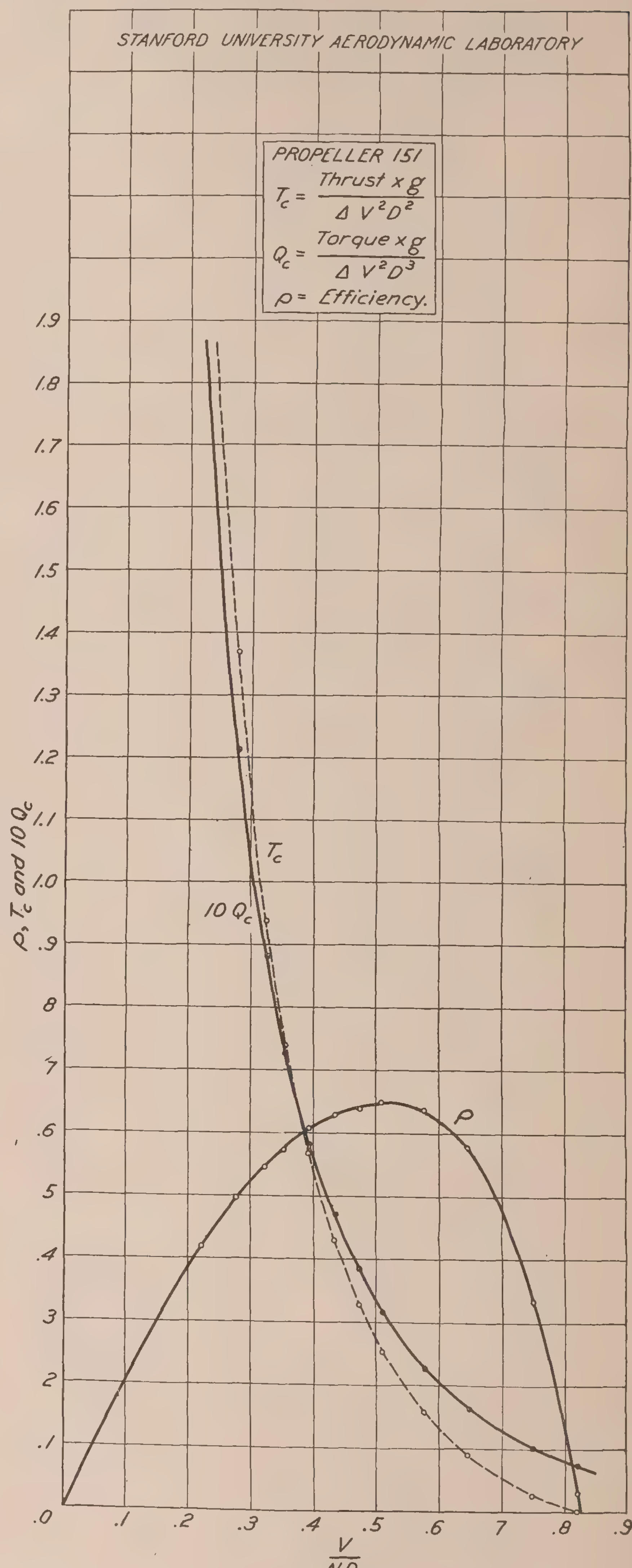


PLATE X.

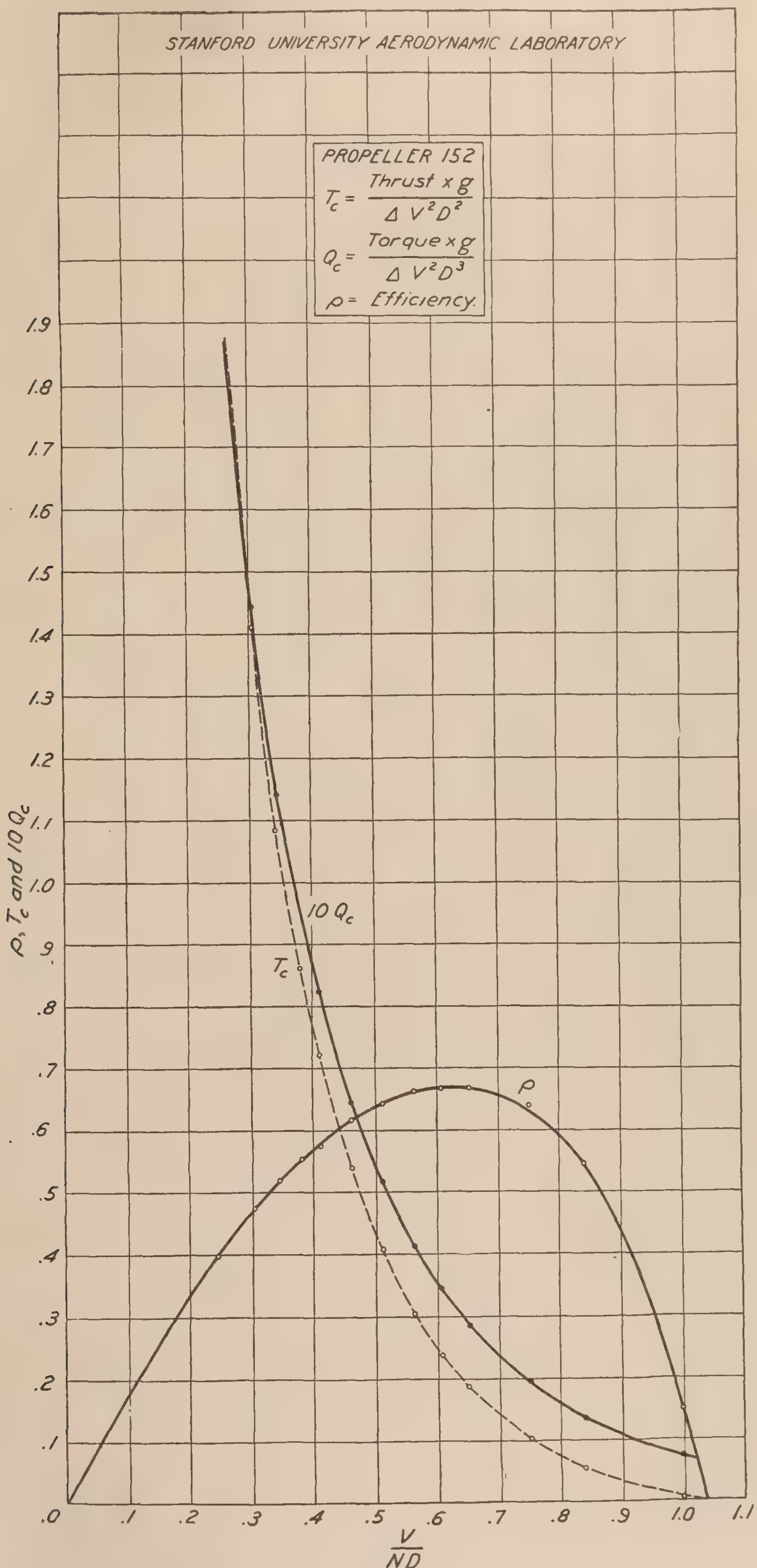

STANFORD UNIVERSITY AERODYNAMIC LABORATORY
PROPELLER 152
$T_c = \frac{\text{Thrust} \times g}{\Delta V^2 D^2}$
$Q_c = \frac{\text{Torque} \times g}{\Delta V^2 D^3}$
ρ = Efficiency.
1.9
1.8
1.7
1.6
1.5
1.4
1.3
1.2
1.1
1.0
.9
.8
.7
.6
.5
.4
.3
.2
.1
.0
ρ, T_c and $10\,Q_c$
$10\,Q_c$
T_c
ρ
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
1.1
$\frac{V}{ND}$

PLATE IX.

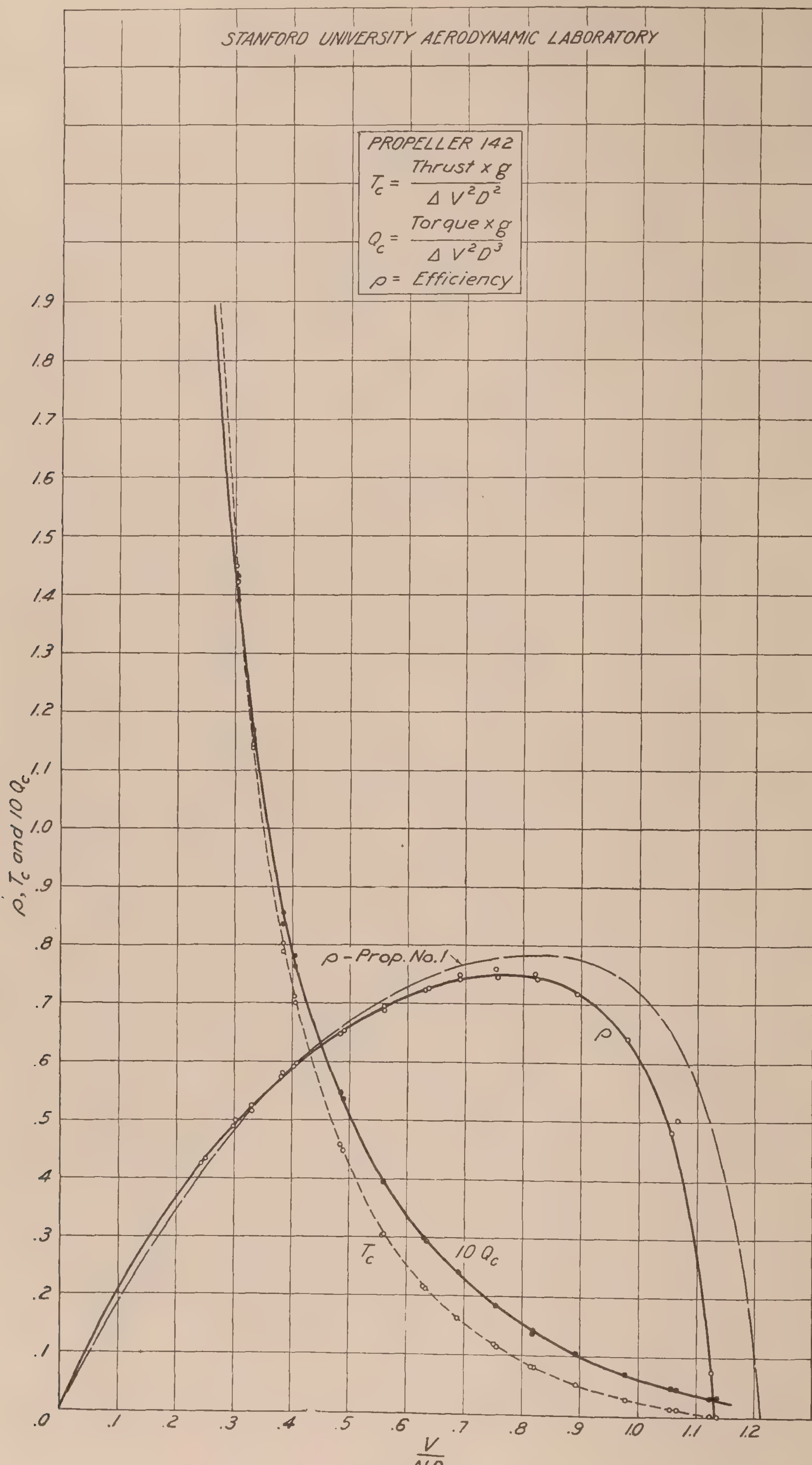

STANFORD UNIVERSITY AERODYNAMIC LABORATORY
PROPELLER 142
$T_c = \frac{\text{Thrust} \times g}{\Delta V^2 D^2}$
$Q_c = \frac{\text{Torque} \times g}{\Delta V^2 D^3}$
ρ = Efficiency
ρ, T_c and 10 Q_c
ρ - Prop. No. 1
ρ
T_c
10 Q_c
$\frac{V}{ND}$
.0 .1 .2 .3 .4 .5 .6 .7 .8 .9 1.0 1.1 1.2
.1 .2 .3 .4 .5 .6 .7 .8 .9 1.0 1.1 1.2 1.3 1.4 1.5 1.6 1.7 1.8 1.9

PLATE XI.

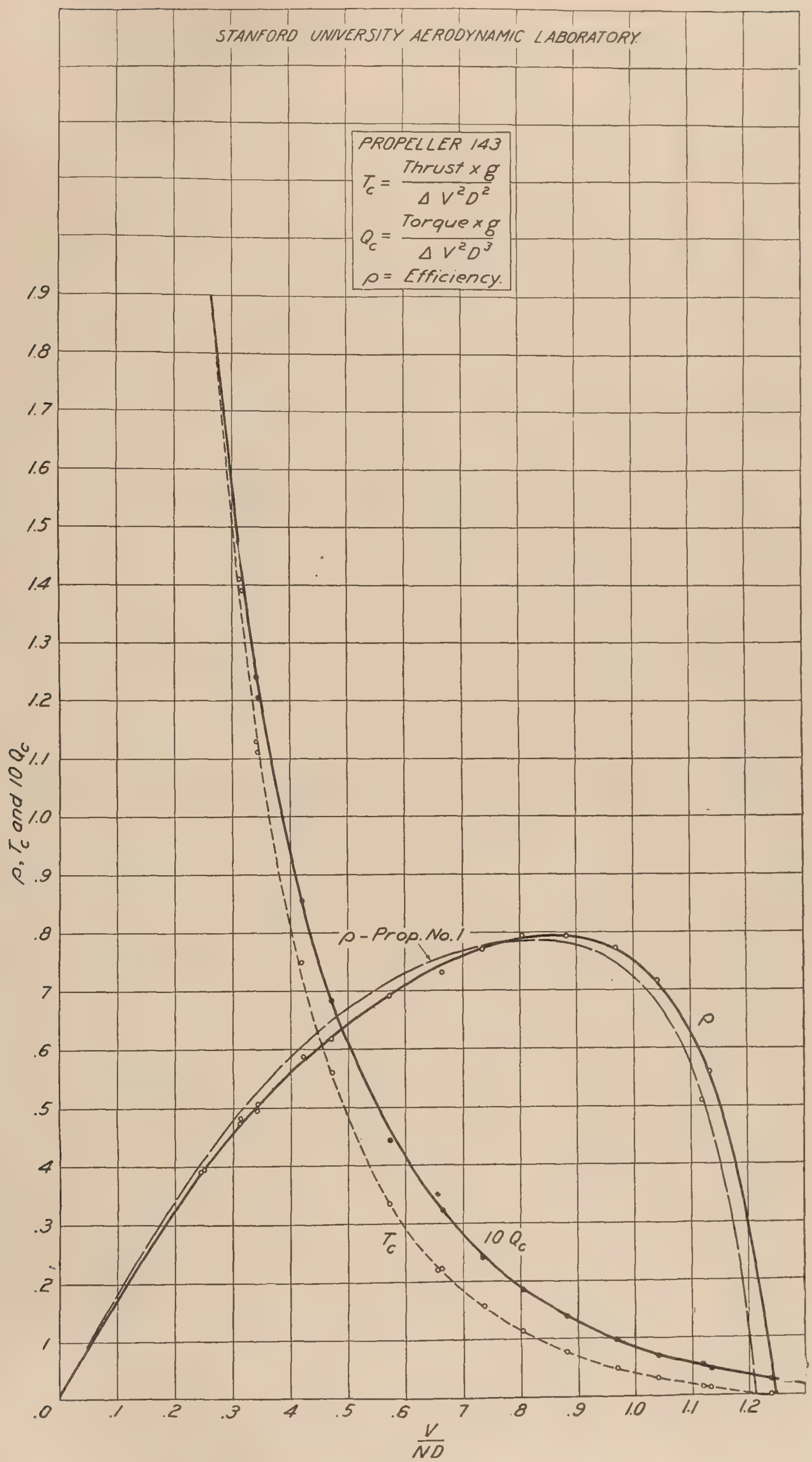

STANFORD UNIVERSITY AERODYNAMIC LABORATORY.
PROPELLER 143
$T_c = \frac{\text{Thrust} \times g}{\Delta V^2 D^2}$
$Q_c = \frac{\text{Torque} \times g}{\Delta V^2 D^3}$
ρ = Efficiency.
ρ, T_c and $10\,Q_c$
ρ - Prop. No. 1
ρ
T_c
$10\,Q_c$
$\frac{V}{ND}$
.0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
1.1
1.2
1.3
1.4
1.5
1.6
1.7
1.8
1.9

PLATE XII.

REPORT No. 110

THE ALTITUDE EFFECT ON AIR SPEED INDICATORS

IN TWO PARTS

By MAYO D. HERSEY, FRANKLIN L. HUNT, and HERBERT N. EATON
Bureau of Standards

FIG. 1.—VENTURI TUBES.

REPORT No. 110.

THE ALTITUDE EFFECT ON AIR SPEED INDICATORS.

By M. D. Hersey, F. L. Hunt, and H. N. Eaton.
(Bureau of Standards.)

PART I.

THEORETICAL INTRODUCTION.

This report was begun in the fall of 1917 in connection with the testing of air speed indicators by the Bureau of Standards, and prepared at the request of the National Advisory Committee for Aeronautics.

1. OUTLINE OF INFORMATION REQUIRED.

In order to convert the readings of an air speed indicator, after correcting for ordinary instrumental errors, into true air speed, two steps are necessary:

(*a*) Arbitrary designation of standard atmospheric conditions near sea level.
(*b*) Determination of the effect on the performance of the instrument, of the departure from those conditions which may be met at any altitude.

In order to take the first step (*a*), it is necessary to have a complete enumeration of the conditions which influence the aerodynamic performance. It would not be enough to specify standard density, unless it were known that changes of density alone could modify the performance at a given speed.

To carry out the second step (*b*), it is necessary to secure, theoretically or experimentally, such information as would provide data for drawing up a complete set of curves connecting the performance of the instrument, on the one hand, with each of the variables governing its performance, on the other hand.

For example, in the case of the Pitot-Venturi type, a family of curves would be necessary connecting the differential pressure with speed, air density, air viscosity, and any other factors which appreciably alter the differential pressure.

Precisely similar information is desirable for all the other types of air speed indicators. These instruments, so far as aerodynamic performance is concerned, may be classified somewhat as follows:

1. Rotating surface type (Morell, etc.).
2. Direct impact type.
   (*a*) With surface in a fixed direction (pressure plate, etc.).
   (*b*) With variable direction of surface. (Pensuti, etc.)
3. Differential pressure type.
   (*a*) Pitot tube (with static or with suction openings).
   (*b*) Venturi tube (with or without static openings; single or double throat).
   (*c*) Pitot-Venturi (Toussaint-Lepère; Zahm nozzle, U. S. Navy; modified Zahm nozzle, U. S. Army).
4. Air flow type (Prouty).
5. Nonmechanical types (hot wire, etc.).

Descriptive details of all these types are to be found in another paper.[1]

[1] General Report on Aeronautic Instruments, Investigation of Air Speed Indicators, by F. L. Hunt: National Advisory Committee for Aeronautics, to be published later.

This investigation was undertaken to supply, in some measure, the information outlined above as desirable. The extent to which such information was previously available is indicated in the following section dealing with the assumptions customarily made.

2. ASSUMPTIONS CUSTOMARILY MADE.

It has ordinarily been assumed that the density alone is sufficient to fix the standard atmospheric condition, (*a*), and that as regards departure from this condition, (*b*), the indication of the instrument at a given speed is directly proportional to the density for air speed indicators of the direct impact and differential pressure types and practically independent of density for indicators of the rotating vane type. In fact, the direct impact and differential pressure types are colloquially, though not scientifically, spoken of as "ρv^2" instruments and the rotating vane type as "true air speed" instruments.

The ρv^2 assumption is fairly satisfactory for direct impact and for Pitot tube instruments, but less so for Venturi tubes. In the case of the Pitot, it is recognized that a more accurate result can be deduced at the higher speeds by allowing for adiabatic compression of the air; and in the case of the Venturi, it is found by the present experiments that the viscosity of the air has to be taken into account at the lower speeds and higher altitudes.

Disregarding compressibility and viscosity, the correction for altitude is customarily made by the formula

$$v=\frac{v_i}{\sqrt{r}} \tag{1}$$

in which v is the true and v_i the indicated speed and r the relative density at the level in question, i. e.,

$$r=\frac{\rho}{\rho_0} \tag{2}$$

where ρ is the actual and ρ_0 the standard density. At 20,000 feet the relative density is about one-half, so for that altitude $v=v_i\sqrt{2}$ or the true speed is about 40 per cent greater than the indicated speed.

For graduating the dials to go with the Pitot-Venturi tubes of American manufacture, such as the United States Army tube illustrated in figure 1, the Zahm nozzle formula [2]

$$p=.00313\ v^2 \tag{3}$$

is employed, in which v denotes the speed in miles per hour corresponding to a differential pressure p inches of water.

This formula assumes for the standard air density 1.221×10^{-3} gms/cm³, which is the density dry air would possess at a temperature of 16° C. under the normal sea level pressure of 760 mm. mercury.

3. LIMITATIONS OF DEDUCTIVE THEORIES.

By means of thermodynamic reasoning, treating the atmosphere as an ideal gas, it is possible to throw the Pitot tube formula into a more general one

$$p=\tfrac{1}{2}\ \rho v^2\ (1+C) \tag{4}$$

in which the correction term C stands for a certain complicated function which vanishes for a perfectly incompressible fluid, or for zero speed. For a speed of 110 miles per hour C is of the order of 0.5 per cent and should not be neglected if the most probable result is required; and yet so many simplifying assumptions have to be introduced in the derivation that the result is not entirely free from doubt.[3]

---

[2] Strictly, $v=17.88\ \sqrt{p}$.

[3] E. Buckingham, on the Theory of the Pitot tube, Technical Report No. 2, National Advisory Committee for Aeronautics, 1915.

In just the same way a thermodynamic formula for the Venturi tube can be deduced, which attempts to take account of the geometrical shape of the tube and the compressibility of the air: but in this case the simplifying assumptions which have to be introduced are so appalling, that the result is only of academic interest. For example, it has to be assumed that the fluid is free from turbulence and at the same time without viscosity: and all disturbances outside of the tube have to be ignored. This means that the problem has to be treated like that of a continuous hydraulic pipe line, and that two instruments having the same interior channel would generate the same suction at the throat, no matter if one of them had a very much bigger bulge on the outside, such as to drag a lot of air along with it—a conclusion contrary to experience.

At best the thermodynamic formula could only hope to show the effect of compressibility, so far as properties of the fluid are concerned, since viscosity and turbulence are excluded at the start. Yet the experiments which will be reported here prove that the compressibility effect is a comparatively negligible one. Thus the Venturi tube problem is too complicated to be handled with advantage by purely deductive methods at present.

4. DIMENSIONAL THEORY.

A fruitful compromise between purely deductive theory on the one hand and interminable experimenting on the other hand, is made possible by dimensional reasoning.[4]

Consider, first, the Venturi tube. Let p denote the differential pressure (i. e., suction) generated at a speed v, the tube being pointed head on (i. e., without yaw) into a perfectly undisturbed medium (the atmosphere, for example, or water). It is understood that the speed v has remained constant for some moments so that a steady state is established. Let the mechanical properties of the medium be specified by its density ρ, viscosity μ, and compressibility modulus of elasticity E. On account of the rapid movements involved, it will be the adiabatic, not the isothermal, elasticity which is needed.

The three properties ρ, μ, and E are defined in the usual way. Thus the density is the mass per unit volume,

$$\rho \equiv \frac{M}{V} \tag{5}$$

The viscosity is the shearing stress per unit rate of shear, i. e.,

$$\mu \equiv \frac{f}{\left(\frac{dv}{dy}\right)} \tag{6}$$

in which f is the tangential force per unit area, or shearing stress, brought into play by distorting or shearing the fluid at such a rate that a velocity gradient $\frac{dv}{dy}$ is set up. Here dv is the difference in speed between the top and bottom surface of a layer dy units thick. The elasticity is the increase of hydrostatic pressure P per unit decrease of volume measured as a fraction of the volume V, i. e.,

$$E \equiv \frac{dP}{-\left(\frac{dV}{V}\right)} = -V\frac{dP}{dV} \tag{7}$$

The medium will be assumed homogeneous so that ρ, μ, and E have the same values at all points, both inside and outside the nozzle. This is admittedly an approximation, for the fluid is slightly warmer where most compressed, and so all of its constants are a trifle different at such a point. Finally let D stand for any agreed-upon linear dimension of the nozzle, as for example, the throat diameter.

Under these circumstances the differential pressure p evidently depends on the speed v, on the mechanical properties of the fluid, ρ, μ, and E, and on the absolute size D and the geo-

[4] Cf. E. Buckingham: Dimensional Theory of Wind Tunnel Experiments, Smithsonian Misc. Papers, v. 62-4, pp. 15-26, 1916; Model Experiments and the Form of Empirical Equations, Transactions A. S. M. E., v. 36, pp. 263-296, 1915.

metrical shape of the nozzle. Under the conception of shape are to be included the roughness of the surfaces and the contour of all adjacent parts that may cause disturbance. If no further physical quantities are apparent which can sensibly influence the phenomenon, then some relation

$$p = \text{funct}\ (v, \rho, \mu, E, D) \tag{8}$$

must exist, the specific form of which remains to be discovered by experiment, but which will be the same for all geometrically similar systems.

The object of our investigation, so far as Venturi tubes are concerned, consists in determining the form of that equation; and there will be an analogous equation for every other type of air speed indicator. It has to be done experimentally, but dimensional reasoning serves to simplify the planning of the experiments, and the interpretation of the observations.

Since equation (8) is physically complete, it must, when written out in full, have the same dimensions on both sides. The dimensions of the constituent quantities are as follows, taking mass (m), length (l), and time (t) for the necessary fundamental units:

$$\begin{aligned} p &: m\, l^{-1}\, t^{-2} \\ v &: l\, t^{-1} \\ \rho &: m\, l^{-3} \\ \mu &: m\, l^{-1}\, t^{-1} \\ E &: m\, l^{-1}\, t^{-2} \\ D &: l \end{aligned}$$

It can now be shown by the π-theorem (Buckingham, loc. cit.) or verified by inspection that the only form (8) can take, which will meet the requirement for dimensional homogeneity, is identical with, or reducible to, the general equation

$$p = \rho v^2\ \text{funct}\left(\frac{Dv\rho}{\mu}, \frac{E}{\rho v^2}\right) \tag{9}$$

It will be useful to write out two modifications of this equation. The velocity of sound in a fluid, C, is given by the well-known expression

$$C = \sqrt{\frac{E}{\rho}} \tag{10}$$

Hence E, where it occurs in (9) above, can equally well be replaced by ρC^2, so (9) can be rewritten

$$\frac{p}{\rho v^2} = \varphi\left(\frac{Dv\rho}{\mu}, \frac{C}{v}\right) \tag{11}$$

using the symbol ϕ to denote some unknown function of the *two* arguments, or independent variables, inside the parenthesis. This equation is of interest in connection with the water-channel experiments, to be described later.

Again, it is a well-known thermodynamic result that

$$E = \kappa P \tag{12}$$

for an ideal gas, κ being the specific heat ratio (about 1.4) and P the barometric pressure. (This relation follows from (7) in conjunction with the adiabatic compression equation $PV^{\kappa} = \text{const.}$) Replacing E by its equivalent κP in (9) gives

$$\frac{p}{\rho v^2} = \psi\left(\frac{Dv\rho}{\mu}, \frac{\kappa P}{\rho v^2}\right) \tag{13}$$

The unknown functions ϕ and ψ are different, though they might have been kept identical by writing $\left(\frac{C}{v}\right)^2$ instead of $\frac{C}{v}$ in (11). This equation, (13), is of interest in connection with the observations in a wind stream at reduced barometric pressure, which remain to be described.

It is essential to realize that in equations (11) and (13) there are only *two* independent variables, not *five*, as in (8). Thus in equation (13) the dependent variable $\frac{p}{\rho v^2}$ is expressed as a function of the two independent variables $\frac{Dv\rho}{\mu}$ and $\frac{\kappa P}{\rho v^2}$. Readers not accustomed to this point of view may be helped by a change of notation: Write y for $\frac{p}{\rho v^2}$, x for $\frac{Dv\rho}{\mu}$, and z for $\frac{\kappa P}{\rho v^2}$ Then (13) becomes simply

$$y = \psi\ (x, z) \tag{14}$$

an ordinary surface in three coordinates. It is by such a surface (or family of plane curves) that the experimental observations ought to be depicted, instead of attempting to separate out the original quantities as in (8). They can be separated later, after the best possible empirical expression (14) has been fitted to the plotted points.

By means of the equations just deduced, especially (11) and (13), a comparatively economical program of experimental work can readily be laid out.

The foregoing analysis applies without change of notation to all differential pressure instruments of rigid shape. To extend it to direct impact instruments requires that the performance be expressed by F, the total force acting, instead of by the differential pressure p. Referring to equation (9), replace p by $\frac{F}{D^2}$ to preserve the dimensions unchanged and the result becomes

$$F = \rho v^2 D^2 \text{ funct} \left(\frac{Dv\rho}{\mu},\ \frac{E}{\rho v^2}\right) \tag{15}$$

This is the general equation for a pressure-plate air speed indicator. Except for extraordinarily low speeds the viscosity can not enter very seriously, consequently as an approximation which is safer the higher the speed,

$$F = \rho v^2 D^2 \text{ funct} \left(\frac{E}{\rho v^2}\right) \tag{16}$$

For since $\frac{Dv\rho}{\mu}$ is a *single* argument, if a change in viscosity causes no change in force, nothing else that controls the magnitude of $\frac{Dv\rho}{\mu}$ can do so either; hence the whole argument drops out. For speeds below, say, 150 miles an hour, where there is not much compression, (16) reduces to

$$F = \text{const} \times \rho v^2 D^2, \tag{17}$$

an example of the ρv^2 law. The constant is the same for pressure plates of different sizes provided they are strictly geometrically similar in all essential parts—including the sharpness of the edges, and proximity of the connections—and also provided, as was stated in the beginning, that the instrument is moving head on into an undisturbed atmosphere. Probably these conditions can be more easily fulfilled for pressure plates, and for Pitot tubes, than they can for Venturi tubes.

When the direction of the surface is not fixed, but free to change under increasing force of impact subject to the control of a spring as in the Pensuti air speed indicator, the problem is not so simple. The stiffness of the spring, S (force per unit displacement), now enters as an additional variable, so that (15) has to be expanded into the form

$$F = \rho v^2 D^2 \text{ funct} \left(\frac{Dv\rho}{\mu},\ \frac{E}{\rho v^2},\ \frac{S}{\rho v^2 D}\right) \tag{18}$$

In the ordinary case where viscosity and compressibility are negligible this reduces to

$$F = \rho v^2 D^2 \text{ funct} \left(\frac{S}{\rho v^2 D}\right) \tag{19}$$

instead of to (17). Some interesting conclusions applicable to the Pensuti and similar instruments can at once be drawn from this equation.

Let X stand for the new argument $\frac{S}{\rho v^2 D}$. Then $f(X)$ must be some function which starts at the origin and approaches asymptotically a maximum value, for an infinitely stiff spring, equal to the constant of equation (17); for equation (19) should reduce to (17) if the geometrical shape is constant. Hence in general the force (and therefore deflection) varies with speed less rapidly than the square, and with density less rapidly than the first power.

The altitude effect on instruments of the Pensuti class can now be deduced from observations made at sea level with varying speed. Suppose that a series of such observations gave

$$F \propto v^n \tag{20}$$

where n is some numerical value probably between 1 and 2. Then since (20) must be a special case of (19),

$$f(X) = X^{1-\frac{n}{2}} \tag{21}$$

$$\therefore\ F = \text{const} \times \rho^{\frac{n}{2}} v^n D^{1+\frac{n}{2}} S^{1-\frac{n}{2}} \tag{22}$$

whence

$$F \propto \rho^{\frac{n}{2}} \tag{23}$$

Thus the observation that the force varies with the nth power of the speed, leads by virtue of (19) to the inference that the force would vary with the $\frac{n}{2}$ power of the density, and with certain other powers of the size D and spring stiffness S.

By going back and differentiating equation (19), a still more general relation for the altitude effect in terms of the speed effect can be obtained, namely,

$$\frac{\partial F}{\partial \rho/\rho} = \frac{1}{2}\frac{\partial F}{\partial v/v} \tag{24}$$

This applies to all direct impact instruments operating over the intermediate range of speeds where neither viscosity nor compressibility have to be considered. Like other results afforded by dimensional reasoning, it is not limited by any assumption or restriction as to the geometrical complexity of the instrument or the irregularity of the motion set up in the fluid.

5. EXPERIMENTAL PROGRAM.

The experiments to be reported in this publication all relate to Venturi tubes and may be grouped as follows:

(*a*) Water channel experiments to determine the degree of dynamical similarity attainable between air and water, and to discover whether compressibility has to be taken into account.

(*b*) Observations in a wind stream at reduced pressure so as to determine the effect of density and viscosity by direct experiment.

(*c*) Airplane observations as a practical check on the foregoing laboratory results.

(*d*) Ordinary wind tunnel tests.

The need for these various experiments and the inferences possible from each can be readily seen in the light of the dimensional theory which has just been developed.

It was thought that for some purposes water channel observations on air speed nozzles would be more convenient than wind tunnel tests provided a reasonable degree of dynamical similarity proved attainable. From equation (11) neglecting compressibility the condition for similarity is found to be

$$\left(\frac{Dv\rho}{\mu}\right)_{\text{water}} = \left(\frac{Dv\rho}{\mu}\right)_{\text{air}}. \tag{25}$$

This condition can be fulfilled by towing the nozzle through the water at a speed slow enough to compensate for the relatively lower value which the kinematic viscosity $\frac{\mu}{\rho}$ has in water compared to air. Thus, denoting kinematic viscosity by ν, and distinguishing the water channel observations by primes, equation (25) reduces to

$$\frac{v'}{v}=\frac{\nu'}{\nu}\cdot\frac{D}{D'}. \tag{26}$$

At a temperature of 16° C. the kinematic viscosity of water is about one-thirteenth that of air; hence the corresponding speed in water would be about one-thirteenth of the speed in air provided the same nozzle is used so that $D'=D$. Under these corresponding conditions equation (9) shows that

$$\left(\frac{p}{\rho v^2}\right)_{\text{air}}=\left(\frac{p}{\rho v^2}\right)_{\text{water}} \tag{27}$$

from which

$$\frac{p}{p'}=\left(\frac{\rho}{\rho'}\right)\left(\frac{v}{v'}\right)^2. \tag{28}$$

Thus the differential pressure p which the nozzle would generate in air at a speed v can be predicted by observing the value p' realized in water at a speed v', provided the assumptions made are correct.

The most violent assumption made is that the effect of compressibility can be neglected in passing from such an incompressible fluid as water to such an easily compressible one as air. The value of E is about 20,000 times greater for water than for air. Thermodynamic theory suggests that the more readily compressible fluid should give the greater differential pressure at any one speed, and if the effect of compressibility is appreciable at all, it will be brought out in an exaggerated degree by the water channel experiments.

To test this assumption, observed values of $\frac{p}{\rho v^2}$ may be plotted as ordinates against $\frac{Dv\rho}{\mu}$ as abscissas, both for observations in air and in water. If the influence of compressibility is negligible the two curves, however irregular, ought to coincide. If they do not, then it is impossible to represent the results satisfactorily on a two-coordinate diagram; a third axis, for values of $\frac{C}{v}$, should be constructed and the data shown on a surface in space as implied by equation (11).

It turns out, as will be shown in detail later, that compressibility is practically but not wholly negligible. The agreement between wind tunnel and water channel observations is sufficient for predicting the order of magnitude of the differential pressure available from any tube of new design, and the water channel tests are also adequate for detecting small differences in the performance of nozzles produced in quantity from the same pattern.

If, however, the effect of varying the compressibility is found to be comparatively small when contrasting two media so different as air and water, then it will undoubtedly be negligible altogether where the use of the nozzle is confined to air alone. This is the most significant result of the water channel investigation, and warrants proceeding to the next series of experiments with attention directed to density and viscosity rather than to compressibility.

These next experiments were made in a wind stream at reduced pressure, the apparatus consisting of a small airtight tank, referred to as the vacuum wind tunnel. In this way air densities corresponding to various altitudes, and viscosities corresponding to various temperatures, could be realized in the laboratory. The results are plotted with $\frac{p}{\rho v^2}$ against $\frac{Dv\rho}{\mu}$ as before, since there is now no question of a third coordinate.

Clearly if the ρv^2 law holds, $\frac{p}{\rho v^2}$ will remain constant over the full range of conditions experienced; that is, the curve will be a horizontal straight line, parallel to the $\frac{Dv\rho}{\mu}$ axis. For a Pitot tube the ordinate of this line will be one-half, since $p = \frac{1}{2}\rho v^2$. For a Zahm nozzle performing in accordance with equation (3) the ordinate would be 3.2.

Any departure from horizontality not only signifies a departure from the ρv^2 law as regards the mathematical form in which ρ and v enter the law, but evidently also signifies that another physical quantity, viscosity, has begun to play a part. This complicates the altitude correction; for the viscosity as well as the density will be different at different atmospheric temperatures. It will be seen that the curve does have a considerable slope at low air speeds such as occur in the flight of dirigibles and the landing of airplanes, but not at the higher speeds.

This result was anticipated from the water channel experiments and also from another interesting circumstance. In developing the smaller modified Zahm nozzle for the Army it has been learned that it was not found possible to keep the new design geometrically similar to the original Zahm nozzle and still preserve the original calibration curve. This fact alone is evidence that viscosity makes a difference. For by (13) the size D can not enter unless the viscosity does also.

Having demonstrated the effect of viscosity and density under laboratory conditions, which had the advantage of direct control of the separate variables but the disadvantage of a restricted space not perfectly simulating free air conditions, it seemed worth while to proceed with airplane tests. This was done, and the results will be found plotted, as before, with $\frac{p}{\rho v^2}$ against $\frac{Dv\rho}{\mu}$. They agree qualitatively with the earlier laboratory results and afford more reliable numerical values, although not extending to such low densities.

Finally the results of ordinary wind tunnel tests on several different types of Venturi tubes are brought together for comparison. These too are reported by the dimensionless coordinate diagram, which is particularly well adapted for drawing inferences in regard to the altitude effect.

It was not feasible or necessary to make all the different kinds of tests on each of the tubes. The actual sequence followed is given below:

1. Water channel and wind tunnel tests on two French Venturi tubes, Badin type, one single and one double.
2. Vacuum wind tunnel tests on one American Pitot-Venturi tube, United States Army modified Zahm type.
3. Airplane flight tests on the foregoing Pitot-Venturi tube.
4. Study of ordinary wind tunnel data, taken at different times, on two of the foregoing tubes and on a French Pitot-Venturi, Toussaint Lepère type, and a German double Venturi, Bruhn type.

These tubes, together with the original Zahm nozzle, are shown in the photograph, figure 1.

REPORT No. 110.

THE ALTITUDE EFFECT ON AIR SPEED INDICATORS.

By M. D. Hersey, F. L. Hunt, and H. N. Eaton.

PART II.

EXPERIMENTS WITH VENTURI TUBES.

1. WATER CHANNEL EXPERIMENTS.

For the purpose of testing the two French Venturi tubes in water the 400-foot towing tank of the Bureau of Standards was placed at our disposal by Mr. W. F. Stutz, whose cooperation in this feature of the work is acknowledged. The nozzle under investigation was mounted about 40 cm. below the surface of the water on a rigid rod extending vertically down from the electric car which runs along a track over the tank. The speed of the car could be controlled and measured with an accuracy of the order of 1 or 2 per cent. The differential pressure was measured on a mercury manometer connected with the nozzle by water-filled

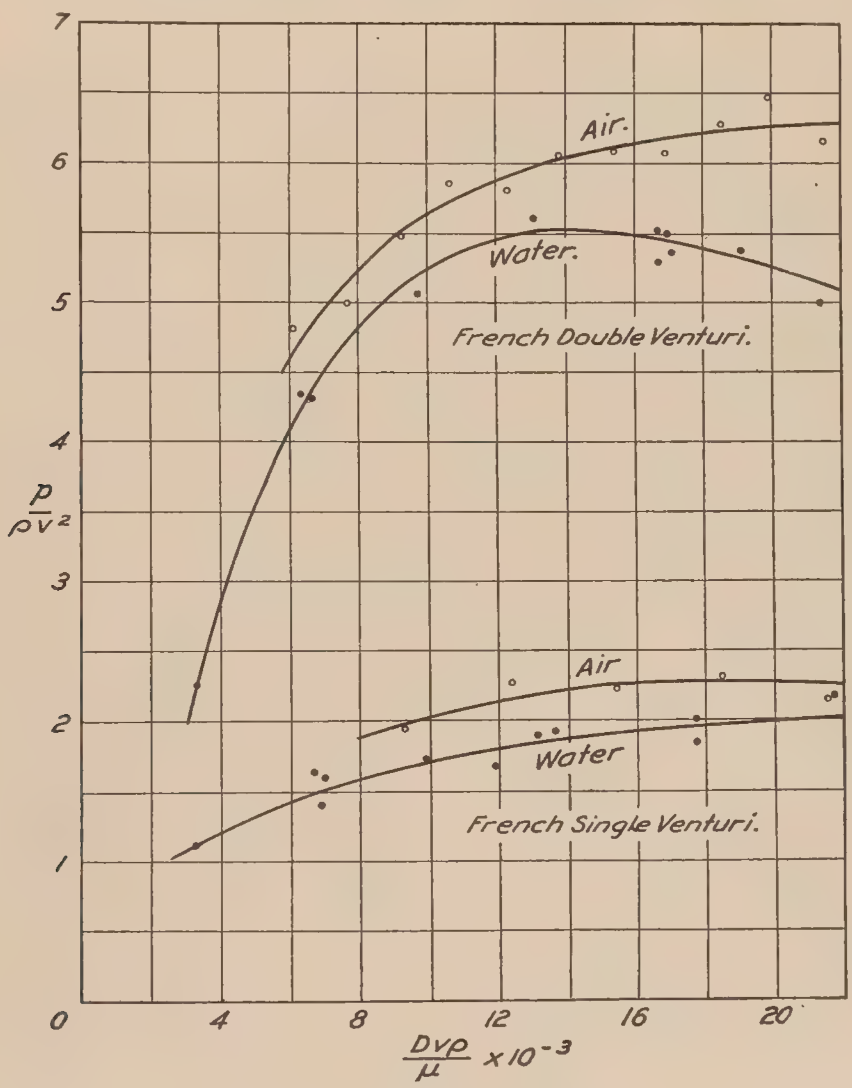


FIG. 2.—Water channel experiments.

tubing. At the high speeds and large suctions met in the last few observations on the double Venturi an error may exist due to time lag, of such a nature as to make the recorded values of the differential pressure too low. In those particular instances it was difficult to be sure that the manometer had risen to its maximum value in the short time interval available. Aside from this, it is thought unlikely that any important errors can have crept in. Due consideration was given to the necessity for avoiding turbulence and general movement of the water, and for keeping the depth of immersion sufficient to avoid surface disturbances.

Thus with a depth of immersion of 13 cm. one test at a speed of 4.77 miles per hour gave a differential pressure of 29.2 inches of water; a second test at the same speed gave 30.8 inches; while a third test at the same speed, upon increasing the depth to 40 cm., gave substantially the same result, 30.3 inches.

The following data were obtained for the French single Venturi (A, Fig. 1) in water and are plotted in fig. 2:

French single Venturi, in water.

Speed (miles per hour).	Differential pressure (inches water).	$\frac{p}{\rho v^2}$	$\frac{Dv\rho}{\mu}$
1.29	1.5	1.12	3,240
2.68	9.4	1.63	6,700
2.73	8.4	1.40	6,830
2.77	9.9	1.60	6,930
3.97	21.8	1.73	9,900
4.77	30.3	1.66	11,900
5.25	42.2	1.90	13,100
5.44	46.1	1.94	13,600
7.08	84.6	2.01	17,700
7.10	80.2	1.85	17,700
8.67	132.0	2.18	21,700

For convenience, the dimensionless variable $\frac{p}{\rho v^2}$ may be termed the *relative performance*, since it shows the ratio of the differential pressure generated by the nozzle in question, to twice that generated by a Pitot tube. Likewise $\frac{Dv\rho}{\mu}$ may be termed the *generalized speed*, since the speed factor v is commonly the most important of the four, and since any given percentage variation of D, ρ, or μ would have just the same effect on the relative performance as the corresponding variation of speed, which is ordinarily the easiest factor to vary. This variable, $\frac{Dv\rho}{\mu}$, occurs frequently in problems of fluid mechanics where it serves to measure the degree of turbulence in the fluid; hence it has been suggested by Dr. E. Buckingham that the term *turbulence variable* would be of some advantage.

In computing the relative performance and generalized speed the values for speed and for differential pressure given in the first two columns were changed over to c. g. s. units; an arbitrary linear dimension of 1 cm. was taken for D in all cases; the density of the water was taken to be $\rho=1$ gram/cm^3; and its viscosity, since the water was at the temperature of melting ice, was assumed to be $\mu=0.0179$ dyne-sec./cm^2. The values of the relative performance were thus found to range from about one to two (i. e., from twice to four times the differential pressure of a Pitot tube) and those of the generalized speed from about three thousand to twenty thousand. Since these variables are dimensionless, the same numerical values would prevail in any other system of normal units, such as the foot, pound-mass, second system; or the foot, pound-weight, second system.

A generalized speed of 20,000 units (with $D=1$ cm.) corresponds to about 66 miles per hour in air having the standard condition

$$\rho=1.221\times10^{-3}\ \text{g/cm.}^3$$
$$\mu=1.81\times10^{-4}\ \text{g/cm. sec.}$$

this last being the viscosity of air at the standard temperature, 16° C. It corresponds to about 130 miles per hour at 20,000 feet altitude, where the density is half as great if the temperature is unchanged.

The results on the double Venturi (B, fig. 1) were worked up in the same manner and are also plotted in figure 2. Solid black circles represent water channel observations; open circles are for the air observations, which will be described directly. In towing the double Venturi a depth of immersion of 60 cm. was maintained.

It is seen that the viscosity effect (slope of the curve) is more pronounced for the double tube than for the single one; a significant point when taken in conjunction with the fact that it has been the French practice to use the double tube on low speed and the single tube on high speed craft.

The foregoing water channel experiments were made in the winter of 1917–18 with the help of Mr. Bailey Townshend.

The same nozzles were given a wind tunnel calibration at the authors' request under the direction of Dr. A. F. Zahm at the Washington Navy Yard, leading to the results which are plotted for air in figure 2. To illustrate the method of reduction, the following table is given containing the data and results for the French single Venturi:

French single Venturi, in air.

Differential pressure (inches water).	Speed (miles per hour).	$\frac{p}{\rho v^2}$	$\frac{Dv\rho}{\mu}$
1.82	30	1.95	9,230
3.55	40	2.26	12,330
5.50	50	2.24	15,400
8.15	60	2.31	18,500
10.45	70	2.16	21,600

In computing $\frac{p}{\rho v^2}$ and $\frac{Dv\rho}{\mu}$ for air the values taken for density and viscosity were, in c. g. s. units, 1.223×10^{-3} and 1.78×10^{-4}, respectively. The former is the value used for standard density at the Washington Navy Yard tunnel; the latter is the viscosity of air at 10° C., which was assumed to be the actual air temperature. In computing ρv^2 it is not necessary to correct for the departure of actual density in the tunnel from standard density, since the data for speed are based on Pitot tube readings. The values of $\frac{Dv\rho}{\mu}$ on the other hand may be several per cent in error due to this cause, but this correction is not worth going into here, because the empirical values for $\frac{p}{\rho v^2}$ are not appreciably influenced by small changes of $\frac{Dv\rho}{\mu}$; the curves are nearly flat. As before D is arbitrarily taken equal to 1 cm.

The final results are plotted in figure 2, using open circles, and are seen to fall slightly higher up on the diagram than the solid circles for water.

The wind tunnel results on the double Venturi were computed and plotted in the same manner. Here the divergence between air and water results is quite pronounced at the high speed end of the range. This may in part be attributed to an experimental error in the water observations already mentioned; that error is probably negligible, but in the right direction to create the observed difference.

For both the single and double Venturi the relative performance in air is higher than in water; this presumably is due to the great difference in compressibility of the two media; the effect operates qualitatively in the direction suggested by thermodynamic reasoning, but is not so large as might have been expected.

The relative performance of the double Venturi is seen to be from two to three times that of the single tube; the viscosity effect, as judged from the slope of the curve, is also decidedly greater for the double Venturi.

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2. EXPERIMENTS AT REDUCED PRESSURE IN A WIND STREAM.

No doubt the most novel feature of the present investigation is the work at reduced pressure in a wind stream, by means of which the conditions at any altitude could be reproduced in the laboratory. This was done by means of a small vacuum wind tunnel, which was placed at the authors' disposal by Dr. H. C. Dickinson. The apparatus consisted of an air-tight iron tank, containing a high speed Sirocco blower and a wooden box with a working space for the wind stream 8 inches square and 2 feet long. The blower was mounted at the exit end of the channel and driven through a stuffing box by a motor outside. At the entrance end a honeycomb was constructed for the usual purpose, together with a piezometer for determining wind speed. The nozzle under test was mounted in the middle of the working space and connected to a water manometer in the room outside. The static connection of the piezometer unit was an annular series of holes in the walls of the wind channel near the grid of impact openings; a tube led from this point directly through the outer wall of the iron container to one side of a water manometer. The impact pressure grid is similarly connected to the other side of the same manometer. A mercurial barometer for determining the absolute pressure in the wind stream is connected to the static side of that manometer. The entire arrangement is sketched out in figure 3. Temperature was measured roughly by a thermometer located outside of the throat in the returning air stream and viewed through a glass window.

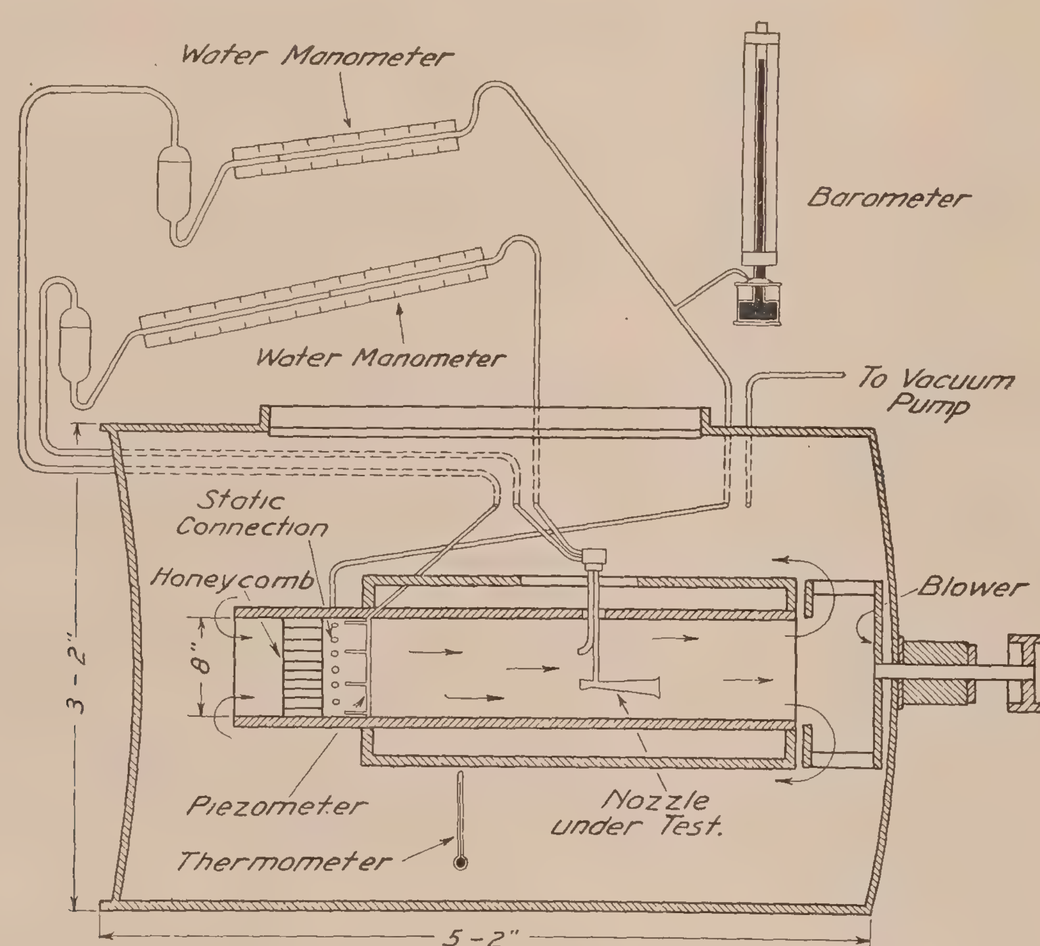


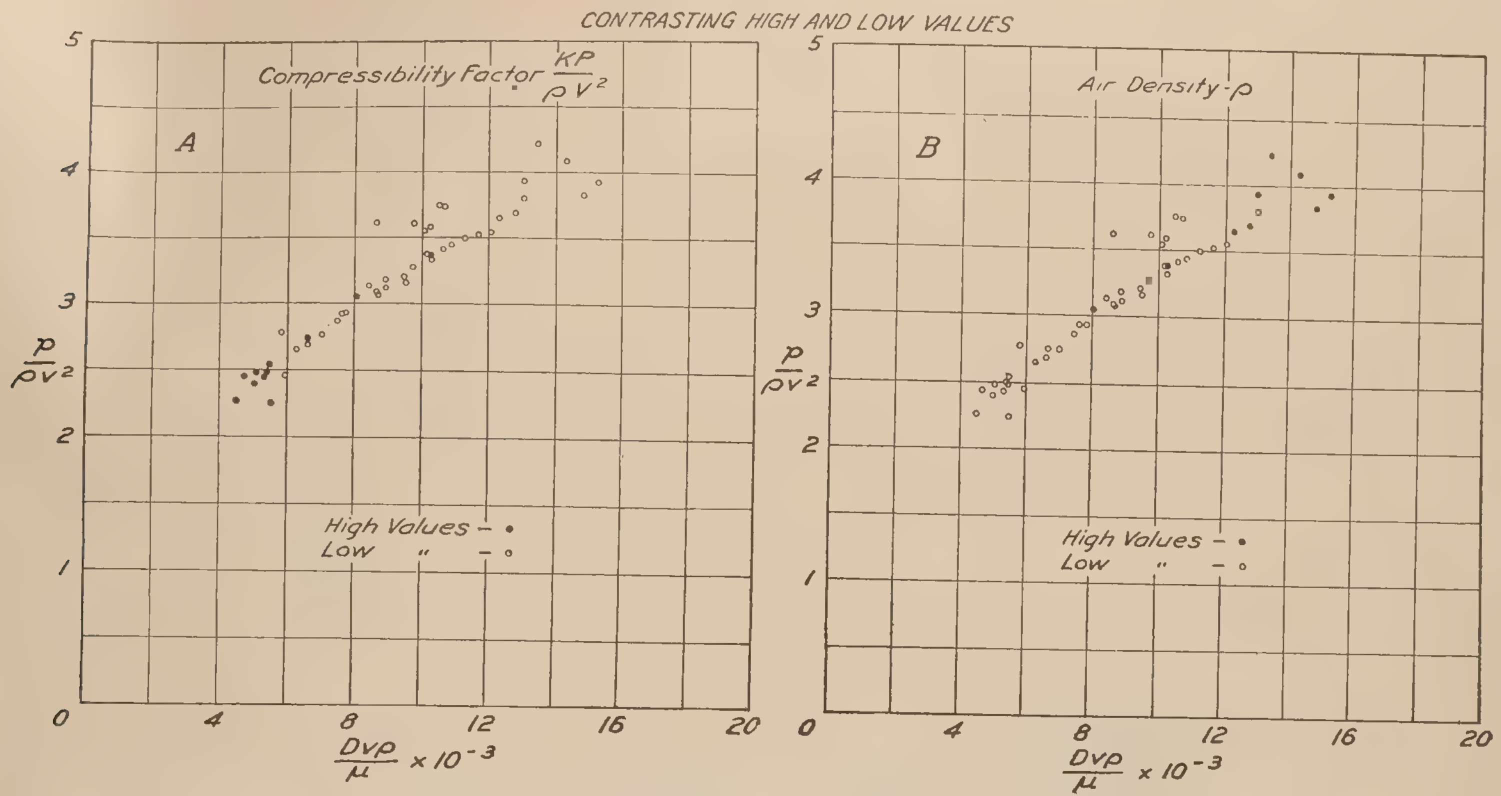
Fig. 3.—Vacuum wind tunnel.

In taking observations the usual procedure was to hold the internal pressure approximately constant by means of the vacuum pump while varying the speed of the blower step by step. This process would then be repeated at a different pressure. Artificial changes of temperature were not undertaken.

The working range of conditions covered was approximately as follows:

Air speed, from 30 to 65 miles per hour;
Pressure, 36 to 76 cms. of mercury;
Temperature, 20° to 28° C.

The final results of the vacuum wind tunnel experiment are plotted with the usual dimensionless variables in figure 4, A nd B. These two diagrams are numerically identical and have been repeated merely to avoid confusion in identifying some of the individual points.



Contrasting high and low values of compressibility factor $\frac{KP}{\rho V^2}$.

High values – • .
Low values – o .

Contrasting high and low values of air density ρ.

High values – • .
Low values – o .

FIG. 4—Vacuum wind tunnel results on United States Army Pitot-Venturi.

Thus in figure 4 A, a contrast has been indicated between points for which the compressibility factor $\frac{\kappa P}{\rho v^2}$ is high, and those for which it is low. Solid black circles are used for high values and open circles for low values. The relative significance of the black and white points is therefore the same in this diagram as it was in the plot for the water channel experiments, figure 2. Evidently there is no correlation between the relative performance and this compressibility variable. The same curve would result from either group of data.

As a matter of interest, although no longer a necessary logical step, figure 4 B has been constructed to show also that there is no correlation between relative performance and density, thus further substantiating the conclusion that the relative performance $\frac{p}{\rho v^2}$ depends solely on the generalized speed $\frac{Dv\rho}{\mu}$. In this diagram the black circles are for high values of the air density ρ, and open circles for low values. Evidently the same curve would be established even if the black points alone, or the white points alone, had been used.

The procedure for putting in the black circles, both in figure 4 A and figure 4 B, was simply to make note of the greatest and least numerical value of the quantity in question—$\frac{\kappa P}{\rho v^2}$, or ρ—and then divide the interval into two equal parts. It was not necessary to specially compute $\frac{\kappa P}{\rho v^2}$ because this is evidently proportional to the ratio of the mercurial barometer reading B to the Pitot pressure h_2 computed from the piezometer reading; actually, this ratio was used instead.

The data from which figure 4 was plotted are given in the following table:

Vacuum wind tunnel data.

1	2	3	4	5	6	7	8	9	10
Run No.	Temp. ° C.	Barom. B mm. merc.	Piezom. h_1 cm. water.	Pit. Vent. h_4 cm. water.	$\frac{Dv\rho}{\mu}\times10^{-3}$.	Pitot h_2 cm. water.	$\frac{p}{\rho v^2}$	Compress. factor $\frac{B}{h_2}$	Relative density $\frac{\rho}{\rho_0}$
1	20	430	1.30	13.20	8.61	1.83	3.62	235	0.559
	21	430	1.68	16.50	9.76	2.28	3.62	189	.557
	22	428	2.04	20.30	10.70	2.72	3.74	158	.551
2	23	425	0.54	4.32	5.47	0.87	2.48	489	.546
		411	0.57	4.65	5.51	0.92	2.55	449	.527
		407	1.52	13.45	8.93	2.12	3.18	192	.523
		406	1.96	17.73	10.15	2.63	3.38	154	.521
		405	2.10	20.95	10.50	2.80	3.75	145	.519
3	24	393	0.52	4.20	5.13	0.85	2.48	461	.503
		391	0.98	8.13	7.03	1.48	2.77	265	.501
		390	1.49	12.95	8.66	2.09	3.09	186	.499
		390	1.88	16.75	9.75	2.56	3.28	152	.499
		390	2.08	19.95	10.25	2.79	3.58	140	.499
4	24	385	0.52	4.08	5.07	0.85	2.41	452	.492
		385	0.92	7.62	6.63	1.41	2.69	273	.492
		385	1.43	12.70	8.42	2.02	3.14	191	.492
		385	1.81	15.80	9.47	2.47	3.21	157	.492
		385	2.05	19.65	10.10	2.76	3.55	139	.492
5	25	415	0.56	4.45	5.43	0.90	2.50	462	.528
		432	1.06	9.14	7.62	1.56	2.93	277	.551
		489	1.81	16.49	10.62	2.41	3.42	203	.625
		539	1.97	18.29	11.70	2.59	3.53	207	.690
		589	2.01	19.18	12.30	2.62	3.65	225	.752
		643	2.07	21.00	13.00	2.68	3.93	239	.821
		707	2.26	23.90	14.30	2.93	4.09	242	.904
		754	2.28	22.60	14.85	2.95	3.83	255	.962
6	26	620	0.82	7.11	8.03	1.18	3.05	526	.788
		618	1.36	12.32	10.30	1.82	3.38	340	.786
		618	2.08	19.90	12.80	2.69	3.70	229	.786
		618	2.31	25.30	13.45	2.99	4.23	206	.786
		618	2.98	30.50	15.25	3.89	3.93	159	.786
7	28	535	0.66	5.59	6.64	1.02	2.75	528	.676
		535	1.18	10.50	8.90	1.65	3.19	324	.676
		534	1.78	16.25	10.90	2.36	3.46	226	.675
		535	2.17	20.15	12.10	2.84	3.55	189	.676
		531	2.55	25.15	13.05	3.31	3.81	160	.671
8	28	440	0.57	4.07	5.56	0.92	2.25	481	.556
		453	0.99	8.38	7.48	1.46	2.87	310	.570
9	27	436	0.53	4.19	5.38	0.86	2.44	506	.553
		440	1.44	12.58	8.94	2.01	3.13	218	.558
		440	1.91	17.25	10.30	2.57	3.37	172	.558
		440	2.31	21.20	11.30	3.04	3.50	145	.558
10	28	377	0.47	3.81	4.73	0.78	2.45	483	.477
		375	0.85	6.99	6.30	1.31	2.66	286	.475
		373	1.31	11.17	7.80	1.90	2.94	196	.471
		373	1.64	14.10	8.72	2.30	3.07	162	.471
		373	1.96	17.15	9.54	2.66	3.16	140	.471
11	28	369	0.45	3.43	4.54	0.75	2.27	491	.466
		367	0.78	6.10	5.98	1.24	2.47	296	.464
		399	1.68	14.72	5.86	2.66	2.78	150	.504
		408	2.08	18.55	10.30	2.79	3.33	149	.515

In this table columns 1, 2, and 3 are self-explanatory. The piezometer reading h_1 of column 4 is given in cms. of water by reducing the observations taken on an inclined manometer. The Pitot-Venturi head h_4 of column 5 was derived in the same manner. The particular tube investigated was United States Army modified Zahm nozzle No. 30. Values of the generalized speed, $\frac{Dv\rho}{\mu}$, with $D=1$ cm., were computed just as in the case of the French Venturi tubes of figure 2. For the viscosity of air at different temperatures, Sutherland's formula

$$\mu=\mu_o\left(\frac{1+\frac{K}{273}}{1+\frac{K}{\theta}}\right)\sqrt{\frac{\theta}{273}} \tag{29}$$

was referred to, with $\mu_o=17.3\times10^{-5}$ for the viscosity at 0° C in c. g. s. units, and with Sutherland's constant $K=119.4$, θ denoting absolute temperature in degrees centigrade.

The air speed v was obtained from the piezometer reading h_1 at any density ρ by the formula

$$v=C\sqrt{\frac{h_1}{\rho}} \tag{30}$$

in which the coefficient C has the approximate value 54.3. This value was found sufficiently close when determining v for the purpose of computing $\frac{Dv\rho}{\mu}$; but a more exact method was followed for determining v when computing $\frac{p}{\rho v^2}$. The coefficient C is not strictly a constant, but in fact a slowly varying function of $\frac{Dv\rho}{\mu}$, dropping off from about 57.6 to 50.5 while $\frac{Dv\rho}{\mu}$ increases from 5,000 to 20,000. It was determined experimentally by calibration against a Pitot tube in the vacuum wind tunnel, this Pitot tube in turn having been calibrated against a standard Pitot tube in the Bureau of Standards wind tunnel. The results of this experiment were plotted in the form of a curve with $\frac{h_2}{h_1}$ as ordinate against $\frac{Dv\rho}{\mu}$ ($D=1$ cm.) as abscissa where h_2 denotes the reading of a standard Pitot tube. This curve was well determined with a large number of points and gives the ratio needed for converting piezometer readings into the equivalent standard Pitot tube readings which are tabulated in column 7. Since by the standard Pitot tube formula the head in cms. of water is

$$h_2=\tfrac{1}{2}\,\rho\, v^2\times\frac{1}{980}$$

and since from (30) in the same units

$$h_1=\frac{\rho v^2}{C^2},$$

it follows that

$$\frac{h_2}{h_1}=\frac{C^2}{1960}$$

or

$$C=44.3\sqrt{\frac{h_2}{h_1}}$$

Thus the average value 1.5 observed experimentally for the ratio $\frac{h_2}{h_1}$ gives rise to the approximate value 54.3, mentioned above, for the coefficient C. Incidentally it is interesting to note the great difference between the performance constant of the piezometer and that of a Pitot tube, the Pitot head reading for a given air speed in the vacuum wind tunnel being 50 per cent greater than that of the piezometer. This is probably accounted for by the fact that the piezometer integrates the air flow over the cross section, while the Pitot reads the maximum velocity. The further fact that this ratio is not constant but decreases slightly for

increasing values of $\frac{Dv\rho}{\mu}$ suggests that the velocity distribution and state of turbulence in the vacuum wind tunnel will appreciably differ for different conditions.

Column 8 shows the relative performance $\frac{p}{\rho v^2}$ for the Pitot-Venturi tube. The pressure p is determined by converting the head h_4 given in column 5 into dynes per square cm. to correspond with the c. g. s. units employed throughout for the density and air speed. The air speed for this column is determined from Formula 30 by using the appropriate value of the coefficient C from the empirical curve. The compressibility factor $\frac{B}{h_2}$ in column 9 is, actually, in millimeters of mercury per centimeter of water. This factor is taken as a substitute for the quantity $\frac{\kappa P}{\rho v_2}$ to which it is proportional. If the ratio $\frac{B}{h_2}$ were given as a dimensionless ratio, for example cms. of water per cm. of water, the values in column 9 would be 1.36 times as large, showing that the impact pressure of the moving air stream in these experiments varied from about 1/650 to 1/165 of an atmosphere. The rarefaction in the throat of the Venturi is about five times as much.

Similarly the relative density $\frac{\rho}{\rho_0}$ is shown in column 10. This varied from about 0.44 up to about 0.91 taking for the standard density, as before, the value 1.221×10^{-3} gms./cm.^{3}, which corresponds to a barometer reading $B_0 = 760$ mm., and temperature 16° C. These values are computed by dividing the relative pressure $\frac{B}{B_0}$ by the relative absolute temperature $\frac{\theta}{\theta_0}$, in which $\theta_0 = 289°$ C. absolute.

Inspection of the final plot A or B, figure 4, shows a very pronounced slope, the observed data for relative performance starting far below the normal value 3.2 assumed in the specifications for the instrument, and rising to a value somewhat higher.

Realizing the difficulty of discovering what might happen on an airplane in free flight 10,000 feet above the earth, from observations conducted in a space 8 inches square by 2 feet long, it was originally expected to attach only qualitative significance to the results of the vacuum wind tunnel experiment.

Nevertheless it seemed worth while to determine in what respect the conditions of the experiment differed from the conditions of free flight, so as to judge in which direction, if at all, the observed data would be expected to deviate.

Aside from errors of observation, four fundamental circumstances are worth considering:

(1) The proximity of the walls of the channel to the instrument might disturb the flow; but it is difficult to judge whether this would increase or decrease the performance of the nozzle.

(2) The velocity distribution over the cross section might vary in such a way, when the density and speed of the air are changed, that the actual velocity in the neighborhood of the instrument orifices would fail to bear a constant ratio to the integrated velocity given by the piezometer. In the modified Zahm nozzle the Pitot opening and the upstream Venturi opening are separated by a transverse distance of several inches, or nearly half the diameter of the channel. At low densities or low speeds, where the medium is not so excessively turbulent, the velocity distribution might conceivably be more sharply parabolic than it would at the higher speeds and densities. In this event the relative performance of the nozzle would apparently increase with increasing values of speed and density. However, no direct evidence of an appreciable change in velocity distribution was detected during an extensive series of experiments in which the velocities at different points in the cross section were explored with a Pitot tube. Up to within one inch of either wall, the velocity at any point bore a practically constant ratio of 1.4 to the average integrated velocity. The velocities varied irregularly as much as 6 per cent or 7 per cent above or below this average ratio. This variation, if systematic, should from dimensional considerations be some function of $\frac{Dv\rho}{\mu}$; that is, practically, some function of the product $v\rho$; but the curves plotted in that manner did not show any systematic tendency.

(3) The unsteady state of the flow might give instrument readings perceptably different from those corresponding to a steady state. Some fluctuation of speed was unavoidable. Now it is commonly recognized that the mean reading of a Pitot tube acted on by a rapidly fluctuating current of air is higher than the value which would result from the same speed if actually steady; for the Pitot head is proportional to the square of the speed, but the square root of the mean value of v^2 is greater than the mean value of v itself. If the same reasoning is extended a step further it leads to the conclusion that the mean reading of an instrument actuated by a force proportional to some power of the speed higher than the second will be more greatly augmented by fluctuations than the corresponding reading of an instrument like the Pitot tube acted on by a force actually proportional to the second power of the speed. The plot of the vacuum wind tunnel results does show that the Pitot-Venturi tube generates a differential pressure proportional to a higher power of speed than the second. This explanation would lead us to expect abnormally high readings for the relative performance of the Pitot-Venturi tube as a result of the unsteady state, although it is not apparent just how large the effect would be, and it is doubtless small.

(4) A further source of explanation lies in the excessive degree of turbulence undoubtedly existing in this small apparatus, which was put together from available parts without the usual refinements of a larger wind tunnel. It seems possible that rotating elements of fluid going into the entrance cone of the Venturi might to some extent become straightened out, thus increasing the actual speed of air through the throat beyond the average linear speed of the approaching fluid. If this hypothesis is correct a relatively greater performance should be expected from the Venturi in a turbulent medium than in a uniform medium. Moreover the same condition of turbulence might diminish the piezometer reading due to the impact of eddies against the static openings, thus further accentuating the same effect. The hypothesis regarding turbulence was tested experimentally, to a limited degree, by repeating the observations on a Pitot-Venturi tube in an ordinary wind tunnel with and without a netting across the tunnel in the approaching air stream. A negative result was obtained. The netting appeared to have no effect. This test was not considered conclusive however, because the amount of turbulence created by the netting was probably trifling compared with that really existing in the vacuum wind tunnel.

At all events comparison of the vacuum wind tunnel results with the flight test results and ordinary wind tunnel data given below, indicates that the relative performance shown in figure 4 is numerically greater than would be the case in free flight, although qualitatively correct as regards the effect of variations in speed, density, and viscosity.

The vacuum wind tunnel experiments were carried on with the help of Mr. Howard O. Stearns, while the velocity distribution was observed by Mr. Atherton H. Mears and Mr. W. G. Brombacher.

3. AIRPLANE OBSERVATIONS.

Through courtesy of the Engineering Division of the Air Service at McCook Field, flight tests have been made by one of the authors on the same Pitot-Venturi tube tested in the vacuum wind tunnel. The results are plotted in figure 5, choosing the same variables and the same scale as before. High density points are shown by solid black circles, low density by open circles. The range of conditions experienced was approximately as follows:

Airspeed, from 56 to 126 miles per hour;
Pressure, 43 to 75 cms. of mercury;
Temperature, $-12°$ to $0°$ C.

Thus the densities are somewhat higher and the speeds about twice as high as in the vacuum wind tunnel, so that the range of variables hardly overlaps, although the plane was flown as slow as 56 miles per hour and at altitudes approximating 15,000 feet.

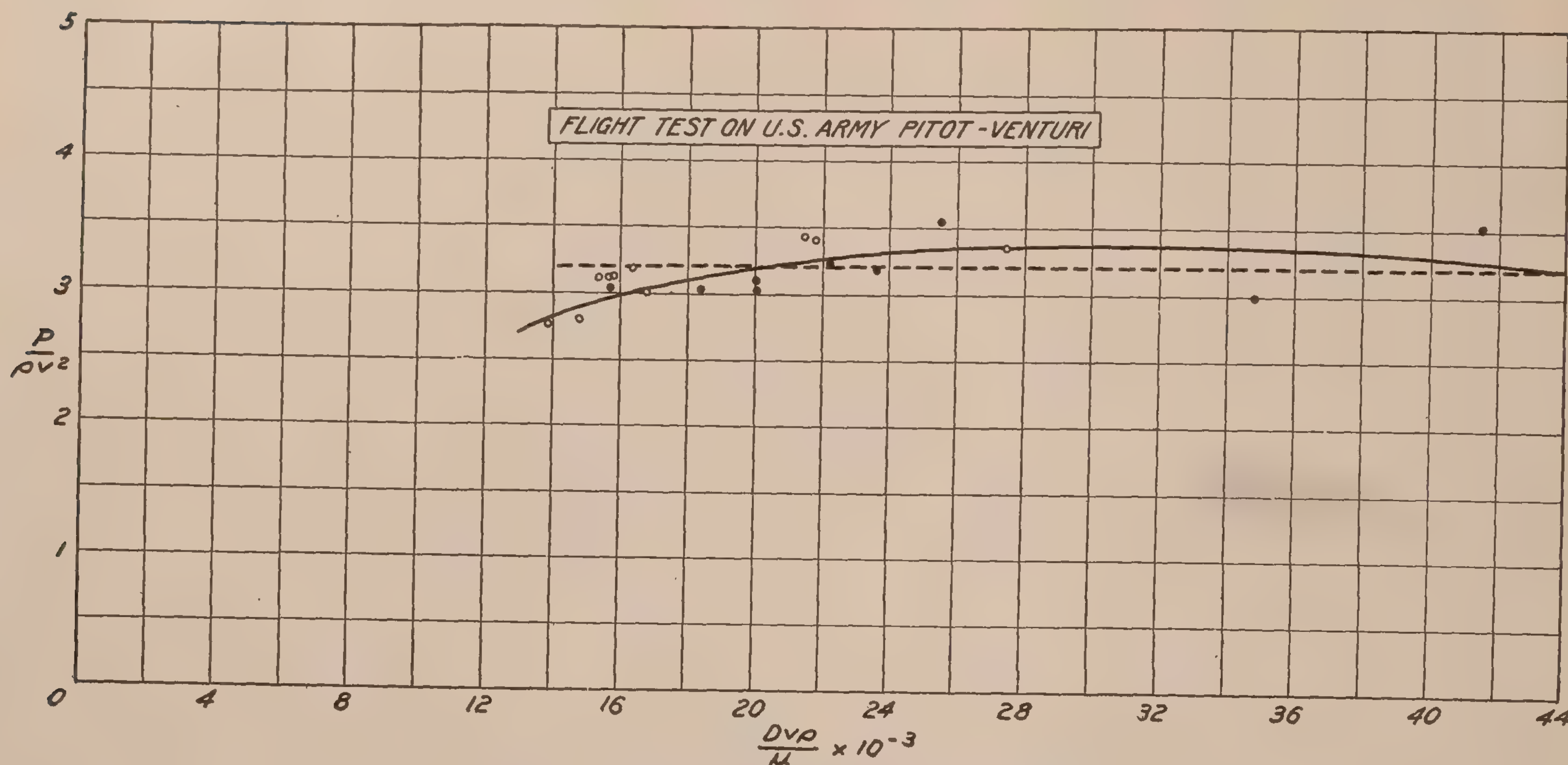


FIG. 5.—Flight test on United States Army Pitot-Venturi.

The data from which figure 5 was plotted are shown, somewhat abridged, in the following table:

Flight test data.

1	2		3	4	5	6	7
Elapsed time (minutes).	Indicated air speed (miles per hour).		$\frac{p}{\rho v^2}$.	Iso-thermal altitude (feet).	Temp. ° C.	Relative density $\frac{\rho}{\rho_0}$.	$\frac{Dv\rho}{\mu}\times10^{-3}$.
	V_1.	V_2.					
0	0	0	....	240	−8	....	....
2	71.4	71.4	3.18	1,340	−9	1.041	23.6
4	70.5	70.0	3.24	3,100	−6	.966	22.2
6	63.7	64.8	3.10	4,045	−5	.931	20.1
8	65.4	67.4	3.02	5,065	−2	.886	20.1
10	61.3	63.0	3.03	6,175	0	.853	18.4
15	57.8	59.6	3.00	8,545	0	.783	16.8
20	56.9	58.0	3.11	10,710	−4	.726	15.7
24	61.5	61.7	3.18	12,645	−7	.684	16.4
28	60.3	61.0	3.12	14,235	−8	.647	15.8
33	58.8	59.5	3.11	14,690	−11	.642	15.4
38	86.7	83.6	3.44	15,765	−12	.619	21.5
40	49.2	53.0	2.76	15,305	−12	.631	13.9
45	101.8	99.5	3.35	10,030	−5	.746	27.6
47	81.7	78.8	3.41	9,810	−4	.749	21.8
50	50.6	54.0	2.80	10,020	−4	.745	14.9
56	113.5	117.5	2.98	5,075	0	.879	34.8
58	90.4	86.0	3.55	4,850	0	.885	25.5
60	51.5	53.0	3.03	4,955	0	.883	15.7
65	131.2	125.0	3.51	500	−7	1.064	41.5

In this table the indicated air speeds V_1 and V_2 are the readings of a King and Munro air speed indicator, respectively, after purely instrumental corrections have been applied. These corrections were obtained after the flight by direct calibration against standard water columns graduated in miles per hour according to the Zahm and Pitot formulas, respectively. The King air speed indicator was a carefully selected instrument of the standard American Army pattern connected with the modified Zahm nozzle No. 30 under investigation. The Munro indicator was a suitable instrument of British make connected to an R. A. F. Pitot head. During

the calibration of the two instruments flight conditions were closely reproduced in the laboratory and corrections determined experimentally for the same readings and instrument temperatures experienced during the flight.

The Pitot-Venturi head was mounted on the left-hand outer strut of the airplane (DH-4), about one-third of the distance down from the upper plane. The Pitot head was mounted in a like position on the right-hand outer strut. The heads were carefully placed so that the distance down from the upper plane was the same in both cases, the object of the flight being to compare the performance of the Pitot-Venturi with a Pitot head rather than to determine absolute air speed. The altimeter and air speed indicators were mounted in a vertical position in the cockpit.

The indicated air speeds V_1 and V_2 are needed for determining the relative performance; the true air speed is computed from the Pitot reading V_2 by making due allowance for the decrease of density at different altitudes; and at the higher altitudes the true air speeds are about 20 per cent greater than the indicated values.

In computing relative performance $\frac{p}{\rho v^2}$, in which p denotes the differential pressure of the Pitot-Venturi nozzle under investigation, ρ the air density and v the true air speed, use is made of the fact that the differential pressure generated by a Pitot tube is one-half ρv^2. The numerical value needed for the relative performance in column 3 is therefore simply one-half the ratio of the head generated by the Pitot-Venturi nozzle to that of a Pitot tube. From the standard formulas

$$h_1=\left(\frac{V_1}{17.88}\right)^2$$

for the Zahm nozzle, and

$$h_2=\left(\frac{V_2}{45.2}\right)^2$$

for the Pitot, it is seen that

$$\frac{p}{\rho v^2}=3.2\left(\frac{V_1}{V_2}\right)^2$$

The isothermal altitude given in column 4 is the ordinary 10° C. altimeter reading corrected for purely instrumental errors by subsequent laboratory comparison under the same pressures and temperatures experienced in flight. The altimeter was set to read 200 feet at the start of the flight, this being approximately the 10° C. altitude corresponding to the actual barometric pressure at the ground.

The air temperatures given in column 5 were observed with a large strut thermometer.

The relative density in column 6 gives as before the ratio of the actual density of the atmosphere to the standard value 1.221×10^{-3} gms./cm.^{3} The density is figured as before from the barometric pressure and absolute temperature, the pressure in turn being derived from the isothermal altitude of column 4 by reference to the standard 10° C. pressure-altitude table.

Finally, in column 7 values of the generalized speed $\frac{Dv\rho}{\mu}$ are computed (taking the arbitrary linear dimension $D=1$ cm. as before) by reference to the formula

$$v=\frac{44.7\,V_2}{\sqrt{r}}$$

in which r denotes the relative density $\frac{\rho}{\rho_0}$; and by taking the viscosity from Sutherland's formula, as in previous computations. As seen from inspection of the table, the procedure during the flight was to secure a wide variation of speed by diving and straightening out at several different altitudes.

The final plot, as indicated before, shows qualitative agreement with the vacuum wind tunnel observations at the lower speeds and densities while approaching the normal value 3.2 for relative performance (as assumed in the instrument specifications) at the higher speeds and densities; or, strictly speaking, at the higher values of the generalized speed, $\frac{Dv\rho}{\mu}$. Practi-

cally the same curve would result had either the high density or low density observations been taken by themselves.

4. ORDINARY WIND TUNNEL DATA.

The same United States Army Pitot-Venturi tube (No. 30) investigated in the vacuum wind tunnel and in free flight had been tested in the 3-foot wind tunnel of the Bureau of Standards. For the purpose of a check on the foregoing experiments the ordinary wind tunnel results, furnished through courtesy of Dr. Lyman J. Briggs, were now recomputed in dimensionless coordinates and have been plotted in figure 6. The results are in close agreement with the flight test and in qualitative agreement with the vacuum wind tunnel results, showing a gradual but pronounced falling off in relative performance toward the lower values of generalized speed.

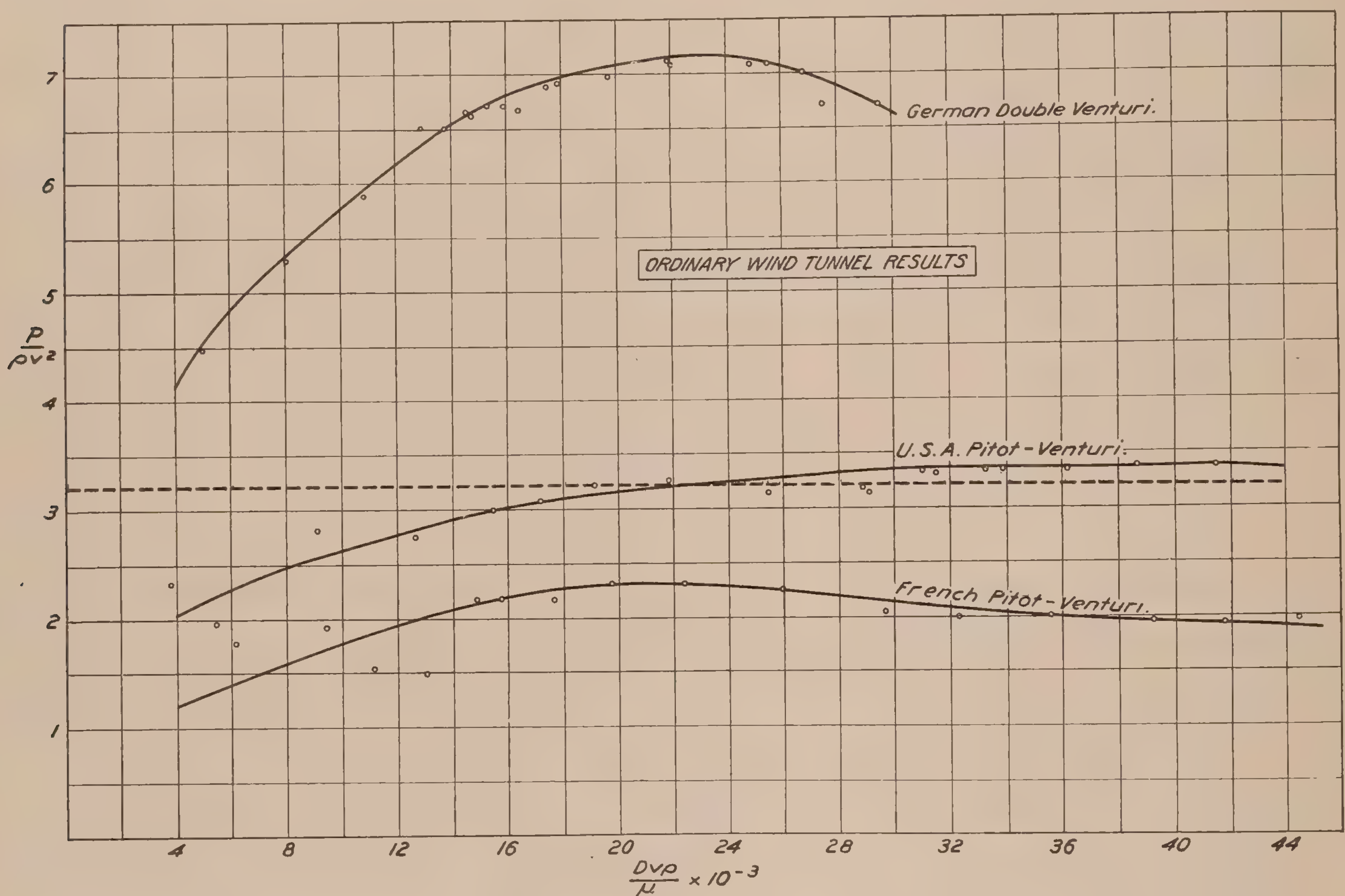


FIG. 6.—Ordinary wind tunnel results.

It is of interest to note the applicability of the dimensionless coordinate diagram to ordinary wind tunnel data with the consequent possibility of inferences regarding the altitude effect, or change in performance at the reduced densities and varying viscosities which may be met at different altitudes. On this account the method has been extended to two other well-known air speed nozzles, the French Pitot-Venturi (Toussaint-Lepère type) and a German double Venturi (Bruhn type), which were tested in the Bureau wind tunnel for the purpose of this investigation. These results also are plotted in figure 6. In all cases the scattering of observations for low values of generalized speed is undoubtedly accidental, due to the very small heads available at the water column under those conditions.

The data for these wind tunnel tests are given in the three accompanying tables. The indicated air speed values in the first column, obtained from the readings of an inclined manometer, afford data for the actual Pitot heads by means of the usual formulas. By comparing this Pitot head with the observed head on the nozzle under test the values of relative performance are derived, and the generalized speed is computed as before.

Ordinary wind tunnel data, United States Army Pitot-Venturi.

Indicated air speed, m. p. h.	Pitot-Venturi head, cm. water.	$\frac{p}{\rho v^2}$	$\frac{Dv\rho}{\mu} \times 10^{-3}$
13.2	1.0	2.32	3.84
19.3	1.8	1.95	5.57
31.4	6.1	2.80	9.15
43.4	12.9	2.74	12.7
53.1	21.1	2.99	15.5
59.0	26.6	3.07	17.2
65.4	34.2	3.21	19.1
75.4	46.1	3.26	21.9
88.1	61.1	3.14	25.7
99.7	77.4	3.14	29.1
108.0	96.0	3.30	31.5
117.6	115.1	3.34	33.9
88.4	60.8	3.14	25.5
100.0	79.4	3.18	28.9
107.2	95.1	3.31	31.0
115.3	110.3	3.32	33.3
125.5	130.9	3.34	36.2
134.0	151.3	3.37	38.7
143.7	173.6	3.38	41.5

Observations.	Approximate temperature, ° C.	Barometer, mm.
First 11	22	747
Last 8	24	747

Ordinary wind tunnel data, French Pitot-Venturi.

Indicated air speed, m. p. h.	Venturi head, cm. water.	$\frac{p}{\rho v^2}$	$\frac{Dv\rho}{\mu} \times 10^{-3}$
21.3	2.0	1.76	6.20
32.7	5.1	1.91	9.53
38.3	5.6	1.53	11.2
45.0	7.5	1.48	13.1
50.8	14.0	2.16	14.9
54.3	16.0	2.16	15.8
60.9	19.9	2.15	17.7
67.9	26.6	2.30	19.8
76.6	33.4	2.28	22.4
89.1	44.6	2.24	26.0
101.9	52.7	2.03	29.7
110.4	60.2	1.98	32.3
121.8	73.6	1.99	35.6
134.9	88.7	1.95	39.3
143.2	99.6	1.94	41.8
152.9	115.2	1.96	44.5

Observations.	Approximate temperature, ° C.	Barometer, mm.
First 8	22	747
Next 5	23	747
Last 3	24	747

Ordinary wind tunnel data, German double venturi.

Indicated air speed, m. p. h.	Venturi head, cm. water.	$\frac{p}{\rho v^2}$	$\frac{Dv\rho}{\mu}\times 10^{-3}$
17.7	3.5	4.47	5.10
28.1	10.5	5.26	8.10
37.6	20.7	5.87	10.9
44.2	31.7	6.5	12.8
47.3	36.2	6.50	13.7
50.1	41.5	6.66	14.5
51.0	43.0	6.60	14.7
53.0	47.1	6.70	15.3
54.9	50.5	6.70	15.8
56.6	53.0	6.66	16.4
60.4	62.6	6.87	17.4
61.3	65.0	6.90	17.8
67.3	87.3	7.70	19.4
67.9	80.3	6.97	19.6
76.0	102.7	7.10	21.9
76.2	102.5	7.07	22.0
86.5	131.8	7.07	24.8
88.2	137.8	7.07	25.5
92.4	150.2	7.00	26.7
101.6	172.5	6.70	29.4

Observations.	Approximate temperature, °C.	Barometer, mm.
1, 2	23	747.3
3, 4, 6, 8, 9, 11, 13, 15	24	747.3
5, 7, 10, 12, 14, 16–20	25	747.3

In this connection the statement made earlier may be recalled with regard to the numerical relation of the generalized speed scale to actual air speed under sea-level conditions; namely, that a generalized speed of 20,000 units, with $D=1$ cm. (approximately the middle of th range)

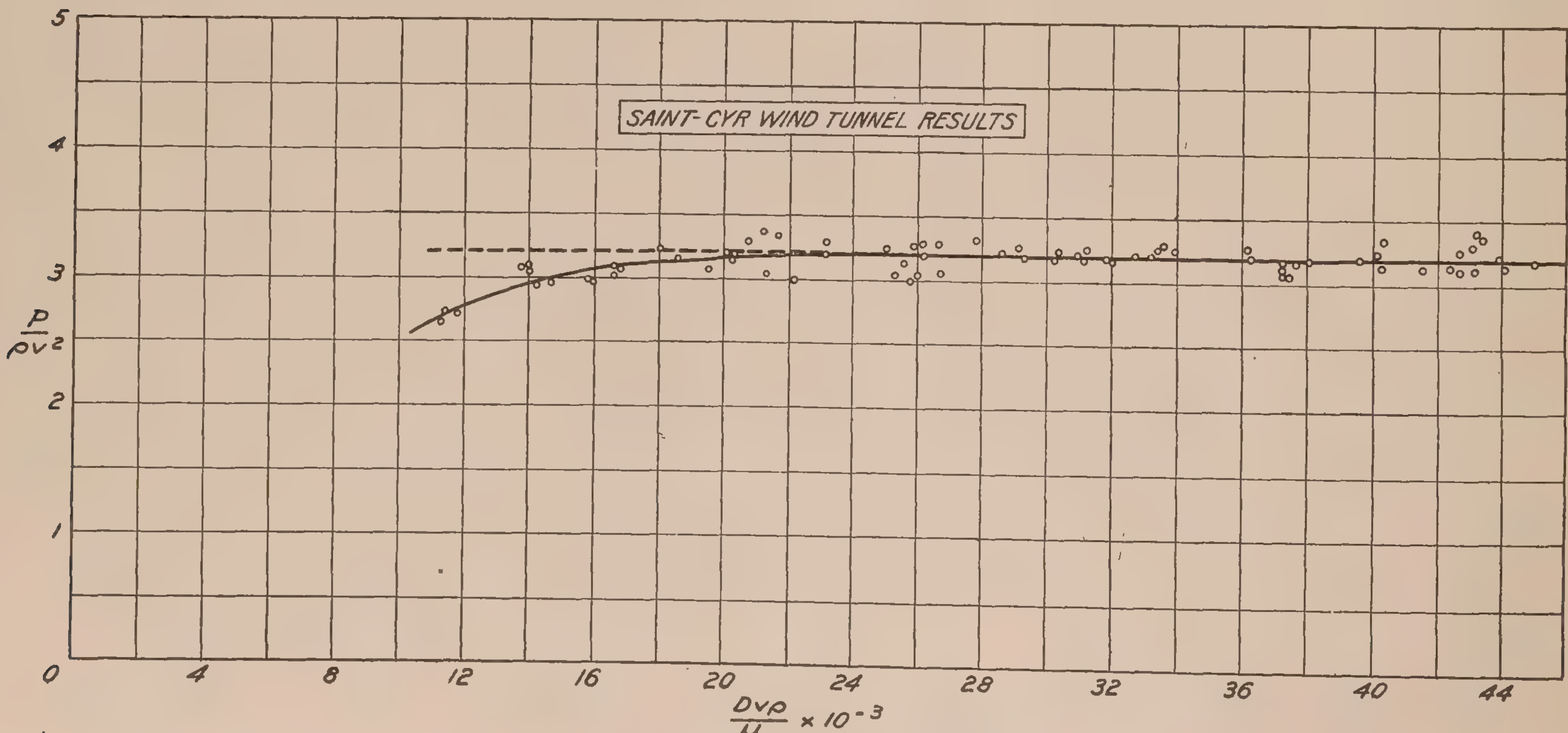


FIG. 7.—Saint-Cyr wind tunnel results.

corresponds to about 66 miles per hour in a sea-level atmosphere. Thus landing speeds, stalling speeds, and speeds of interest in dirigible work lie to the left of the middle, while ordinary airplane speeds are well to the right in figure 6 and similar diagrams.

In connection with the ordinary wind tunnel results, observations which were made in the small instrument wind tunnel at the Aerotechnic Institute at St. Cyr will be of interest, and have been plotted in figure 7. This diagram shows the results of two tests made by Lieut. John A. C. Warner and one of the present authors in November, 1918, on another sample of the small United States Army type Pitot-Venturi tube (No. 23). This investigation was made under the general direction of Maj. George M. Brett, Chief, Airplane Instrument and Testing Division, A. E. F. Air Service, through courtesy of the French Section Technique.

To illustrate the procedure it is sufficient to give the data from one of the two tests, which will be found in the following table:

St. Cyr wind tunnel data, United States Army Pitot-Venturi.

Temperature, 11° C; barometer, 744.5 mm.

Pitot h_1, cm. water.	Pitot-Venturi h_2, cm. water.	$\frac{p}{\rho v^2}$	$\frac{Dv\rho}{\mu}\times 10^{-3}$
2.00	10.7	2.68	11.8
1.85	9.7	2.62	11.3
3.95	23.7	3.00	16.6
4.05	24.8	3.06	16.8
3.70	21.8	2.95	16.0
6.6	40.0	3.03	21.4
6.5	38.9	2.99	22.2
9.6	57.4	2.99	25.8
9.7	58.7	3.02	26.0
10.1	60.7	3.01	26.6
13.2	84.1	3.19	30.3
13.8	88.4	3.20	31.0
13.9	88.1	3.19	31.1
18.8	122.7	3.27	36.2
19.0	121.5	3.20	36.3
20.0	126.7	3.17	37.3
20.4	128.7	3.16	37.7
22.5	144.2	3.21	39.6
23.4	147.0	3.14	40.3
20.8	132.7	3.19	38.1
26.9	168.6	3.14	43.2
26.2	164.5	3.14	42.7
28.0	176.5	3.15	44.1
29.0	184.7	3.18	45.0
25.0	156.7	3.13	41.6
25.8	162.7	3.15	42.4
14.6	93.1	3.19	31.9
14.7	93.1	3.17	32.0
9.2	54.9	3.02	25.3
9.4	56.7	3.12	25.6
5.8	35.7	3.20	20.1
5.5	33.7	3.06	19.6
3.6	21.4	2.97	15.8
3.1	18.3	2.95	14.7

The Pitot head h_1 given by the first column in cm. of water, when compared with the observed head on the Pitot-Venturi in the second column, affords values for the relative performance reported in the third column. In computing the $\frac{Dv\rho}{\mu}$, density and viscosity are derived as usual from the barometer and thermometer readings. The true air speed v was obtained from the Pitot reading by reference to a calibration curve furnished by the Section Technique, showing the result of a comparison between the St. Cyr Pitot and the French standard Pitot at the Eiffel Laboratory.

As in the previous experiment, the relative performance approaches quite closely to the numerical value 3.2 for high values of $\frac{Dv\rho}{\mu}$, but falls off gradually at the lower values.

5. GRAPHICAL COMPARISON OF RESULTS.

The results previously discussed for the performance of five different types of air-speed nozzles in air are brought together for convenient comparison in figure 8. To avoid confusion, the plotted points are left out but all may be seen upon consulting the previous diagrams. The curve shown for the United States Army modified Zahm type of Pitot-Venturi is an average of the flight test and ordinary wind-tunnel results for No. 30. It is seen to agree very closely with the St. Cyr test on the other nozzle, No. 23.

These curves will not be further discussed in the present paper, but evidently merit careful examination by those interested in the details of performance of the various types of air-speed indicator, and provide the necessary experimental basis for inferences regarding the altitude effect.

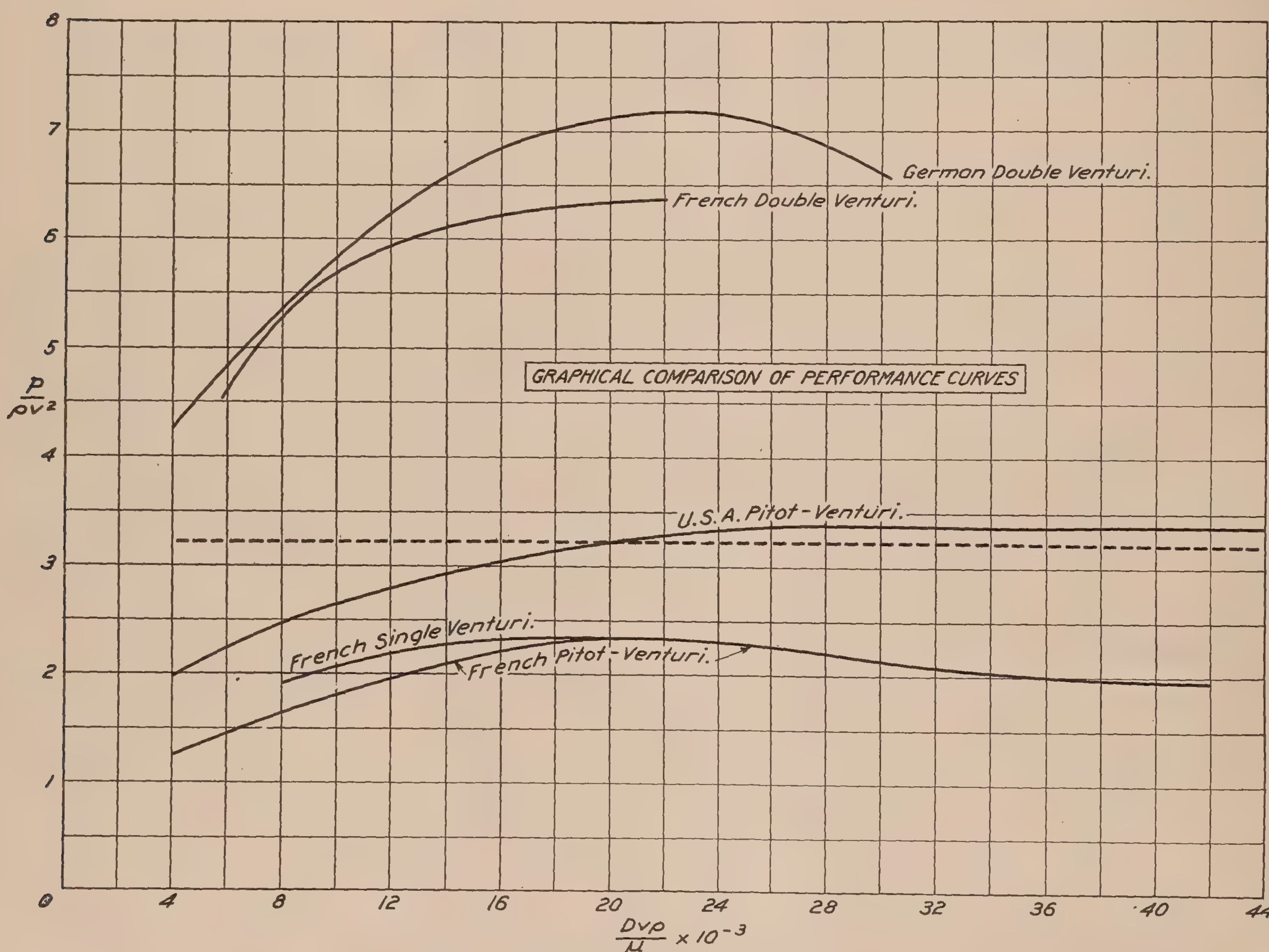


FIG. 8.—Graphical comparison of performance curves.

6. SUGGESTIONS FOR FURTHER INVESTIGATION.

In conclusion, it is suggested that convenient graphical or analytical methods should be developed for computing the altitude correction in practical problems from empirical data such as are afforded by figure 8. Moreover, for the purpose of securing the most exact numerical values, the vacuum wind-tunnel work might well be continued with the use of improved facilities.

Such a program has in fact been begun. A new tunnel with a working space nearly four times the cross section of the previous one is under construction; it is intended to install this tunnel in one of the large altitude chambers of the Bureau of Standards, in which both the temperature and pressure can be controlled over a wider range than before. Finally, additional types of air speed indicator other than Venturi tube should be investigated, verifying the observations with reference to several sample instruments of each type. The laboratory experiments should also be closely paralleled, so far as practicable, by observations taken at low speeds in lighter-than-air craft, and at high altitudes in airplanes. This investigation of the altitude effect is primarily of importance in connection with low-speed or high-altitude flight; for the altitude correction under the conditions of high-speed flight near sea level is sufficiently well given for most instruments by the simple ρv^2 law.

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